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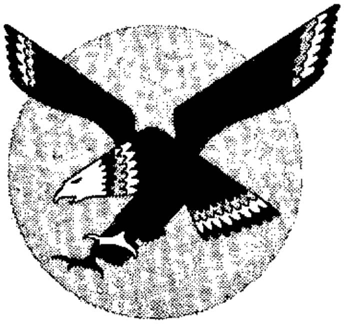

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A DIVISION OF GENERAL DYNAMICS CORPORATION

SAN DIEGO

REPORT EFT-P-8-711
DATE 1 July 1956
MODEL F-102

TITLE

ENGINEERING FLIGHT TESTS
YF & F-102A
CROSS REFERENCE

1 July 1956

PREPARED BY J. Specht
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GROUP Engineering Flight Test Planning

REFERENCE

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INTRODUCTION

This report is an Engineering Flight Test subject breakdown of YF-102's and F-102A airplanes. The Engineering Flight Tests and Reports have been catalogued under the index on the next page.

Test status and report entries will be revised monthly. Data for this report will be obtained from the Flight Test Status Index also revised monthly.

The outline contained in this report is not intended to be used as an official F-102 airplane series outline. However, the outline is applicable to the YF and F-102A programs as a guide to the test categories in which testing has been accomplished or is scheduled to be accomplished.

FLIGHT TEST OUTLINE*

1. Performance
   - 1.1 Performance Instrumentation Calibration
   - 1.2 Power Plant Characteristics
   - 1.3 Lift-Drag Characteristics
   - 1.4 Mission Capabilities
   - 1.5 Take-off and Landing
   - 1.6 Minimum Speeds

2. Flying Qualities
   - 2.1 Longitudinal Stability and Control
   - 2.2 Lateral-Directional Stability and Control
   - 2.3 General Control and Trimmability
   - 2.4 Minimum Speed
   - 2.5 Structural - Aeroelastics, Buffet, and Loads

3. Aircraft Systems
   - 3.1 Power Plant
   - 3.2 Fuel
   - 3.3 Flight Controls
   - 3.4 Hydraulics
   - 3.5 Pneumatic
   - 3.6 Electrical
   - 3.7 Air Conditioning
   - 3.8 Pilot's Cockpit
   - 3.9 Landing Gear
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4. Auxiliary
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   - 4.2 Armament
   - 4.3 Armament Fire Control
   - 4.4 Oxygen
   - 4.5 Test Instrumentation
   - 4.6 Fire Detection
   - 4.7 Safety Equipment
   - 4.8 Ground Support Equipment

5. Human Factors
   - 5.1 Working Environment
   - 5.2 Visibility
   - 5.3 Man Machine Compatibility

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*EFT-P-8-44

YF&F-10[illegible]A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO.	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
.1	Flight Instrumentation	1787	ZC-8-118	3.0		EFT-P-8-74	Alt.Airspeed Sys. Calib.			X			ZC-8-512-46
.1	" "	"	"	3.01			Airspeed Calib.-Pacer			X			EF-8-465
.1	" "	"	"	3.02			Airspeed Calib.-Fly-by			X			ZC-8-511-1
.1	" "	"	"	2.04		EF-8-519	Rev.of Aft Fus.Temp.Instr			X			303-2-011
.1	" "	"	"	3.02		EFTP-8-511	Airspeed Calibration		X				
.1	Flight Instrumentation	1788	ZC-8-119	6		EF-8-704	Angle of Attack Meas.			X			EF-8A-3 Inconclusive
.1	" "	"	"	6		EFTP-8-22	Angle of Attack Calib.	X					
.1	Flight Instrumentation	1791	ZC-8-120	3.00		EFT-P-8-65	Determination of Time Lag			X			
.1	" "	"	"			EF-8-723	Airspeed Calib.-Points				X		
.1	" "	"		3.01			Airspeed Calib.-Pacer			X			
.1	" "	"	"	3.02		EF-8-806	Stab.& Control Inst.Chk			X			
.1	Flight Instrumentation	1798	ZC-8-167		1		Test Airspeed Calib.			X			
.1	" "	"	"		1	EFTP-8-219	Alt.Static Source Invest.			X			
.1	" "	"	"			EFTP-8-600	Low Speed A/S Sys. Calib.			X			
.1	" "	"	"		3		O.A.T. Calibration			X			
.1	Flight Instrumentation	1799	ZC-8-143		1d	EFTI-8-51	Inst.& Chkout Test Instr.			X			
.1	" "	"	"		1f	EFTI-8-43	Inst.Strike Camera,Align & Oper.			X			
.1	" "	"	"		1j	EFTP-8-278	Airspeed Calib. & Air Data Computer Check			X			ZC-8-512-60

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO.*	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
.1	Flight Instrumentation	1351	ZC-8-140	3.01			Airspeed Calibration			X			
.1	" "	"	"	3.01		EFTP-8-181	Airspeed Calibration			X			ZC-8-512-60
.1	" "	"	"	3.02			Angle of Attack Calib.	X					
.1	Flight Instrumentation	✓7994	ZC-8-101	4			Airspeed Calibration			X			
.1	" "	"	"	5			Angle of Attack & Yaw						
							Pickup Calib.				X		
.1	Flight Instrumentation	✓7995	ZC-8-103	10			Airspeed Calibration			X			ZC-8-501-3-4-8-10-12-18-20
													EF-8-501-SR2
.1	" "	"	"	12			Angle of Attack & Yaw Cal.					1780	
.1	" "	"	"		6		Airspeed Calib.-Transonci					X	Superceded by Addendum 20
.1	" "	"	"		20		Airspeed Calibration			X			ZC-8-501-18,-20 EF-8-318
1.1	Flight Instrumentation	1779	ZC-8-108	1			Airspeed Calib. Subsonic			X			ZC-8-502-1
1.1	" "	"	"	2			Airspeed Cal. Supersonic				X		Not Accomplished
1.1	" "	"	"	3			Angle of Attack & Yaw Cal.					1780	EF-8-294
1.1	Flight Instrumentation	1780	ZC-8-111	10			Angle of Attack & Yaw						
							Pickup Calib.			X			ZC-8-513
1.1	" "	"	"	10		EF-8-314	Angle of Attack & Yaw Cal.			X			
1.1	" "	"	"	10		EF-8-330	Angle of Attack & Yaw						
							Determination			X			EF-8-532
1.1	" "	"	"	10			Calib. Proc. for Angle						
							of Attack				X		
1.1	" "	"	"	12		EF-8-609	Production OAT Survey			X			ZC-8-503-3-4 EF-8-609

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
.1	Flight Instrumentation	1780	ZC-8-111	Misc. 1		EF-8-332	Tape Recorder			X			
.1	" "	"	"	" 3			I.L.S. Tests			X			EF-8-632
.1	Flight Instrumentation	1781	ZC-8-112	8			Airspeed System Calib.					1779	
.1	Flight Instrumentation	1783	ZC-8-114	1			Angle of Attack & Yaw Pickup Calib.				X		
.1	" "	"	"	2		EF-8-463	Airspeed Calib.Subsonic			X			ZC-8-506-9 EF-8-463
.1	" "	"	"	3		EF-8-463	OAT Pickup & Location Study			X			ZC-8-506-7
.1	Flight Characteristics	7994	ZC-8-101	9			Prelim.Flight Character. Test						EF-8-84 F.T. SummaryFlt.1-6
.1	Flight Characteristics	7995	ZC-8-103	11			Prelim.Flight Character.			X			EF-8-442,-416,-614
.4	Climb	1787	ZC-8-118	4.02			Climb Performance			X			ZC-8-511-1 EF-8-454 EFT-A-8-17

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO.*	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
1.4	Climb	1791	ZC-8-120	5.02			Climb			X			6 Flights Required
1.4	Climb	1351	ZC-8-140	4.02			Climb			X			ZC-8-512-55
1.4	Dive	1787	ZC-8-118	4.04			Dive				X		
1.4	Dive	1791	ZC-8-120	5.04			Dive		X				
1.4	Dive	1351	ZC-8-140	4.04			High Speed Dive		X				
1.4	Dive	"	"	8.02			Hi Speed Flt. Req. (Dive)			X			

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
1.5	Take Off & Landing	1791	ZC-8-120	1.05			High Speed Taxi & Lift Off Characteristics			X			
1.5	" "	"	"	5.01			Take Off			X			
1.5	" "	"	"	5.05			Landing			X			
1.5	Take Off & Landing	1798	ZC-8-167		5		Drag Chute Deploy. Time		X				
1.5	Take Off & Landing	1351	ZC-8-140	1.04			Low Speed Taxi			X			
1.5	" "	"	"	1.05			Hi Speed Taxi & Lift Off			X			
1.5	" "	"	"	4.01			Take Off		X				
1.5	" "	"	"	4.05			Landing	X					
1.1	Performance	7994	ZC-8-101	7			Prelim. A/C Performance					7995	
1.1	Performance	7995	ZC-8-103	13			Prelim. A/C Performance			X			ZC-8-501-7-10-21 EF-8-354 EF-8-429
1.1	"	"	"	16			Airplane Performance			X			EF-8-402 EF-8-245

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
1.5	Take Off & Landing	7995	ZC-8-103	16		EF-8-595	Landing Loads & Distances			X			ZC-8-501-2, -3, -8, -12
1.5	Take Off & Landing	1780	ZC-8-111		6		Nose Gear Lift Off Speed				X		Design Change
1.5	Take Off & Landing	1787	ZC-8-118	1.04			Low Speed Taxi & Nose Wheel Steering			X			ZC-8-511-1
1.5	" "	"	"	1.05			High Speed Taxi & Lift Off Characteristics			X			ZC-8-511-1
1.5	" "	"	"	2.05			Brake Chute Operation			X			EF-8-602 EF-8-623
1.5	" "	"	"	4.01		EF-8-454	Take Off			X			EF-8-485 & 511 EFT-A-17
1.5	" "	"	"	4.05			Landing			X			ZC-8-511-1
1.5	" "	"	"	8.04			Max. CL Landing					1791	
1.5	" "	"	"	8.05			Take Off					1791	
1.5	Take Off & Landing	1788	ZC-8-119		2		Drag Chute Deployment			X		1787	

* EFT-P-3B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO.*	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
2.1	Stability & Control	1351	ZC-8-140	7.01			Static Long. Stability	X					Phase III
2.1	" "	"	"	7.02			Elevator Contr. Effect.	X					Phase III
2.1	" "	"	"	7.03			Max. CL Land. Contr. Effect	X					Phase III
2.1	" "	"	"	7.04			T.O. Elevator Control						
							Forces Effectiveness	X					Phase III
2.1	" "	"	"	7.05			Max. Load Factors	X					Phase III
2.2	" "	"	"	7.06			Static Direc. Stability	X					Phase III
2.2	" "	"	"	7.07			Rudder ContrForceEffect.	X					Phase III
2.2	" "	"	"	7.08			Adverse Yaw	X					Phase III
2.1	" "	"	"	8.01			High Speed Dive Require.	X					Phase III
2.1	" "	"	"	8.03			Long. Trim Changes	X					Phase III
2.1	" "	"	"	8.04			Maneuver. Flt. Character.	X					Phase III
2.1	" "	"	"	8.02			High Speed Flt. Require.	X					
2.1	Stability & Control	1798	ZC-8-167	4.01			Dynamic Long. Character.	X					
2.1	" "	"	"	4.02			Static Long. Character.	X					
2.2	" "	"	"	4.03			Elevator Control Effect.	X					
2.1	" "	"	"	4.04			Ma. Load Factors					1794	
2.1	" "	"	"	4.05			Max. CL Landing	X					
2.1	" "	"	"	4.06			Take Off	X					
2.1	" "	"	"	4.07			High Speed Flight	X					
2.1	" "	"	"	4.08			Longitudinal Trim Change	X					
2.1	" "	"	"	4.09			Maneuver. Flt. Character.	X					
2.2	" "	"	"	4.10			Dynamic Lateral Directional Stability	X					
2.2	" "	"	"	4.12			Adverse Yaw	X					
2.2	" "	"	"	4.13			Rudder Control Effect.	X					
2.2	" "	"	"	4.14			Aileron Characteristics	X					
2.1	" "	"	"	4.16			Trimming Devices	X					
							Reduced Rudder Steering Forces			X			
2.1	" "	"	"	4.11			Static Direct. Stability	X					

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
2.2	Stability & Control	1813	ZC-8-171	2.01			Static Direct. Stability			X			EFT-A-8-81
2.2	" "	"	"	2.02			Dynamic Lateral Directional Stability			X			EFT-A-8-59
2.2	" "	"	"	2.03		EFT-P-8-217	Roll Coupling			X			
2.1	Stability & Control	1351	ZC-8-140	5.01			Dynamic Long. Stability		X				EFT-A-8-166
2.1	" "	"	"	5.02			Static Long. Stability	X					
2.1	" "	"	"	5.03			Maneuv. Flt. Character.	X					
2.2	" "	"	"	5.04			Dynamic Direct. Stabil.		X				
2.2	" "	"	"	5.05			Static Direct. Stability		X				
2.2	" "	"	"	5.06		EFTP-8-459	Positive Quadrant Roll Coupling Tests			X			
2.1	" "	"	"	6.00			Dynamic Response	X					Phase III
2.1	" "	"	"	6.01			Dynamic Long. Stability	X					Phase III
2.2	" "	"	"	6.02			Dynamic Lateral Stability	X					Phase III
2.1	Stability & Control	1791	ZC-8-120	6.02			Dynamic Long. Stability		X				
2.1	" "	"	"	6.03			Static Long. Stability			X			
2.2	" "	"	"	6.05			Dynamic Direct. Stability		X				
2.2	" "	"	"	6.06			Static Direct. Stability		X				
2.1	" "	"	"	6.07			Roll Coupling Tests	X					
2.2	" "	"	"	7.00		EFTP-8-117	Rudder Effectiveness		X				2 Flights Required
2.1	" "	"	"	7.01			Dynamic Long. Stability					1798	
2.1	" "	"	"	7.02			Static Long. Stability					1798	
2.1	" "	"	"	7.04			Max. C_L Landing					1798	
2.1	" "	"	"	7.05			Take Off			X			

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO.*	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO.	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
2.1	Stability & Control	1791	ZC-8-120	7.06			Max. Load Factors					1798	
2.1	" "	"	"	7.09			Long. Trim Changes					1798	
2.1	" "	"	"	7.10			Maneuv. Flt. Character.					1798	
2.2	" "	"	"	7.11			Dynamic Lat.Direct.Stab.					1798	
2.2	" "	"	"	7.12			Static Direct. Stability					1798	
2.2	" "	"	"	7.13			Adverse Yaw					1798	
2.2	" "	"	"	7.14			Rudder Cont. Effect.					1798	
2.2	" "	"	"	7.15			Aileron Characteristics					1798	
2.1	" "	"	"	7.17			Trimming Devices						
2.1	Stability & Control	7994	ZC-8-101	8			Dynamic Response Tests			X			
2.1	" "	"	"	16			Stability & Control Tests			X			EF-8-84 F.T.Summary Flt 1-6
2.2	Stability & Control	7995	ZC-8-103	11		EF-8-520	A/C Response to High Rate			X			EF-8-558,-618
2.1	" "	"	"	12			Angle Attack & Yaw Calib					1780	
2.1	" "	"	"	18			Dynamic Response					1780	
2.1	" "	"	"	19			Stab.&Cont.Spec.Compliance					1791	1798
2.2	" "	"	"		5		Dyn. Direct. Stability			X			ZC-8-501-10,-15,-16, & 17
2.1	" "	"	"		14A		Long. Sensitivity			X			ZC-8-501-21,-24, EF-8-537
2.2	" "	"	"		24A	EF-8-562	Wing Fence Evaluation			X			ZC-8-512-10,-11,-13,-14,-15, 16, & 18
2.2	" "	"	"		25A		Adverse Yaw & Direct.						
							Characteristic			X			ZC-8-512-4

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
2.2	Stability & Control	1779	ZC-8-108	4			Dyn. Lat. Directional Stab. w/wo Damp			X			ZC-8-502-2,-3,-4,-6,-7,-8
2.2	" "	"	"	5			Dyn. Lat Directional Stab. with Damp						ZC-8-502-2,-3,-4,-6,-7,-8
2.1	" "	"	"	6			DynLong.Stab.w/wo Damp			X			ZC-8-502-7,-8,-9
2.1	" "	"	"	7			Dyn.Long.Stab. with Damp			X			ZC-8-502-7,-8,-9
2.1	" "	"	"	8			Dyn. Response to Speed Brake Opera.				X		
2.1	" "	"	"	9			Dyn. Response to Arm.Bay Door Oper.			X			ZC-8-502-3
2.2	Stability & Control	1780	ZC-8-111	1			Dyn. Lat. Direct. Stab.			X			ZC-8-503,1,2,5,7,8,9,12
2.2	" "	"	"	2			Direct. Dynamic Response			X			ZC-8-503,1,2,5,7,8,9,12
2.1	" "	"	"	3			Dynamic Long. Stability			X			ZC-8-503-2,-3, to 12 inc.
2.1	" "	"	"	3c			Determ.most Aft C.G.Limit			X			EF-8-627
2.1	" "	"	"	4			Long. Dynamic Response			X			ZC-8-504-3,-4 thru 12
2.1	" "	"	"	7			Static Long. Stability					X	
2.2	" "	"	"	9			Static Lat. Direct.Stab.					X	
2.1	" "	"	"				Dyn. Response Character.			X			For HAC
2.1	Stability & Control	1781	ZC-8-506	1c			Arm.Dr.Oper.Effects on					1788	
2.1	" "	"	ZC-8-131	4			A/C Response to Rkt.Oper.		X				
2.2	" "	"	ZC-8-112	9			Dyn.Response to Arm.Oper.			X			ZC-8-504-1,2,3,4,6,7,8,9

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO.*	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
2.1	Stability & Control	1783	ZC-8-114	12			Dynamic Response				X		Not Accomp. for this A/C
2.1	Stability & Control	1787	ZC-8-118	5.02			Manual Mode Pitch DampTest			X			
2.1	" "	"	"	5.01			Gen. Handling Character.			X			ZC-8-511-1
2.1	" "	"	"	5.02			Dyn. Long. Stability			X			ZC-8-512-20
2.1	" "	"	"	5.03			Static Long. Stability			X			EF-8-583, EF-8-552
2.1	" "	"	"	5.04			Maneuv. Flt. Character.			X			ZC-8-512-5 EFT-A-8-52
2.2	" "	"	"	5.05			Dyn. Direct. Stability			X			ZC-8-512-5&20 ZC-8-512-25
2.2	" "	"	"	5.06			Static Direct. Stability			X			ZC-8-512-27, EF-8A-1
2.2	" "	"	"	5.07			Lateral Control			X			ZC-8-512-21,-25
2.1	" "	"	"	5.07		EF-448-A	Phase I Stab. & Control			X			Rev. Procedures for #5.02 thru #5.07
2.2	" "	"	"	8.0		EF-8-520	Response to High Roll Rate Maneuvers			X			EFT-A-8-31
2.2	" "	"	"	8.0		EF-8-572	Ail. Roll & Stab. Tests			X			EFT-A-8-31 & EF-8-572A
2.2	" "	"	"	8.0		EF-8-572A	Ail. Roll & Stab. Tests			X			
2.2	" "	"	"	8.0		EF-8-778	Phase III Roll Coup.Tests			X			EFT-A-8-31 & EF-8-778-1
2.2	" "	"	"	8.0		EF-8-778-1	Phase III Roll Coup.Tests			X			
2.2	" "	"	"	8.0		EFTP-8-44	Roll Coup.w/Sideslip F.B.			X			EFT-A-8-31
2.2	" "	"	"	8.0		EFTP-8-112	Roll Coupling to Determine Max. Rate			X			
2.1	" "	"	"	8.01			Dyn. Long. Stability					1791	
2.1	" "	"	"	8.02			Static Long. Stability					1791	
2.1	" "	"	"	8.06			Max. Load Factors					1791	
2.1	" "	"	"	8.07			High Speed Dive Require.					1791	
2.1	" "	"	"	8.09			Long. Trim Changes					1791	
2.1	" "	"	"	8.10			Maneuv. Flt. Character.					1791	Spec. Compliance
2.2	" "	"	"	8.11			Dyn. Lat. Direct. Stab.					1791	
2.2	" "	"	"	8.12			Static Direct. Stability					1791	

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO.	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
2.2	Stability & Control	1787	ZC-8-118	8.13			Adverse Yaw					1791	
2.2	" "	"	"	8.14			Rudder Control Effect.					1791	
2.2	" "	"	"	8.15			Aileron Characteristics					1791	Spec. Compliance
2.2	" "	"	"	8.17			Trimming Devices					1791	
2.4	Stall & Spin Character.	1787	ZC-8-118	8.16			Stall					1791	
2.4	Stall & Spin Character.	1791	ZC-8-120	7.16			Stall Characteristics					1798	
.4	" " "	"	"	10.00			Spin Test	X					
2.4	Stall & Spin Character.	1798	ZC-8-167	4.15			Stall Characteristics	X					
2.4	Stall & Spin Character.	1351	ZC-8-140	8.06			Stall Characteristics	X					
2.5	Structural Integrity	✓7995	ZC-8-103		4		Inlet Duct Stresses(Spec)				X		EF-8-234

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO.	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
2.5	Structural Integrity	1351	ZC-8-140		1		Struc. Integ. Flt. Dem.	X					
2.5	Aerodynamic Loads	✓7995	ZC-8-103	9B			Yellow Canary Pressures			X			ZC-8-501-18 & 20, EF-8-574
2.5	" "	"	"	13		EF-8-475	Wing Tip Struc. Loads			X			EF-8-522, -472
2.5	" "	"	"	8			Pressure on Long Nose			X			ZC-8-512-2, EF-8-626
2.5	Aerodynamic Loads	1781	ZC-8-112	3		EF-8-594	Missile Bay & Fwd. Fus. Air Loads				X	1788	EF-8-627 In Part
2.5	" "	"	"	4			Captive Missile Loads				X		In Part ZC-8-504-8, -9
2.5	Aerodynamic Loads	1788	ZC-8-119	5			Armament Bay Airloads			X			EF-313-2
2.5	Aerodynamic Loads	1791	ZC-8-120	7.00		EF-8-790	Air Loads & Def. Nose Wheel Door			X			
						EFTP-8-431	Air Loads & Def. Nose Wheel Door	X					

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC
2.5	Aerodynamic Loads	1793	ZC-8-121	4			Armament Bay Air Loads	X					
2.5	Aerodynamic Loads	1794	ZC-8-122				High Drag Config.Maneuv.	X					
	" "	"	"				Symmetrical Maneuvers	X					
	" "	"	"				Unsymmetrical Maneuvers	X					
2.5	Flutter,Buffet&Vibration	7994	ZC-8-101	6			Flutter & Vibration			X	X	7995	EF-8-84 Prelim Flts 1-6
2.5	Flutter,Buffet&Vibration	7995	ZC-8-103	9			Flutter & Vibration						ZC-8-501-2,3,4,5,6,7,9,10, 16,17,20 ZC-8-013
2.5	Flutter,Buffet&Vibration	7995	ZC-8-103	9		EF-8-560	Noise Level Measurements			X			
2.5	" " "	"	"	9		EF-8-591	Cockpit Noise & Vibration			X			ZC-8-501-19 & 20
2.5	" " "	"	"		19		Instr Panel Vibration					1791	
2.5	" " "	"	"		22		Vibration & Accel. Cond.			X			ZC-8-501-22
2.5	" " "	"	"		24A		Buffet Character. & W.F. Eval.			X			ZC-8-501-20,-21,-22,-23
2.5	Flutter,Buffet&Vibration	1787	ZC-8-118	5.05		EFTP-8-35	Eval.of Windshield Fairing			X			EFT-A-8-12, ZC-8-512-33
2.5	" " "	"	"		1		High Speed Vib. Invest.			X			
2.5	" " "	"	"		1	EF-8-521	Airloads & Defl. Nose Wheel Door			X			Data Distributed

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO. ETC.
.5	Flutter, Buffet&Vibration	1787	ZC-8-118		1	EF-8-554	A/C Mod. For H.S. Vibration Program			X			EFT-A-8-12
.5	" " "	"	"		1	EF-8-567	Flight Procedure for "Buzz" Investigation			X			
.5	" " "	"	"		1	EF-8-568	Add. A/C Mod. for H.S. "Buzz" Program			X			
.5	" " "	"	"		1	EF-8-569-1	Ditto			X			
.5	" " "	"	"		1	EF-8-569-2-3	Ditto			X			
.5	" " "	"	"		1	EF-8-569-4	Ditto				X		
.5	" " "	"	"		1	EF-8-701	Vibration Survey			X			
.5	" " "	"	"		2		Duct Rumble Investigation			X		1791	
.5	" " "	"	"	5.04		EF-8-832	Deter. of Buffet Bound.			X			EFT-A-8-36
.5	" " "	"	"	5.04		EFTP-8-28	Transonic Buf.Tuft Survey			X			
.5	" " "	"	"	5.04		EFTP-8-172	Transonic Buffet Survey			X			
.5	Flutter, Buffet&Vibration	1791	ZC-8-120	2.01		EFTP-8-45	Buzz, Elimination of			X			
.5	" " "	"	"	2.01		EFTP-8-61 EFTP-8-94	Buss, Elimination of			X			
.5	" " "	"	"	4.00		EF-8-785	Elimination of Low Speed Vibration				X		EFT-P-8-4
.5	" " "	"	"	4.00		EFTP-8-46	Flt.Flutter w Ext. Tanks			X			Flt. Test Progam Ext. Tanks
.5	" " "	"	"	4.00		EF-8-807	In Flight Vibration		X				
.5	" " "	"	"	4.00		EFTP-8-188	Rev.F-102A Flt.Flut.Prog			X			
.5	" " "	"	"	4.01			Det.&Invest.of Vibr.Modes			X			
.5	Flutter, Buffet&Vibration	1806	ZC-8-168			EFTP-8-428	Mis.Bay Door Flut.Invest.			X			Completed 5/1/56

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
2.5	Flutter, Buffet&Vibration	1813	ZC-8-171	1.01			Flight Flutter Invest.			X			
2.5	" " "	"	"	1.01		EFTP-8-195	Flutter Investigation			X			
2.5	" " "	"	"	1.01		EFTP-8-250	Flutter Investigation			X			
2.5	Flutter, Buffet&Vibration	1351	ZC-8-140	4.03		EFTP-8-113	Investigation of Buffet		X				
2.5	" " "	"	"	4.03		EFTP-8-113A	" " "			X			
2.5	" " "	"	"	4.03		" -113B	" " "			X			
2.5	" " "	"	"	4.03		" -113C	" " "			X			
2.5	" " "	"	"	4.03		" -113D	" " "				X		
2.5	" " "	"	"	4.03		" -113E	" " "				X		
2.5	" " "	"	"	4.03		" -113F	" " "			X			
2.5	" " "	"	"	4.03		" -113G	" " "			X			EFT-A-8-107, 173, 178
2.5	" " "	"	"	4.03		" -113H	" " "						
2.5	" " "	"	"	4.03		" -113I	" " "	X					EFT-A-8-98 EFT-A-8-107
2.5	" " "	"	"	9.00			Cockpit Noise Level	X					
2.5	" " "	"	"	4.03		" -113J	Optimization Vortex Gen.			X			
2.5	" " "	"	"	4.03		" -113K	" " "			X			
2.5	" " "	"	"	4.03		EFT-P-8-328	Performance Tests			X			ZC-8-512-41,43,44,45,48,50, 51,53,63
2.5	Flutter, Buffet&Vibration	1353	ZC-8-	1.01			Qualitative Eval of Buff.			X			
2.5	" " "	"	"	2.01			Cut Down Canopy			X			ZC-8-512-63
2.5	" " "	"	"	2.02			Cut Down Canopy Plus Minor Modifications			X			
						EFT-P-8-426	Landing Gear Extension Time Tests			X			EFT-A-8-141, ZC-8-512-61
2.5	System Loads	1781	ZC-8-117	1		EF-8-541	Arm.Dr.Cyl.Rod End Loads			X			EF-8-510
2.5	" "	"	"	16		EF-8-505	Door Accel. Measurements			X			EF-8-534, ZC-8-512-5

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO.*	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADC. NO.	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
.5	System Loads	1788	ZC-8-119	6		EF-8-531	MLG Door Link Loads			X			EF-8-643
.0	Additional Sys. Testing	1813	ZC-8-171										
.0	" " "	"	"	2.04		EFTP-8-349	Sund. Const. Speed Drive					1380	EFT-8-142
.0	" " "	"	"			EFTP-8-353	Lateral Control System			X			
.0	" " "	"	"			EFTP-8-364	Linear Thrust Fuel Contr.		X				
.0	" " "	"	"			EFTP-8-356	MLG Operating Time &						
							Fairing Loads Link					1791	
.0	" " "	"	"			EFTP-8-361	Elevon Camera Fairing			X			
.0	" " "	"	"			EFTP-8-361	Lateral Control System				X		
	NOTE:												
	Air Force Testing												
	Phase II	~~7995~~								X			USAF Phase II Evaluation
	Phase IV	1783								X			
	Phase II	1787								X			USAF
	Phase IV	1791								X			
3.1	Power Plant	1380	ZC-8-240	1.01			Engine Operation		X				
3.1	" "	"	"	1.02			Engine Equipment Oper.		X				
3.1	" "	"	"	1.03			Static Thrust Calib.		X				
3.1	" "	"	"	1.04			Low Speed Taxi Tests	X					
3.1	" "	"	"	2.01			Functional Test Flts		X				
3.1	" "	"	"	4.00			Anti-SurgeBld.Vlve Optim		X				
3.1	" "	"	"			EFTP-8-576	A/B Shroud Surface Temp.		X				

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO.	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.1	Power Plant	1380	ZC-8-240	5.02			A/B Ignitor Valve	X					
3.1	" "	"	"	6.01			Nozzle Actuating Temp.	X					
3.1	" "	"	"	7.01			Neg. "g" Check	X					
3.1	" "	"	"	7.02			Development Changes	X					
				7.03			ChkoutRev.Oil Breath V.S.	X					
						EFTP-8-613	Landing Gear Loads Test		X				
				8.01			N1 Bleed Ducts	X					
				8.02			Plastic Ducts	X					
				8.03			Relocated Oil Breather						
							Outlet	X					
				8.04			External Fuel Tanks	X					
				9.00			Combust. Starter Changes	X					
				10.01			Eval. of Existing Fire						
							Detector System	X					
				10.02			Eval.of W.Kidde Detector	X					
						EFTP-8-576A	Cabin Press. Surge Test.	X					
				11.01			Spec.Conform.(Dive&Climb)	X					
				11.02			Carbon Monoxide Testing	X					
				11.03			Hamilton Standard System	X					
				12.00			Engine Relight Curve	X					
				13.00			Engine Changes Develop.	X					
					1		Skewed Landing Gear Test.	X					
				14.00			Fuse.Skin Drain Adequacy						

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.1	Power Plant Cooling	7795	ZC-8-103	17			Eng.Oil Temp w Oil Cooler			X			ZC-8-501-8&22, ZC-8-512-4,21
3.1	" " "	"	"	17		EF-8-334 &	Engine & Struct. Cooling			X			ZC-8-501-2 thru 11,13,14,15
						EF-8-403							18,19,20, EF-8-334, EF-8-403
3.1	" " "	"	"	18			Aft Section Cooling			X			ZC-8-501-11
3.1	" " "	"	"		15A		Eng. Outer Wall & Nozzle						
							Act. Temp.			X			ZC-8-501-12,13,14,15,18,20
3.1	Power Plant Cooling	1780	ZC-8-111	21			Oil Cooler Tests					X	Not Accomplished
3.1	" " "	"	"		3		Eng. Outer Wall & Nozzle						
							Act. Temp.			X			ZC-8-506-2,-3, EF-8-344
3.1	Power Plant Cooling	1783	ZC-8-114	11			P.P. & Struct. Cooling -						
							Spec. Compl.				X		
3.1	" " "	"	"		1A		Eng. Outer Wall & Nozzle						
							Act. Temp.			X			ZC-8-506-2,-3, EF-8-334
3.1	Power Plant Cooling	1787	ZC-8-118	2.04			Engine & Struct. Cooling			X			
3.1	" " "	"	"	2.04		EF-8-476	Struct Temp-Aft Fuse.			X			303-2-004
3.1	" " "	"	"	2.04		EF-8-557	Oil Cooler Airflow &						
							Struct. Temp.					1791	
3.1	" " "	"	"	2.04		EF-8-568	NozzleAct.Cyl.Temp w Flap					1791	
3.1	" " "	"	"	2.04		EF-8-593	T-Eng. Diff. Heat Shield						
							Removed					1791	
3.1	" " "	"	"	2.04		EF-8-519	Rev.ofAft Fuse.Temp.Instr.			X			303-2-011
				2.04		ER-8-486	Press. Across Eng.Shroud						
							Sta. 620						

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.1	Power Plant Cooling	1791											
3.1	" " "	"		2.04		EF-8-87	Flap Type A/B Temps.				X		
3.1	" " "	"		2.04		EF-8-800	Tests with J-57 Diff						
							Heat Shield Removed				X		
3.1	" " "	"		2.04		EF-8-825	A/O Cooler Airflow Optim.				X		Per EFT-P-8-187
3.1	" " "	"		7.00		EFT-P-8-2	Deter. of Dive Brake Temp.			X			"5" Flights Required
3.1	Power Plant Cooling	1351	ZC-8-140	2.05			Eng. & Structural Cooling			X			
3.1	Power Plant Cooling	1380	ZC-8-240	2.03			Eng. & Structural Cooling	X					
3.1	" " "	"	"	2.04			Eng. Diffuser Heat						
							Shield Removal	X					
3.1	Power Plant	1995	ZC-8-103	26			Eng.&Assoc.Equip. Ph.III			X			
3.1	Power Plant	1787	ZC-8-118	1.01			Engine Operation			X			ZC-8-511-1
3.1	Power Plant	1791	ZC-8-120	1.01			Engine Operation			X			

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.1	Power Plant	1351	ZC-8-140	1.01			Engine Operation			X			ZC-8-512-37
3.1	Power Plant Performance	✓1995	ZC-8-103	1			Static Thrust Calibration			X			ZC-8-501-1
3.1	" "	"	"	2			Inlet Duct Press.				X		Superseded by 16A
3.1	" "	"	"		10A		Bleed Air Consump. Effect			X			EF-8-247, EF-8-501-15
3.1	" "	"	"		16A		Compressor Stall Test			X			ZC-8-501-19-20-21-22, EF-8-215 & 513
3.1	" "	"	"	21			Engine Oper. Air-Starts					1781	
3.1	Power Plant Performance	1781	ZC-8-131	6			Engine Snuffing		X				
3.1	Power Plant Performance	1787	ZC-8-118	1.03			Static Thrust Calibration			X			EF-8-499
3.1	" "	"	"		3		Eng. Compressor Stall Test	X					
							Ejector Tail Pipe Eval.			X			ZC-8-177
3.1	Power Plant Performance	1797	ZC-8-166	2.04			Engine Response to Firing			X			

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.1	Power Plant Performance	1806	ZC-8-168	2.04			Engine Response to Firing		X				
3.1	Power Plant Performance	1351	ZC-8-140	1.03			Static Thrust Calibration		X				
				2.04			Power Plant Operation			X			ZC-8-512-49, 62 EFT-A-8-110
3.1	P.P. Component Operation	7995	ZC-8-103	4			Engine Control Operation			X			ZC-8-501-9-18 Flt.Oper.Rpt.
3.1	" " "	"	"	8			Engine Equipment Oper.			X			ZC-8-501-2 thru 10, EF-8-258
3.1	" " "	"	"	26		ZC-8-527	Grnd Test Combust Starter			X			ZC-8-512-15 Flt.Oper.Rpt.
3.1	P.P. Component Operation	1780	ZC-8-111	19			Power Control Tests						
3.1	" " "	"	"	20			Engine Oil System Test						
3.1	" " "	"	"		5		Throttle Idle Stop Oper.				X		
3.1	" " "	"	"		7		Engine Exhaust Nozzle Man Override			X			ZC-8-515
3.1	P.P. Component Operation	1787	ZC-8-118	1.02			Engine Equipment Oper.			X			
3.1	" " "	"	"	1.02		EF-8-553	Starter Ground Tests			X			Data Distributed

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.1	P.P. Component Operation	1788	ZC-8-119		4		Eval. Emer. Alternator			X			EFT-A-8-23
3.1	P.P. Component Operation	1791	ZC-8-120	1.02			Engine Equipment Oper.			X			
3.1	" " "	"	"	1.02		EF-8-809	AC Freq. Maint. at "G"				X		Per EFT-P-8-96
3.1	" " "	"	"	1.02		EFTP-8-3	Sundstrand Test at Red.						
							Oil Tank Press.				X		
3.1	" " "	"	"	1.02		EFT-P-8-5	Anti-SurgeBldVlvOprLimits				X		
3.1	" " "	"	"	1.02		EFT-P-8-97	Sund.Drive Recycling Test				X		
3.1	P.P. Component Operation	1798	ZC-8-167		2		P-41 Eng.Fuel Contr.Eval.			X			3 Flights Required
3.1	P.P. Component Operation	1351	ZC-8-140	1.02			Engine Equipment Oper.			X			ZC-8-512-37
3.2	Fuel System Operation	7994	ZC-8-101	13			Fuel Consumption Tests				X		
3.2	" " "	"	"	14			Emer. Fuel System Oper.				X		

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO.	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.2	Fuel System Operation	✓7995	ZC-8-103	23			Emergency Fuel Contr.Sys.				X		
3.2	" " "	"	"		4		DummyFuelTankVentPressure				X		
3.2	Fuel System Operation	1780	ZC-8-111	22			Alt.&Fuel Cell Press.Test					X	
3.2	" " "	"	"	23			Fuel Availability Tests					X	
3.2	Fuel System Operation	1781	ZC-8-112			EF-8-437	Fuel Venting Press.Meas.			X			ZC-8-504-6,7,8, ZC-8-508, ZC-8-512-1
3.2	Fuel System Operation	1787	ZC-8-118	2.00		EF-8-540	Perform.Eval.of Whittaker						303-2-012
3.2	" " "	"	"				Fuel Flow Equaliser			X			
3.2	" " "	"	"	2.03			Emergency Fuel Control			X			
3.2	Fuel System Operation	1351	ZC-8-140	2.03			Emergency Fuel Control	X					
3.2	Fuel System Operation	1380	ZC-8-240	3.01			Flow Joiners	X					
3.2	" " "	"	"	3.02			Twin Boost Pump	X					
3.2	" " "	"	"	3.03			Modified Eng. Fuel Contr.	X					
3.2	" " "	"	"	3.04			Fuel Tank Neg'g'VentValve	X					
							Emer. Fuel Control Sys.	X					

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.3	Flt. Contr. Sys. - Manual	1351	ZC-8-140	2.07			Contr. Sensitivity Check	X					
3.3	" " " "	"	"	7.09			Trimming Device	X					
3.3	Flt. Contr. Sys. Augmented	1780	ZC-8-111	2		EF-8-494	YawRateGyroVibr.Survey					1791	
3.3	" " " "	"	"	2		EF-8-787	TimeConst.&Rate Gain Tests					1787	
3.3	" " " "	"	"	3			Bow Spring Tests			X			EF-8-369
3.3	" " " "	"	"	3		EF-8-477	Cont.Sys.Pitch Oscill.Prog.					X	EF-8-388
3.3	" " " "	"	"	4		EF-8-487	DampPitchRateGyroMountEval.					1798	
3.3	" " " "	"	"	4		EF-8-489	Damper Amplifier Environ.					1787	EF-8-628
3.3	" " " "	"	"	4		EF-8-787	TimeConstant&RateGainTest					1787	
3.3	" " " "	"	"	4		EF-8-782	Non Linear Pitch Rate						
							Feedback Pots.			X		1787	EF-8-636
3.3	" " " "	"	"	4		EF-8-786	RelocatedSurf.FeedbackPots			X			
3.3	" " " "	"	"	6		EF-8-335	Turn Co-ordinator Tests			X			EF-8-640, EFT-A-16
3.3	" " " "	"	"	6		EF-8-575	Line Lag, in A/S Compen-						
							sator of Turn Co-Ord.			X			EF-8-630
3.3	" " " "	"	"	6		EF-8-781	Turn Co-ordinator Tests			X			Superceded by EF-8-781A
3.3	" " " "	"	"	6		EF-8-781A	Turn Co-ordinator Tests			X			
3.3	" " " "	"	"	6		EF-8-793	Pnue.Limiter&Turn Co-Ord.			X			
3.3	Flt. Contr. Sys. Augmented	1787	ZC-8-118	6.05		EFTP-8-206	Lateral Accel. Protection			X			EFT-A-8-73
3.3	" " " "	"	"			EFTP-8-185	HIG 4 Gyro & M.H. with						
							Doelcam Gyro	X					
3.3	" " " "	"	"	2.06		EF-8-794	Turn Co-ordinator Tests			X			ZC-8-512-8

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO. ETC.
3.3	Flt. Contr. Sys. Augmented	1791	ZC-8-120	2.00		EFTP-8-134A	Prod. Config. Sideslip						
							Feedback Device			X			6 Flights Required
3.3	" " " "	"	"	2.02		EFTP-8-55A	Yaw Rate Gyro Vib. Survey			X	X		3 Flights Required
3.3	Flt. Contr. Sys. Augmented	1798	ZC-8-169	2.02			Turn Co-ordinator			X			2 Flights Required
3.3	" " " "	"	"	2.03			Manual Mode Yaw Damper			X			2 Flights Required
3.3	" " " "	"	"	2.04			Manual Mode Pitch Damper			X			1 Flight Required
3.3	" " " "	"	"	3.02			Turn Co-ord. Development	X					
3.3	" " " "	"	"	3.03			Pitch Damper Development	X					
3.3	" " " "	"	"	3.04			Yaw Damper Development	X					
3.3	" " " "	"	"			EFTP-8-629	Hi-Speed Lat-Dir Oscil.			X			
3.3	" " " "	"	"			EFTP-8-653	Hi-Speed Osc. Correct Meas.		X				
3.3	" " " "	"	"	3.05			Trim Servo Development	X					
3.3	" " " "	"	"			EFTP-8-166	Fin Pitot Static Invest.		X				
3.3	Flt. Contr. Sys. Automatic	7995	ZC-8-103	20			Pilot Assist Sys. Dev.					1780	
3.3	Flt. Contr. Sys. Automatic	1780	ZC-8-111	1		EF-8-515	Lin. Feedback Pot Environ						
							Lateral			X		1798	
3.3	" " " "	"	"	3		EF-8-515	Lin. Feedback Pot Environ						
							Longitudinal			X		1798	
3.3	" " " "	"	"	4		EF-8-725	Elec. Pitch "G" Limiter					1787	EFT-P-8-6
3.3	" " " "	"	"	5			Full Auth. Elevon Cont.						
							& Prelim. Assist			X			ZC-8-503-6-7-8-9-10-11-12
3.3	" " " "	"	"	5			Elevon Surf. Pot. Temp.			X			EF-8-610
3.3	" " " "	"	"	5		EF-8-342	Pilot Assist Sys. Flt. Test						EF-8-640, EF-8-625
3.3	" " " "	"	"	5		EF-8-705	Gnd Test, Plt Assist Mode			X			

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADC. NO	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.3	Flt. Contr. Sys. Automatic	1780	ZC-8-111	8		EF-8-803	Pilot Assist Beep Trim						
							Authority Eval.					1787	
3.3	" " " "	"	"		4		F/T of Pitch "G" Limiter					1787	EF-8-601
3.3	" " " "	"	"		4	EF-8-555	F/T of Pitch "G" Limiter Test					1787	EF-8-478
3.3	Flt. Contr. Sys. Automatic	1783	ZC-8-114	13			Pilot Assist System						Test Proc. Rev. by Memos.
3.3	Flt. Contr. Sys. Automatic	1787	ZC-8-118	2.06			Automatic Flight System			X			EF-8-633 EF-8-602
3.3	" " " "	"	"	6.05			Pilot Assist System			X			Test Proc. Rev. by Memos.
3.3	" " " "	"	"	6.05		EF-8-722	Pilot Assist System			X			EFT-A-8-38
3.3	" " " "	"	"	6.05		EF-8-722-1	Pilot Assist System Test			X			EFT-A-8-136
3.3	" " " "	"	"	6.05		EFTP-8-37	Elec. Pitch "G" Limiter Test			X			EFT-A-8-78
3.3	" " " "	"	"	6.05		EFTP-8-48	Damper Ampl Drift Test			X			
3.3	" " " "	"	"	6.05		EFTP-8-52	Pilot Assist Beep Trim Dev.			X			EFT-A-8-136
3.3	" " " "	"	"	6.05		EFTP-8-57	Westinghouse "G" Limiter						
							Evaluation			X			
3.3	" " " "	"	"	6.05		EFTP-8-66	Plt Assist Up Elev. Func.			X			EFT-A-8-136
3.3	" " " "	"	"	6.05		EFTP-8-79	Low Force Lockout					1798	Ref. EFT-P-8-155
3.3	" " " "	"	"	6.05		EFTP-8-106	Rev. to P.A. Sys. Config.			X			
3.3	" " " "	"	"	6.05		EFTP-8-128	Pitch&Roll Att. GainOptim					1798	
3.3	" " " "	"	"	6.05		EFTP-8-144	Statnum Angular Accel.			X			
3.3	" " " "	"	"	6.05		EFTP-8-176	Lagged Attitude-Plus Rate				X		EFT-A-8-136
3.3	" " " "	"	"	6.05		EFTP-8-241	Pitch "G" Limiter Tie						
							into Plt Assist Sys.			X			DP-8-15
3.3	" " " "	"	"	6.05		EFTP-8-155	Low Force Lockout Flt. Test					1798	

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.3	Flt. Contr. Sys.Automatic	1791	ZC-8-120	2.06			Pilot Assist System			X			3 Flights Required
3.3	" " " "	"	"	2.06		EF-8-805	Spec.Plt Assist Checkout			X			
3.3	Flt. Contr. Sys.Automatic	1798	ZC-8-167	1.03			Elevon System Operation			X			
3.3	" " " "	"	"	2.01			Pilot Assist System			X			3 Flights Required
3.3	" " " "	"	"	2.01		EFTP-8-199	P.A. Up Elev. Checkout			X			2 Flights Required
3.3	" " " "	"	"	2.01		EFTP-8-200	P.A. Anti-Engage Checkout	X					1 Flight Required
3.3	" " " "	"	"	3.01			Pilot Assist Development	X					
3.3	" " " "	"	"	3.01		EFTP-8-110	Increased Surf.F.B. Gain			X			
3.3	Flt. Contr. Sys.-Manual	1780	ZC-8-111			EF-8-539	Decrease in Control Cable						
							Tension with Alt.					1795	
3.3	" " " "	"	"	8			Longitudinal Contr.Forces			X			ZC-8-504-3-4-5-6-7-8
3.3	" " " "	"	"	8		EF-8-452	Elev & Aileron Breakout				X		
3.3	" " " "	"	"		8	EFT-A-8-24	Roll Rate Limiter Dev.			X			EFT-A-8-24
3.3	Flt. Contr. Sys.-Manual	1787	ZC-8-118	2.06		EF-8-716-1	Trim Servo Evaluation			X			ZC-8-512-28
3.3	Flt. Contr. Sys.-Manual	"	"	2.06		EF-8-716	Trim Servo Evaluation			X			ZC-8-512-28
3.3	" " " "	"	"	2.07			Control Sensitivity			X			303-2-003
3.3	" " " "	"	"	2.07		EF-8-535	Elev Contr. Force Link					1791	
3.3	" " " "	"	"	2.07		EF-8-586	Diff. Linkage Eval. Tests					1791	

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADC. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC
3.3	Flt. Contr. Sys.-Manual	1791	ZC-8-120	2.02		EFT-P-8-8	Man.ModePitchDamperChkout				X		
3.3	" " " "	"	"	2.02		EFT-P-8-27	Interim Trim Servo				X	1798	2 Flights Required
3.3	" " " "	"	"	2.04		EFT-8-796	Shelley Unit Compt. Static Press.			X			
3.3	" " " "	"	"	2.07			Control Sensitivity			X			2 Flights Required
3.3	" " " "	"	"	2.07		EF-8-776	Hyd. Fluctuation Re. Contr. Movement			X			
3.3	" " " "	"	"			EF-8-819	Diff. Linkage Eval.Tests			X			
3.3	Flight Controls	7995	ZC-8-103		20	EF-8-428	Relocate "Q" Pot Static Source			X			EF-8-642
3.3	Flight Controls	1779	ZC-8-108		1		Control System Ground Test			X			ZC-8-502-4
3.3	Flight Controls	1787	ZC-8-118	2.02			Flt. Contr. Sys. Oper.			X			EF-8-683
3.3	Flight Controls	1798	ZC-8-167	1.01			Hysterisis&BrakeoutForces			X			
3.3	" "	"	"	3.06			Components Development		X				
3.3	" "	"	"	3.06		EFTP-8-166	Fin Pitot Static Invest.			X			
3.3	" "	"	"			EFTP-8-311	Rate Gyro Environ. Test			X			
3.3	" "	"	"			EFTP-8-424	Western Pitch "G" Limiter	X					
3.3	" "	"	"			EFTP-8-442	Shelley Unit Tests		X				
3.3	" "	"	"			EFTP-8-495A	Elect. Pitch "G" Limit.		X				
3.3	" "	"	"			EFTP-8-486	H.E.P. Valve Tests		X				

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.3	Flight Controls	1798	ZC-8-167			EFTP-8-582	Aileron Ratio Changer	X					
3.3	" "	"	"			EFTP-8-350	AC Rate Gyro Environ Test		X				
3.3	" "	"	"			EFTP-8-599	Revised Shelley Unit Test		X				
3.3	" "	"	"			EFTP-8-644	Elec. Pitch Limiter-Func.						

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADD. NO	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO. ETC.
3.3	Flight Controls	1351	ZC-8-140	2.02		EFTP-8-402	Turn Co-Ordinator & Damper System Oper.			X			
3.3	" "	"	"	2.02			Flight Control Sys. Oper.			X			
3.3	" "	"	"	8.05			Aileron Characteristics	X					
3.3	Flt. Contr. Sys.-Manual	✓7995	ZC-8-103	25			Flight Control Sys. Oper.			X			ZC-8-501-8-9-12-16-17-19-21-22-23
3.3	" " " "	"	"										ZC-8-512-5-6-&-26, EF-8-32 -420-287-608-617-519
3.3	" " " "	"	"	25			Elevon System Force Link			X			ZC-8-501-2
3.3	" " " "	"	"		23		Cable Tension Valves			X			ZC-8-512-5
3.4	Hydraulic Systems Oper.	✓7994	ZC-8-101	11			Emerg. Ram Air Hyd. Power				X		
3.4	Hydraulic Systems Oper.	✓7995	ZC-8-103	2			Brake Operation Eval.			X			ZC-8-501-1-5-6-15-18
3.4	" " "	"	"	24			Ram Air Turbine Tests					1781	Add. 12B Reinstates this Test to 7995
3.4	" " "	"	"		1		Sim Landings with R.A.T.				X		Superceded by Add. 12B
3.4	" " "	"	"	6			Speed Brake Operation			X			ZC-8-501-2
3.4	Hydraulic Systems Oper.	1781	ZC-8-112		12 &	13	Preliminary R.A.T. Tests			X			ZC-8-504-1-2-3
3.4	" " "	"	"			ZC-8-103	" " "			X			

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO.*	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADC. NO	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
.4	Hydraulic Systems Oper.	1783	ZC-8-114		5		Perform. of R.A.T. Emerg.			X			ZC-8-506-1-2-8
.4	" " "	"	"		5A		Perform.of Mod R.A.T.Sys.					7995	ZC-8-506-8
.4	Hydraulic Systems Oper.	1787	ZC-8-118	2.01			Shakedown Flights			X			ZC-8-511-1
.4	" " "	"	"	2.01		EF-8-449	Hydraulic System Temps.			X			Data Distributed
.4	" " "	"	"	6.02			Braking System					8-10	Series
.4	" " "	"	"	6.04		EF-8-814	Ram Air Turbine Tests			X			Data Distributed, no Memo
.4	" " "	"	"			EFT-P-8-72	Ram Air Turbine Tests			X			EFT-A-8-73
.4	Hydraulic Systems Oper.	1798	ZC-8-167		4		R.A.T. Checkout			X			2 Flights Required
.4	Hydraulic Systems Oper.	1797	ZC-8-166				R.A.T. Cylinder Rod Loads					1806	EFT-P-8-548
.5	Low Press.Pneu.Sys.Oper.	✓7994	ZC-8-101	10			Fuel Tank Press.Sys.Tests				X		
.5	Low Press.Pneu.Sys.Oper.	✓7995	ZC-8-103	7			Fuel Tank Pressurisation			X			ZC-8-501-4 (3 May 56)
.5	" " " " "	"	"		8	EF-8-592	Redome Glycol Dispersal			X			ZC-8-512-15-16-29:EF-8-637

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. #	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
.5	Low Press.Pneu.Sys.Oper.	1780	ZC-8-111	13			A.T.M. Power Capabilities			X			ZC-8-517
.5	" " " " "	"	"	13		EF-8-329	A.T.M. Perform & Instru.			X			
.5	" " " " "	"	"	13		EF-8-427	A.T.M. Cooling & Load						
							Capabilities			X			EF-8-604
.5	" " " " "	"	"	15			Cooling Capabil.of ATM Sys			X			ZC-8-517
.5	" " " " "	"	"	16			A.T.M. Sys. Stability			X			ZC-8-517
.5	" " " " "	"	"	17			Ground Operation A.T.M.			X			ZC-8-517
.5	" " " " "	"	"	18			Water Tightness Tests						
.5	" " " " "	"	"		2		Boundary Layer Recovery						
							Inlet D			X			
.5	Low Press.Pneu.Sys.Oper.	1783	ZC-8-114	4			Ground Test of "W"-T &						
							Engine Anti-ice			X			ZC-8-506-6
.5	" " " " "	"	"	5			W & T & Eng. Inlet & W/S						
							(NESA) Anti-ice			X			ZC-8-506-6-7 Climb&Descent
.5	" " " " "	"	"	6			Ditto & Level Flight			X			ZC-8-506-6-7
.5	" " " " "	"	"	7			Radome Anti-icing			X			ZC-8-506-8-9
.5	" " " " "	"	"	8			Windshield & Canopy Defog			X			ZC-8-506-6-7
.5	" " " " "	"	"	9			Heat&Vent Sys.Spec.Chkout			X			
.5	" " " " "	"	"	10			Cabin Press. Check-Spec.			X			ZC-8-506-4
.5	" " " " "	"	"		3A		Cabin Air Temp.			X			ZC-8-506-4-5-6-7
.5	" " " " "	"	"		5		Windshield & Canopy Defog			X			ZC-8-506-7
.5	Low Press.Pneu.Sys.Oper.	1787	ZC-8-118	6.01			Boundary Layer Bleed Air					8-10	
.5	" " " " "	"	"	6.03			Low Press. Pneumatic Sys.					8-10	

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.5	Low Press.Pneu.Sys.Oper.	1791	ZC-8-120	9.00			Cockpit Carbon Monoxide				X		2 Flights Required
3.5	" " " " "	"	"	11.00			Cockpit & Inclosure						
							Watertightness	X					1 Flight Required
3.5	" " " " "	"	"	12.00			Cabin Pressurization				X		
3.5	Low Press.Pneu.Sys.Oper.	1799	ZC-8-143	1a		EFT-P-8-11	Fire Control Sys.Cooling			X			
3.5	Low Press.Pneu.Sys.Oper.	1389								X			
3.5	Low Press.Pneu.Sys.Oper.	1351	ZC-8-140	2.01		EFTP-8-114	Ram Air Press. & Temp.						
							Regulator 148			X			
3.5	" " " " "	"	"	2.01		EFTP-8-271	Irreversible Power Contr.			X			
3.5	" " " " "	"	"	2.01		EFTP-8-271	Emerg.Land.Gear Ext.Time			X			EFT-A-8-141 ZC-8-512-61
3.9	Landing Gear	1780	ZC-8-111	24			Emerg. Opr.of Land. Gear					X	
3.9	" "	"	"	25			Normal Opr.of Land. Gear					X	

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO.	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.9	Landing Gear	1791	ZC-8-120	1.04		EFT-P-8-43	Main Land.Gear Flexibility			X			
3.9	" "	"	"	2.00		EFTP-8-140A	MLG Sidebrace Loads			X			30 Flights Required
3.9	" "	"	"			EFTP-8-163	MLG Fore&Aft Resonant Freq		X				
3.9	" "	"	"			EFTP-8-228	Landing Gear Oper. Time				X		3 Flights Required
3.9	" "	"	"			EFTP-8-232	Natural Vibration Freq.			X			
3.9	Landing Gear	1797	ZC-8-166	11			Landing Gear Oper. Time			X			
							MLG WingFairingLinkLoads			X			
						EFTP-8-548	Deter.In-Flt.Loads onMLG	X					
3.10	Systems Operation	1791	ZC-8-120	2.05			Drag Chute Operation		X				
3.10	Systems Operation	1351	ZC-8-140	2.01			Functional Flight Test		X				
3.10	" "	"	"	2.06			Brake Parachute Oper.		X				
3.10	Systems Operation	1787		2.05			Brake Parachute Oper.			X			EF-8-602, EF-8-623

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO.	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.10	Drag Determination,etc.	7994	ZC-8-101	12			Drogue Chute Deploy.Test				X		
3.10	" " "	"	"	15			Speed Brake Tests				X		EF-8-84
3.10	Drag Determination,etc.	7995	ZC-8-103	16		EF-8-597	Wing Tip-Drag Polars&Eval			X			EF-8-274,274A, EF-8-353
3.10	" " "	"	"	22			Drogue Chute Tests			X			ZC-8-501-10-18-20,EF-8-276
3.10	" " "	"	"		11		Drag Effects Near M 0.9			X			ZC-8-501-21
3.10	" " "	"	"		21		Drag Characteristics			X			ZC-8-501-16-18,19,20,& 21
3.10	Drag Determination,etc.	1787	ZC-8-118	4.03			Level Flight			X			EFT-A-8-17
3.10	Drag Determination,etc	1791	ZC-8-120	5.03			Level Flight		X				4 Flights Required
3.10	" " "	"	"	2.05		EFTP-8-54A	DragChuteSys.Deploy.Time			X			
3.10	" " "	"	"	2.05		EFTP-8-54B	" " " " "			X			
3.10	Drag Determination,etc.	1793	ZC-8-121	5.10		EFTP-8-104	Drag Chute Deploy. Time	X					
3.10	Drag Determination,etc.	1798	ZC-8-167		5		Drag Chute Deploy. Time			X			

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO.*	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
3.6	Electrical Systems Oper.	1780	ZC-8-111	26			Electrical System Tests					X	
4.1	Radio & Radar Sys. Oper.	1782	ZC-8-113	1			Antenna-AN/ARN-18, Glide						
							Pathe			X			ZC-8-505-3
4.1	" " " "	"	"	2			Antenna-AN/ARN-14, Omni			X			ZC-8-505-1-2
4.1	" " " "	"	"	3			Antenna-AN/ARN-25, VHF/DF			X			ZC-8-505-3
4.1	" " " "	"	"	4			Antenna-AN/ARC-34, VHF COMM			X			ZC-8-505-2
4.1	" " " "	"	"	5			Antenna-AN/ARN-12, Marker						
							Beacon			X			ZC-8-505-2
4.1	" " " "	"	"		1		DataLinkAnt.-Isol. Study					1783	
4.1	" " " "	"	"		2		Glide Path Ant.-Location			X			ZC-8-505-3
4.1	" " " "	"	ZC-8-114		4		Antenna Flight Test			X			ZC-8-506-2-3-5
4.1	Radio & Radar Sys. Oper.	1797	ZC-8-166	3.00			XMG-14 System	X					
4.1	" " " "	"	"	3.01			XMG-14 Checkout	X					20 Flights Required
4.1	Radio & Radar Sys. Oper.	1798	ZC-8-124	1			Antenna, Omni, AN/ARN-14			X			Parts A,B,C Complete
4.1	" " " "	"	"	1			Ant. Glide Path, AN/ARN-18			X			Revised per Add. 2
4.1	" " " "	"	"	2			Ant. F/T AN/ARC-34 VHF			X			Parts B,C,D,E Complete
4.1	" " " "	"	"	3			Ant. Flt. Test Data Link			X			Parts A,B,C,D,E,H Complete
4.1	" " " "	"	"	4			Ant. Flt. Test APX-27, IFF			X			Parts A,B Complete
4.1	" " " "	"	"	5			Ant. Flt. Test APX-6, IFF			X			Parts A,B Complete

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO.	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
4.1	Radio & Radar Sys. Oper.	1798	ZC-8-124	6			Ant. Flt. Test ARN-12						
							Marker Beacon			X			
4.1	" " " "	"	"		1		Ant. Study AN/APX-6			X			
4.1	" " " "	"	"		2		Ant., AN/ARN-14 - Omni			X			
4.1	" " " "	"	"		2		Ant., AN/ARN-18 Glide Path			X			
4.1	Radio & Radar Sys. Oper.	1806	ZC-8-168	3.01			XMG-14 Checkout	X					20 Flights Required
4.1	" " " "	"	"	3.02			XMG-14 Evaluation	X					10 Flights Required
4.2	Armament	~~1995~~	ZC-8-103	15			Missile Bay Door Oper.					1780	
4.2	Armament	1781	ZC-8-112	1			Arm. Bay Door Operation			X			ZC-8-504-1-2-3-4-6-7
4.2	"	"	"	2			Arm. Launcher Ext. Arm Oper.			X			ZC-8-504-2-4-6-7-8-9
4.2	"	"	"	2			Arm. Launcher Operation			X			EF-8-397, EF-8-707
4.3	"	"	"	5			Missile Separation			X			ZC-8-504-4-6-8-9
4.3	"	"	"	5	2		Missile Sep. & Traj. at						
							V_{max} & H_p Max.			X			ZC-8-512-1-19
4.3	"	"	"	5	2A		Ditto			X			ZC-8-512-1-19
4.3	"	"	"	6			Missile Trajectory			X			ZC-8-504-4-6-9
4.3	"	"	"	7			Rocket Sep. & Dispersion			X			ZC-8-504-7
4.2	"	"	"	7	3		Two Inch Rocket Package				X		ZC-8-512-25-23
4.2	"	"	"	10			Rkt&Miss Exhaust Effects			X			ZC-8-504-6-7-9-4

* FFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS - SUBJECT BREAKDOWN

GROUP NO.*	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO.	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO. ETC.
4.2	Armament	1781	ZC-8-131	1			Tethered Test Firing			X			EFT-A-144&123, ZC-8-512
4.2	"	"	"	2			Jettison Inert Rocket			X			ZC-8-512-48 2nd Jettison Cancelled by EFT-P-8-249
4.3	"	"	"	3			Separation & Trajectory		X				EFT-A-8-137,144&159 ZC-8- 512-52-55-64
4.2	"	"	"	5			Blast Effect of Rocket		X				
4.2	"	"	"	7			Exhaust Duct Effects		X				
4.2	"	"	"	1		EFTP-8-129	Ramp Jettison			X			EF-313-68
4.2	Armament	1788	ZC-8-119	1			Hardware Check			X			EF-8-707, EF-8-638
4.2	"	"	"	1		EF-8-502	Arm. Hdwre. Ground Test			X			ZC-8-512-8
4.2	"	"	"	1		EF-8-511	Rkt Dr. Aeroelastic Stab.			X			EF-8-585,504,512-11
4.2	"	"	"	1		EF-8-565	Inst.Req'dToLiftArmRestrict			X			EF-8-707
4.3	"	"	"	2			Missile Sep. & Trajectory			X			Except Neg "G" Firings
4.2	"	"	"	3			Rocket Separation			X			ZC-8-512-23
4.2	"	"	"	4			Rkt Traject. & Dispersion			X			ZC-8-512-25
4.2	"	"	"	4		EF-8-791	HAC 2" Rocket Program				X		ZC-8-512-23,40,45 EFT-A-27
4.2	"	"	"	7			Exhaust Duct Effects			X			ZC-8-512-17
4.2	"	"	"	4		EF-8-802	Blast Pan&Flipper Dr.Temp				X		
4.2	"	"	"		3		HAC Captive Missile Test				X		Per Add. 3A
4.2	"	"	"		3A		HAC Captive Missile Test			X			
4.2	"	"	"	5			Missile Traject. Program		X				ZC-8-512-54
4.2	Armament	1806	ZC-8-168	1.00			Inert Rocket Firings						
4.2	"	"	"	1.01			Rocket Ejection Tests		X				
4.2	"	"	"	2.01			Separation & Trajectory	X					

* EFT-P-8B-44

YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
4.2	Armament	1806	ZC-8-168	2.02			Blast Effects	X					
4.2	"	"	"	2.03			A/C Response to Firing	X					
4.2	"	"	"	2.05			Exhaust Duct Effects	X					
				3.01			XMG-14 Checkout	X					
				3.02			XMG-14 Evaluation	X					
							MissileBayDrFlut Invest.			X			
				2.04			Eng. Response to Firing	X					
4.2	Armament	1793	ZC-8-242	1.01			Struc. Qual. Arm. &						
							Displmnt Hardware	X					
4.2	"	"	"	1.02			24 Rkt Door Config.	X					
4.2	"	"	"	1.03			High Press. Pneu. System	X					
4.2	"	"	"	2.01			Missile Sep. Verification	X					
4.2	"	"	"	2.02			Ballistic Data&LaunchVer	X					
4.2	"	"	"	2.03			Missile&Rkt Blast Effect	X					
4.2	"	"	"	2.04			Arm. Bay Air Loads	X					
4.2	"	"	"	2.05			Ext. Hdwre Temp Study	X					
4.2	"	"	"	3.01			In Flight Wind Trans. vs						
							True A.O.A.	X					
4.2	Armament	1797	ZC-8-166	1.00			Armament Hardware			X			
4.2	"	"	"	1.01			Armament Gear Check			X			7 Flights Required
4.2	"	"	"	1.02			Rkt Fly Around Check			X			EFT-P-8-491
4.2	"	"	"	1.03			Rocket Jettison Check			X			2 Flights Required
4.2	"	"	"	2.00			Rocket Firing						
4.2	"	"	"	2.01			Separation & Trajectory			X			26 Flights Required

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG	TEST ITEM NO	ADC. NO	MEMO NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER TO A/C	DATA ANALYSIS REPORTS. MEMO. ETC.
4.2	Armament	1797	ZC-8-166	2.02			Blast Effects			X			
4.2	"	"	"	2.05			Exhaust Duct Effects			X			
4.2	"	"	"	4.00			Rocket Firing						
4.2	"	"	"	4.01			Firing at Target		X	X			10 Flights Required
4.2	"	"	"				MissileBayDr.Flut.Invest.			X			
4.2	Armament	1799	ZC-8-143	1			Prep. of Equip.& Chkout						ZC-8-128, EF-8-817, EFTP-8-56
4.2	"	"	"	1b		EFTP-8-150	MLG Side Brace Load Test			X			EFT-A-8-69
4.2	"	"	"	1c		EFTP-8-153	Launch.RestrictorSizeChk			X			
4.2	"	"	"	1e		ZC-8-137	Calib. & Gnd. Checkout of Fire Control			X			
4.2	"	"	"	1g			Miss.RailAlign.Test&Corr.			X			Mis.
4.2	"	"	"	1h			Shakedown Flt. F.C.S. & Tele. Missile			X			
4.2	"	"	"	1i			Simulated Attack			X			
4.2	"	"	"	1k			Miss.Eval.Co.Alt.Attacks			X			
4.2	"	"	"	1l			Inst. & Checkout of F.C.S. Snap-up			X			4138-649
4.2	"	"	"	1m		EFTP-8-263	Snap-up Perform. Eval.			X			ZC-8-512-57
4.2	"	"	"	IIa			Gnd Sys Chkout with Miss		X				EFT-A-8-148
4.2	"	"	"	IIb			Co-Altitude Sys. Verif.			X			Single Missile Firings
4.2	"	"	"	IIc			Co-Altitude Salvo Firing	X					Instrumented Missiles
4.2	"	"	"	IId			Co-Altitude Salvo Firing			X			Tactical Missiles
4.2	"	"	"	IIg			Snap-up Sys. Verif.	X					
4.2	"	"	"	IIh			Snap-up Salvo Firing	X					
4.2	"	"	"	IIf			Snap-up Performance Flt.			X			
							Snap-up Sys.Verification & Miss-Evaluation		X				
							Snap-up Salvo Firing	X					

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YF&F-102A FLIGHT TEST PROGRAM STATUS ~ SUBJECT BREAKDOWN

GROUP NO. *	TITLE	A/C S/N	FLIGHT PROG.	TEST ITEM NO	ADD. NO	MEMO. NO	TEST	FUTURE	CURRENT	COMPLETE	CANCEL	TRANSFER A/C	DATA ANALYSIS REPORTS, MEMO, ETC.
4.2	Armament	1799	ZC-8-143		1		Production Missile Range						
							Gate Servo Evaluation		X				
4.2	"	"	"		2		W.S.D. w/High Speed Targ.	X					
4.2	"	"	"		3		Missile & Missile Bay						
							Temperature Eval.	X					
4.2	Armament	1354	ZC-8-141	1.01			StrikeCameraBoresighting	X					
4.2	"	"	"	1.02			R.W.T. Boresighting Per						
							SE-0972	X					
4.2	"	"	"	2.01			Arm. Hardware Checkout	X					
4.2	"	"	"	3.01			Separation & Traj. Rkts	X					
							Missiles	X					
4.2	"	"	"	3.02			A/C Response to Arm.Opn.	X					
4.2	"	"	"	3.03			Engine Response to Rkt						
							& Missile Exhaust	X					
4.2	"	"	"	4.01			Inlet Duct Anti-icing						
							System-Spec Compliance	X					Ref. A.F. Spec. 40395-C

1814

F/TF-102A

PRODUCTION FLIGHT TEST PROCEDURES

SECTIONS II & III

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322-1A

JANUARY 1957

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO CALIF.

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UNCLASSIFIED

F/TF-102A

PRODUCTION FLIGHT TEST PROCEDURES

SECTIONS II & III

Classification Changed to:
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Authorized by: SAAMA Date 12-12-66
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R. G. Avritt 130-1 4-4-67

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UNCLASSIFIED

[redacted]

CONVAIR-PALMDALE

17 DECEMBER 1956

PRODUCTION FLIGHT DEPARTMENT

1. THE BASIC F-102A PRODUCTION FLIGHT TEST PROCEDURE DATED 1 NOVEMBER 1955 WAS APPROVED BY AMC LETTER MCPHFC (RDZSFG/19A) WES/eh DATED 28 MAY 1956.

2. THE FOLLOWING PROCEDURE MODERNIZES THE BASIC PROCEDURE IN ELABORATION OF TOLERANCES AND TECHNIQUES GAINED THROUGH ONE YEAR OF EXPERIENCE. IT DOES NOT IN ANY MANNER CHANGE THE INTENT OR OBLIGATION INCURRED BY PARAGRAPH 1 ABOVE.

3. THE PURPOSE OF THIS PROCEDURE IS TO PROVIDE STANDARD GUIDANCE FOR PRODUCTION FLIGHT TESTS IN DETERMINING SPECIFICATION CONFORMANCE OF THE AIRCRAFT AND ITS SYSTEMS. IT DOES NOT SUPPLANT ANY PUBLISHED TECHNICAL ORDERS PERTAINING TO THE F/TF-102A.

R. W. WARDEN, JR.
CHIEF PILOT

FOREWORD

The F/TF-102A Production Flight Test Procedures are divided into three sections:

Section I - Preflight Inspection consisting of all checks, inspections, and operations performed by the Flight Test Engineer prior to the release of the interceptor for flight from Convair San Diego.

Section II - Functional Flight Test including all functional flight checks performed on taxi check, initial flight out of Convair San Diego, and subsequent flights.

Section III - Radar Flight Test scheduled upon completion of the functional flight test in conjunction with a target aircraft. Radar Flight Test includes complete check of the fire control and armament systems. Mechanical discrepancies may be carried forward from Section II but must be checked and cleared on the radar flights.

Generally an aircraft shall have performed satisfactorily during Section I of the Production Flight Test Procedures before being released for Sections II and III. Minor discrepancies may be waived at the discretion of the pilot providing they are cleared before completion of the final section - Radar Flight Test.

These procedures cover Sections II and III only. Section I is published separately.

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SECTION II

FUNCTIONAL FLIGHT TEST

I

LOADING

1 - Release: sign and note aircraft servicing, special limitations, and shortages.

2 - Weight and Balance: check center of gravity (cg) limits 26.5% to 30.5% mean aerodynamic chord (mac).

3 - Engineering Limitations: read and note limitations for this particular aircraft.

Note: in addition to these items the preflight inspection form and engine-run sheet will be reviewed prior to acceptance of a "green" aircraft for ferry flight to Palmdale. The ferry pilot will retain a copy of all five (5) aforementioned forms and turn them in to the Palmdale Flight Dispatch upon arrival.

II

EXTERNAL CHECK

1 - left-hand intake and boundary layer ducts clear

2 - left-hand duct extension secure

3 - left-hand electronic bay door secure

4 - rocket jump angle vane for security and torque paint

5 - left-hand static ports clear

6 - radome condition and security

7 - pitot mast and head secure, pitot cover removed, and static ports clear

8 - right-hand static ports clear

9 - right-hand electronic bay door secure

10 - nose wheel door and well area - no leaks, door seal, taxi light, circuit breakers, battery, dust curtains, microswitch, oleo inflation (should measure 5 inches between scissors axle bolt), tire inflation, tire condition, slippage marks, steering damper, and gear down lock pin removed

11 - right-hand intake and boundary layer ducts clear

12 - right-hand duct extension secure

13 - lower electronic door secure

14 - right-hand missile doors, hinges, and contour

15 - right-hand wing leading edge and wing fences

16 - ram air turbine compartment - primary and secondary fluid level - reservoir caps, primary and secondary accumulator pressure 750 pounds per square inch (psi), leakage, general condition, and security; foreign materials - (ram air turbine (RAT.) kit installed if required); ram air turbine turns freely (clockwise); rat. door will be closed by Inspector after leak check

17 - right-hand main gear well area, tire condition, slippage mark, brake and lines, fairing doors, landing light, gear microswitch under roller, rate gyros warm and turning (external power ON). All circuit breakers IN, strut extension between axle scissors bolt 5-5/8 inch. Static trailing wire grounded

18 - right-hand engine compartment - leaks - fire warning conduit, flight controls

19 - right wing (lower side) fuel panels for leaks, elevon actuator for leaks, and fairing binding

20 - right-hand fillets for elevon rubbing

21 - vertical fin condition and drag chute installation

22 - rudder area for leakage

23 - afterburner nozzles for closure and warpage

24 - afterburner flame holder and general condition of tail pipe

25 - left-hand fillets for elevon rub

26 - left wing (lower side) fuel panels for leaks, elevon actuator for leaks, and fairing binding

27 - left-hand engine compartment - leaks, condition and security of all lines, Shelly unit pin (flush), and throttle linkage

28 - left-hand main gear well area - tire condition, slippage mark, brake and lines, fairing doors, landing light, gear microswitch under roller, pitch gyro warm and turning (external power ON), strut extension between axle scissors bolt 5-5/8 inch static trailing wire grounded, high pressure pneumatic system gage 3000 psi, ±200 psi, combustion starter selector switch for external air start and pre-charge missile bay doors (circuit breakers IN, depress switches and receive amber and green lights)

29 - left wing leading edge and wing fences

30 - left missile doors, hinges, and contour

31 - fuselage external canopy jettison lanyard cover for security

32 - oxygen filler cover on and secure

III

BEFORE ENTERING COCKPIT

1 - wheel chocks in place

2 - bridle installed (ground run only)

3 - area clear (blast area and forward intake areas)

4 - fire truck standing by

5 - canopy hold safety bar installed

IV

ON ENTERING COCKPIT

1 - while on ladder, check entire upper surface of aircraft for damage, loose plates, "Q" intake ports and general condition

2 - mechanical linkage from canopy to arm ejection seat for security and connections

3 - canopy and windshield glass for damage, i.e., scratches, distortion, discoloration, delamination, etc.

4 - initiator pins all removed

5 - arm rests down and locked

6 - right arm rest safety pin installed

7 - canopy jettison handle in detent

V

PRE-START COCKPIT CHECK

1 - external AC and DC power applied

2 - adjust rudder pedals and seat. Check pedals aligned and seat for excessive play (1/4-inch maximum)

3 - fasten shoulder harness and lap belt:

  a - check inertia reel in locked and unlocked position

  b - check force required to open lap belt, should not be less than 5 pounds or more than 15 pounds

4 - personal equipment leads fastened and checked, (oxygen hose, "G" connection if applicable, radio lead, etc.)

VI

COCKPIT LEFT-RIGHT CHECK

1 - left aft circuit breaker panel

2 - spare lamp supply

3 - map case

4 - "G" suit control valve

5 - mask de-fog set as desired

6 - Fuel Control:

  a - tank selector - both tanks to ENGINE, selector knob must be pushed in slightly to turn; also check security of the knob

  b - boost pumps - check one at a time and leave the left forward pump ON to recharge the combustion starter fuel flask for next start

  c - push to test fuel quantity. If totalizer needle moves this indicates system working normal. If button is held the system is grounded and will indicate approximately 200 pounds below the empty mark

  d - check fuel quantity reading against actual fuel on board (total in gallons X fuel specific weight = total pounds on gage reading) tolerance is ±160 pounds and ±4% of indicator reading. Fuel specific weight is 6.5 pounds per gallon @59°F.)

  e - check left and right position readings for each No. 3 tank. (No. 3 tank capacity is 159 gallons of which 157 gallons are usable. To find the correct reading in pounds multiply 157 gallons by fuel specific weight. Tolerance is the sum of ±160 pounds and ±4% of indicator reading.)

7 - ARC-34 - OFF

8 - ram air turbine handle - IN

9 - throttle - OFF

10 - speed brakes switch - NEUTRAL

11 - emergency fuel (turn ON and check for light on master warning panel). Return switch to NORMAL guard down

12 - Oxygen Equipment (MB-2 Regulator):

  a - check pressure (1800 pounds)

  b - check ON - OFF lever for freedom of movement. Set to ON

  c - set 100% or NORMAL

d - press to test EMERGENCY. Check for pressure flow

e - check tear drop blinkers

13 - landing gear control handle DOWN. Gear indicators down and locked. Press test button - warning light in handle should illuminate

14 - Landing - Taxi Light Switch:

a - check taxi light (ground man to check)

b - check landing light (ground man to check)

c - switch to neutral - (OFF)

15 - lower left circuit breaker panel (all IN)

16 - emergency gear extension handle IN. Safety clip installed - if applicable

17 - Cabin Pressure Switch:

a - check installation, should have to pull spring loaded switch out to place in RAM position

b - turn switch to OFF position

c - note that cabin altimeter reads approximately field elevation

18 - drag chute handle IN

19 - needle and ball (reads aircraft attitude)

20 - clock set

21 - altimeter set (field elevation)

22 - rate of climb needle (reads zero)

23 - MM-2 attitude indicator. (Check that adjust knob changes reference and turns freely.)

24 - air-speed indicator (reads zero and index knob turns freely)

25 - Omni course setting knob turns freely. (VHF Nav power switch OFF. Flags should be showing.)

26 - Engine Instruments:

  a - tachometer - zero - check limit markings

    (1) GREEN band - 90%-98%

    (2) RED mark - 102%

  b - exhaust temperature - zero - check limit markings

    (1) GREEN band - 470°-610°

    (2) RED mark at 660° and 670°

  c - ratiometer. Knob turns freely and index moves in the right direction. Check cruise and takeoff position

  d - hydraulic pressure gage: select primary and secondary. Red light should be illuminated, if not, check circuit breaker. Check limit markings

    (1) GREEN band - 2500 psi and 3000 psi

    (2) RED mark - 1000 psi and 3100 psi

  e - fuel flow - zero

27 - check fire warning and overheat circuit, FLASH in over-heat position. STEADY in fire position

28 - radar intensity controls

29 - turn IF gain control DOWN, fully counter-clockwise

30 - anti-clutter switch OUT

31 - storm lights - ON - check lights then turn OFF

  Note: with storm lights ON flight instrument and all warning lights will be at full brilliance.

32 - canopy lock handle OUT. (Light illuminated CANOPY UN-LOCK)

33 - Master Warning Panel:

  a - check TEST position, all positions should illuminate plus master warning light

Note: there are two light bulbs in each panel, if one is burned out only half the labeling will be readable.

b - check RE-SET position

34 - check electrical panel:

a - AC generator - OFF

b - DC generator - OFF

c - battery - OFF

d - AC bus - Normal (guard down)

e - master - Normal (guard down)

35 - check VHF Nav:

a - power switch OFF

b - selector knobs turn freely

36 - J-4 Compass Control:

a - N on hemisphere selector

b - set latitude dial (34° @ PMD and 33° @ SAN)

c - select DG position and turn compass card by the SET ± control. Turn off heading 10-40°

d - select mag and note that the compass card fast slaves to the approximate aircraft heading

37 - check list (security and legibility)

38 - check Identification Friend or Foe (IFF) control for selection, installation, then to STANDBY

39 - check right circuit panel (all breakers IN)

40 - Armament Panel:

a - arm switch SAFE - (guard down)

b - arm selector SNAKE

41 - Automatic ~~Frequency~~ FLIGHT Control System (AFCS) Control Panel:

- a - direct manual (yaw damper/turn coordinator) - OFF
- b - altitude hold - OFF
- c - AFCS - OFF
- d - AILAS - OFF

42 - Radar:

- a - elevation scan - NEUTRAL
- b - azimuth scan - BROAD
- c - power - OFF
- d - 30 mile search

43 - Center Pedestal:

- a - pitch damper - OFF
- b - trim servo - OFF
- c - Nesa - NORMAL (must pull to place OFF)
- d - temperature control - check auto and manual positions
- e - exterior lights - OFF
- f - anti-ice - OFF
- g - de-fog - OFF
- h - rain clear - OFF
- i - pitot heater - OFF
- j - all light rheostats ON for approximately 15 minutes. Check for shorts, etc. (may be turned OFF during flight)
- k - make general quality check of cockpit (splitter screen, paint, glare shield, scope alignment, etc.)

VII

PRE-START CHECK

1 - emergency fuel OFF (warning light OUT)

2 - fuel selector (both tanks to ENGINE)

3 - left and right forward pumps ON

4 - master switch ON

5 - battery switch ON - (to assure start if external DC power fails)

6 - throttle - OFF

7 - signal for external air to be turned ON

Note: One start will be accomplished utilizing the aircraft's high pressure pneumatic system during Convair shakedown.

VIII

START

Depress ignition button and hold. Move throttle outboard. When combustion starter fires, move throttle inboard and up to IDLE in one continuous motion. Continue holding ignition button until starter cuts out at approximately 30%. START may be aborted at any time by releasing ignition button and placing throttle in OFF position.

Check:

1 - idle rpm 55-65% N_2

2 - fuel flow 800 pph minimum

3 - TT_7 340° maximum stabilized IDLE

4 - oil pressure warning light - OUT

5 - hydraulic pressures 3000 ± 150 psi

IX

POST START CHECK

1 - signal for external starter air to be removed

2 - AC generator switch - re-set and ON

3 - DC generator switch - re-set and ON

4 - signal for external electrical power to be removed

Note: Check master warning panel. If AC or DC warning light illuminates, re-set generators.

5 - Check electrical system as follows:

a - place DC generator switch OFF. Check that master warning panel reads DC power failure. (If not, check external power removed). If warning lights remain illuminated, battery is connected. Generator switch back to ON position.

b - place battery switch to OFF. If warning lights remain illuminated the DC generator is operating.

Note: If at any time all DC power is lost the armament system must be recharged.

c - AC generator switch OFF. Master warning panel should read AC POWER FAILURE. (If not, check external power removed).

d - place AC bus switch to EMERGENCY. Check voltmeter selector. Should read 103-140 volts on all phases except 26 volts which should read 26 volts. Note boost pump lights ON. (Non-essential bus).

e - AC generator switch to ON. Then turn AC bus switch to NORMAL. (Check AC warning light OUT). Check all phases, should read 115 volts ±3 on all phases, 26 volts position should read 26 volts.

6 - electronic cooling light, OUT. (If not OUT advance throttle slightly and return to idle, light should go out).

7 - Radar to STANDBY or ON

8 - ARC-34 ON

9 - boost pumps ON individually and note AC voltage. (Aft pumps first and note lights OUT).

10 - Flight Control Checks:

a - move rudder pedals full throw and check for binding, pedal force, correct rudder travel, and centering

b - deflect ailerons fully (7°) both directions and check correct travel, centering, and the following forces:

(1) break-out force (2 pounds ±1/2 maximum)

(2) 5° stop (force is 13 pounds ± 1 1/2)

(3) 7° stop (8 pounds increase in force)

c - deflect elevons fully both directions (up 25°, down 8°) and check correct travel, centering, and the following forces:

(1) break-out force (2 pounds ±1/2 maximum)

(2) up and down stops (force under static conditions is approximately 40 pounds ±4 up and 12 pounds ±3 down)

Note: Control break-out forces check. By use of fish scale check the force necessary to start control movement. It will be necessary to station a man at the elevon to signal when movement is started.

11 - Hydraulic Check:

a - select primary system on the gage and from neutral move the stick full forward, full aft, and back to neutral, in one continuous motion. The pressure should not fall below 1200 psi and hydraulic warning light should not illuminate.

b - select secondary system and check the same as above, except the pressure should not fall below 1000 psi.

c - engage trim-servo; operate speed brakes through two complete cycles and check gage for normal fluctuation, trim servo tie in and restrictor action.

d - during all stick movements the hydraulic pressure should stabilize at 3000 pounds ±150 pounds within two seconds after control movement is stopped. (Recovery rate is the most important check of the hydraulic systems).

12 - Damper Check:

a - engage manual mode (yaw damper and turn coordinator)

b - engage pitch damper

c - engage trim servo

d - repeat controls full throw check

Note: Rapid lateral movements of stick may induce rudder kicks. This condition normal during ground checks only.

e - press Emergency Direct Manual on right-hand grip. Manual mode, pitch damper, and trim servo should disengage.

13 - Trim Rate Checks:

a - rudder: authority is 7 1/2° L and R. Rate is 1.5°/second or total throw (15°) in 10 ±4 seconds.

b - aileron: authority is 3° up or down. Rate is .6°/second or total throw (6°) in 10 ±4 seconds.

c - elevator: authority is 2°down and 9° up. Rate is 1.1°/second or total throw (11°) in 10 ±4 seconds.

14 - Take-Off Trim

Check from maximum deflection each way for dead spots, i.e., right aileron, forward elevator and right rudder. Surfaces should center in the following order: aileron, elevator, rudder, and take-off trim light should illuminate. Take-off trim limits are elevator 5° up ±1/2°, rudder 0° ±1/2°, and ailerons equal. Visual reference to these surfaces may vary due to manufacturing tolerances within the elevon and wing tip installation.

15 - Automatic ~~Frequency~~ Flight Control System (AFCS) Check:

Note: AFCS may be ground checked and operated in flight with radar in STANDBY or ON.

a - fast erect radar horizon upon completion of 4 1/2 minute warmup cycle.

b - engage manual mode, pitch damper, and AFCS in this order.

Note: Engage AFCS with momentary interrupt switch depressed.

c - release MI, check AFCS remains ON, controls remain stable, and that lockout valves engage after 1 1/4 second time delay.

d - depress MI and move controls laterally and longitudinally. Control movement should be normal and AFCS will disengage during lateral movement as ailerons reach limit switch at approximately 3° left and right.

e - beep trim laterally left and right. AFCS and pitch damper will disengage as ailerons reach limit switches at approximately 3° left and right. Pitch damper may not disengage due to collapse of lockout valves causing premature AFCS disengagement via lockout switches. Premature disengagement is not considered a discrepancy as long as 3° aileron movement is not exceeded.

f - reengage AFCS and beep trim longitudinally. Trim will be normal through full trim authority and AFCS will remain ON. Stick will continue to move approximately 1.5 seconds after release of beep button due to smoothing circuitry. Trim follow-up may also be checked at this time by turning AFCS switch OFF. If trim follow-up is satisfactory, controls will hold the beeped trim position. A slight stick transient is allowable (0.4° maximum). Do not beep trim beyond normal trim limits.

g - manually over-power AFCS by moving controls laterally and again AFCS and pitch damper should disengage up on reaching aileron limit switches (10-15 pounds stick force). Pitch damper may again not disengage due to reasons explained in step e - above.

Note: Abrupt up elevator should disengage AFCS and abrupt down elevator may or may not disengage AFCS. This is a by-product of low force lockouts and lockout switches. Such disengagement is not necessary for protection of the aircraft since the pitch "G" limiter provides this safety feature.

h - reengage AFCS. Hit EMERGENCY DISCONNECT on stick. This should disengage manual mode, pitch damper, trim servo, and AFCS.

i - re-set take-off trim, check elevons visually.

16 - Nesa Control: NORMAL, note that glass warms and that no excessive distortion, discoloration, delamination, and/or cracks appear.

17 - Anti-Ice: place switch in the MANUAL position at stabilized idle and note rpm drop. Place switch directly to AUTO and note that the rpm recovers. The master warning panel light should be OFF in manual and auto positions. Do not attempt flight if engine rpm cannot be regained with switch in OFF position. While making this check no glycol flow should be noted as this is rendered inoperative by main gear door close switch.

18 - mask de-fog: set at approximately 12 o'clock for pressure suit flight

19 - Radar Pre-flight check. (See Section III).

20 - ARC-34: make radio check on channel 10 and obtain mean ratiometer setting

21 - ratiometer: set ratiometer (T. O. window).

22 - Canopy: signal for the ground crew to remove canopy hold fixture and perform the following:

a - depress canopy hold button and check counter balance. The force required to move the canopy up or down should not exceed 15 pounds.

b - release button with the canopy in an intermediate position and note that the canopy will hold. (If taxi hold position is inoperative, have crew chief check external release button on upper left duct).

c - close canopy and allow canopy to cinch down - the force required to initiate cinch down action should not exceed 15 pounds. Canopy should cinch down in approximately 4 seconds from 2 inches to 6 inches above longeron

d - push lock handle in - the canopy locks should not bind on the rollers, the handle should move freely, and the light should extinguish.

e - pull canopy handle out and note that cinch down pressure is relieved and that canopy will open freely

f - taxi with canopy as desired

X

TAXI CHECK

During taxi electronic cooling will cease above 75% N_2 rpm and a noise level change will be noted.

1 - Nose Wheel Steering:

a - check directional control and centering by depressing Nose Wheel Steering (NWS) button and operating rudders left and right.

Note: Turning of nose wheel by steering button is limited to 50° left or right of center, at which point nose wheel is in free wheeling, and operation of the NWS switch is overridden by braking action. Braking action must be applied to bring the nose back to the 50° range, while at the same time holding the NWS switch depressed, in order to regain power steering.

2 - Brakes:

make functional check of the brakes for pedal force, uneven response, and drag

3 - Nose Wheel Strut:

should not bottom during normal brake application

XI

TAKE-OFF CHECK

1 - Emergency Fuel System:

a - with engine at idle, select fuel control switch to EMERGENCY. The emergency fuel ON light should illuminate, and fuel flow should read 750 to 1050 pph.

b - advance throttle to full military power and check fuel flow. Minimum is 6400 pph at San Diego (sea level) and 6000 pph at Palmdale (elev. 2550 ').

Note: TT_7 below 640°.

c - retard throttle to 80% N_2 rpm and reselect NORMAL fuel system. (Note emergency fuel warning light OUT).

2 - Power Check:

a - advance throttle to military power noting any binding or restriction in throttle movement b - check ratiometer against calculated value. (May take 30 seconds to peak).

Note: If this reading is questionable, stabilize engine at full military for 3 minutes and re-check.

c - check N_2 rpm. (Max. 102%)

d - check that brakes hold in full military power

e - check fuel flow (6000 min.)

f - check exhaust gas temperature. 670° maximum on acceleration and 640° maximum stabilized

XII

TAKE-OFF

1 - engage nose wheel steering button

2 - release brakes and assure NWS is engaged

3 - light afterburner by moving power lever outboard. If exhaust nozzle fails to open when afterburning starts there will be a rapid rise in temperature and loss of approximately 4% in rpm. If evidence of either of these conditions exists terminate afterburning immediately by moving power lever inboard

4 - check fuel flow, TT_7, and N_2 rpm during initial takeoff roll

5 - raise gear when safely airborne and check indicators for UP indication

XIII

CLIMB

1 - continue afterburning climb maintaining airspeed below 300 knots indicated air speed (IAS) on green airplanes; keep close surveillance on all engine instruments and be prepared to select the emergency fuel system if a drop in N_2 rpm and fuel flow is indicated

2 - during climb-out trim aircraft in pitch only, then at 300 knots IAS release stick and note lateral roll-off and/or directional trim shift; if lateral roll-off exceeds 30° per second or directional trim shift exceeds 1/2 ball width record the amount and direction as a flight squawk

3 - shutoff afterburner by moving the throttle inboard; check ratiometer to determine if nozzle closed

4 - relight afterburner by moving the throttle outboard; check time to light (2 seconds maximum) and engine instruments to determine if nozzle opened

5 - retard throttle to aft stop in afterburning range; check continuance of afterburning in power range; (min. 2% N_2 rpm)

XIV

GEAR WARNING LIGHT

Note: The landing gear warning system has 3 switches incorporated to illuminate the warning light under all landing situations in the event the landing gear is not in the down and locked position. One micro-switch is located at the military stop on the power quadrant, one meletron switch is actuated by air-speed and one meletron switch is actuated by altitude.

1 - moving the power lever to full military will put out the light, as will increasing the airspeed above 321 knots or altitude above 13,500 feet ±1000 feet, or at any time the gear is down and locked

2 - during descent the light will come on at 10,000 feet ±500 feet if the air-speed is below 250 ±15 knots, providing the throttle is aft of the military stop

XV

COCKPIT PRESSURIZATION (Level Off 15,000 feet)

1 - check cockpit altimeter should read 10,000 feet ±320 feet

2 - check for fumes in cockpit

3 - select OFF. (Pressure should bleed slowly to approximately 12,000 feet) with no in-flow to the cockpit).

4 - select RAM; the dump valve should operate; some ram pressure will be noted depending on airspeed. The cabin altimeter will read approximately 13,500. If suspected that the cabin dump valve did not operate, re-pressurize, select switch straight to RAM and pressure should dump immediately.

5 - re-select PRESS and note that the cabin altimeter returns to 10,000 feet ± 320 feet within 45 seconds

XVI

COCKPIT TEMPERATURE CONTROL

1 - select AUTO-HOT and note temperature change (this may require 3 minutes)

2 - select AUTO-COLD, note that inflow air cools (this may also require 3 minutes)

3 - select MANUAL-HOT, the temperature change should be apparent within approximately 30 seconds

4 - select MANUAL-COLD (cool air should be felt almost immediately)

5 - select comfortable level

Note: The rheostat knob must be held in the detent position for MANUAL HOT and COLD.

XVII

SPEED BRAKES

1 - operate speed brakes through a complete cycle; note that the opening and closing is symmetrical

2 - check pitch response; pitch up will be noted on extension and slight pitch down on retraction if restrictors installed

3 - re-check with trim servo ON; this should compensate for pitch changes

XVIII

LANDING GEAR

1 - cycle gear at 240 knots IAS; gear should extend and lock in 6 ± 1 seconds

2 - observe secondary hydraulic system, aircraft response, and noise level

3 - at least once during shakedown, the gear should be retracted by the EMERGENCY RETRACT BUTTON; depress the button with the gear handle in the down position and the gear should retract in 6 ± 1 second. The red light will remain on in the gear handle; regain control of the normal system by placing the gear handle in the UP position

Note: The EMERGENCY RETRACT BUTTON does not have to be held in for gear retraction when airborne.

XIX

TRIM

1 - check rudder trim, left and right

2 - aileron trim, left and right

3 - elevator trim, nose up and down

4 - actuate takeoff trim-button; there should be no aircraft response

XX

CLIMB TO 26,500 FEET CHECK

1 - time turns: check turn and bank indicator during climb; hold 1 needle-width turn; aircraft should turn 45° in 30 seconds (±5 seconds); check both left and right

2 - check airspeed indicator against Mach indicator; when climbing through 22,500 feet at 300 knots IAS, Mach indicator should read .68 ± .01.

Note: This is only an instrument check as the Mach meter and the airspeed both obtain information from the same source.

XXI

LEVEL AT 26,500 FEET ALTITUDE CHECK

1 - Cabin Altimeter: should read 10,000 feet (±320 feet) at full differential pressure (5.0 ±0.1 psi)

2 - Electrical Check: check a-c voltages; all phases should read 115 (±3) volts, except 26 volts which should read 26V; check d-c power by testing any warning light.

3 - Emergency Fuel System:

a - at 80% N_2 rpm, switch to the EMERGENCY fuel system; advance throttle to full military; check N_2 rpm, fuel flow and TT_7

b - cycle afterburner, check re-light, aft stop for blowout, and afterburner range (2% N_2 rpm)

c - retard throttle to 80% N_2 rpm, re-select the NORMAL system

XXII

CLIMB TO 35,000 FEET ALTITUDE AND ESTABLISH LEVEL FLIGHT

1 - trim aircraft hands off to 0.8 Mach

2 - engage manual mode; check engage transient. (1/4 ball maximum)

3 - manual mode OFF; check dis-engage transient

4 - push rudder and note natural aircraft damping

5 - manual mode ON; push rudder as above and note damping to be approximately dead beat

6 - engage pitch damper; check for engage transient; (any pitch transient is unacceptable)

7 - pitch damper OFF; check disengage transient

8 - induce pitch oscillation and note natural damping

9 - pitch damper ON; induce pitch oscillation and note damping to be approximately dead beat

XXIII

VMAX LEVEL AT 35,000 FEET

Check the following:

1 - Manual Mode:

a - engage manual mode

b - accelerate to Vmax level; this will normally be approximately 1.16 Mach at standard temperature (-55°C at 35,000 feet). For each degree off standard add or subtract .005 Mach.

c - check that all short period oscillations are damped through entire speed range

Note: Aileron forces should be equal to those of lower speeds. Elevator and rudder forces should increase.

2 - Trim:

a - during acceleration note lateral or directional trim changes; directional trim changes displacing more than 1/2 ball are unacceptable; lateral roll-off that requires aileron re-trimming in such an amount as to reduce level Vmax is unacceptable; if either of these conditions exist repeat check in direct manual mode

Note: In direct manual mode lateral/directional short period oscillations will persist during acceleration or deceleration through 0.9 Mach to 1.1 Mach; a loss of elevon effectivity will be noted at approximately 0.92 Mach with trim servo OFF causing tuck down during acceleration and pitch up during deceleration.

b - if any unacceptable engage transients or trim changes exist as stated above, then prior to landing, trim aircraft hands off level, 300 knots IAS at 10,000 feet altitude; do not retrim for remainder of flight; advise maintenance crews of this set trim condition before leaving aircraft

3 - Trim Servo:

a - check that during acceleration trim servo induces a nose-up trim from 0.8 to 1.3 Mach

b - check that during deceleration trim servo action is nose-down

Note: The normal trim should always be operational and over-ride the trim servo when engaged.

XXIV

AFCS CHECK

Note: As of this writing only heading hold and roll hold modes are operational.

1 - airborne engagement of AFCS is the same as the ground procedure; however, it is recommended that the aircraft be in trimmed straight and level flight with the radar horizon properly erected to wings level ±3° prior to engagement

2 - upon releasing MI switch aircraft should hold attitude corresponding to radar horizon; aircraft response should be stable; a 1 1/4 second delay will be noted after releasing MI before AFCS engages

3 - depress MI and fly to various attitudes, release MI, note 1 1/4 time delay, and aircraft stability in new attitude

Note: Roll hold will not be operational for angles of bank less than 5° or more than 71°. Attitude should be established prior to release of MI. AFCS will not engage at pitch angles in excess of 77°.

4 - a check of heading hold may be made by establishing a desired heading; the aircraft should maintain this heading under stable conditions and within the allowable limits of the J-4 compass; heading hold should be checked in both MAG and DG position of the J-4 compass

5 - speed and altitude changes should not adversely effect AFCS operation; do not exceed 480 knots IAS below 12,000 feet with AFCS engaged

6 - check longitudinal beep trim for correct aircraft response and trim follow-up

7 - check lateral beep trim for correct aircraft response; a .3 second delay will ensue upon release of the trim button within either heading hold or roll hold; a 1 1/4 second delay will ensue when trimming from one mode to another

8 - the recommended procedure for checking the pitch G limiter is to depress MI and enter a wind-up turn at approximately 450 knots IAS; AFCS should disengage at a positive 4.5 G's or a negative 1.5 G; however, due to accelerometer error, and varied pilot technique a tolerance of ±1 G will be acceptable; check AFCS switch OFF and that lockout valves have released

XXV

CLIMB TO 45,000 FEET

1 - check N_2 rpm

2 - check exhaust gas temperature (660° maximum above 35,000 feet, 670° for 2 minutes on acceleration)

3 - check cabin altimeter at 40,000 feet; should read 16,000 feet ±1,000 feet

4 - check partial afterburner power

XXVI

LEVEL AT 45,000 FEET ALTITUDE CHECK

1 - engage manual mode

2 - accelerate to Vmax level; (this will normally be approximately 1.1 Mach at standard temperature (-55° C.)

3 - check N_2 rpm; (due to temperature N_2 rpm will drop approximately 2%)

4 - check exhaust gas temperature; (660° maximum continuous, 670° allowable for 2 minutes on acceleration)

5 - check aircraft trim and flight controls

XXVII

VNE DIVE

1 - trim servo OFF

2 - manual mode engaged and Pitch Damper engaged

3 - obtain Vmax level at 45,000 feet or above

4 - dive to Vne of Mach 1.5 or 655 knots IAS, whichever occurs first

5 - check noise level at Vne

6 - check stability and control

7 - check anti-G suit equipment in dive recovery

8 - reduce power to military, extend speed brakes, and check aircraft response in slowing to subsonic speed

Note: Do not exceed 5.6 G in symmetrical pull-out or 4.0 G in an asymmetrical pull-out. Zero G is limited to 3 seconds.

XXVIII

HIGH ALTITUDE CHECK

Once during shakedown processing the aircraft will be flown in-excess of 50,000 feet for the following checks:

1 - engine performance

2 - afterburner blowout

3 - radar operation

4 - cabin pressurization

Note: The MC-1 partial pressure suit will be worn for all flights above 45,000 feet.

XXIX

ESTABLISH DESCENT

1 - check Nesa effectivity; distortion and discolorization not excessive

2 - check canopy de-fog ON; air should blow out canopy upper center de-fog outlets; cockpit noise level will increase slightly

3 - at 35,000 feet retard throttle rapidly to stabilized idle, then advance rapidly to full military; if compressor stall occurs, use the following procedure:

a - throttle - IDLE

b - reduce to subsonic speed and decrease altitude

c - if stall continues and engine starts to over-temperature, throttle - OFF; make a normal air start

XXX

ROLL

1 - perform aileron rolls left and right; check for aileron centering and normal aircraft response; roll rate is unrestricted above 20,000 feet and restricted 1/2 (3.5°) aileron throw below 20,000 feet in excess of 450 knots IAS

Note: Do not perform maximum deflection aileron rolls between 0.88 Mach and 1.0 Mach without operative stability augmentation. Radar should be in standby position during maneuvers. This condition prevents gyro damage and maintains the antenna in a rigid position.

XXXI

LOW ALTITUDE SPEED RUN

1 - attain 0.95 Mach maximum between 5,000 feet and 10,000 feet above terrain

2 - check stability augmentation performance

3 - check aircraft trim

4 - check flight controls for centering, forces, and response

XXXII

ARC-34 RADIO

1 - while in flight check communications on several channels through the frequency range of the set: (225.0 to 339.9 mc)

2 - check operation for receiver squelch; the squelch should function to keep receivers quiet except during reception of signals above the preset level

3 - check for presence of sidetone in the headsets

XXXIII

IFF CHECK

1 - select Channel 12 (364.2) "Duly" or "Swanee" and check operation on all modes

XXXIV

ARN-12 MARKER BEACON RECEIVER

1 - check operation of the marker beacon receiving set as aircraft is flown at 1,000 feet and 25,000 feet over a fan-type marker of CAA instrument landing markers

2 - check at outer and inner markers during ILS approach

XXXV

ARN-14 OMNI-DIRECTIONAL RANGE SYSTEM

1 - General communication, omni-directional range signals and course indicator should constantly indicate direction of omni-range station from aircraft, as measured from magnetic North. The compass card, controlled

by the remote indicating magnetic compass, should rotate so that magnetic heading of aircraft is indicated against reference marker at top of indicator face. Vertical pointer of the course indicator should indicate lateral positional deviation from the selected omnirange station. Warning flag should prevent false on-course indications by obscuring part of center-pointing vertical needle of course indicator when airborne equipment is not receiving the proper omnirange signal. Complete operational check as follows:

a - turn on VHF NAV system and set VHF frequency selector to 112.2

b - check for station identification

c - the vertical pointer of the course indicator should give "on-course", "fly-left" or "fly-right" indications

d - check volume control and set at desired level

e - establish a known inbound heading to station by varying aircraft heading so that the No. 2 needle on RMI falls under top indice

f - hold above heading and set same heading on OMNI indicator with course indicator knob; the OMNI needle should be centered and the RMI reading should equal ID249 window reading ±2 degrees (no wind condition)

g - establish drift correction, if necessary, and cross-check that OMNI deviation needle reading corresponds to RMI card needle reading ±2°; check that OMNI needle deflection is correct directionally

h - when flying over station check that RMI No. 2 needle swings 180° and that the TO-FROM indication is correct; also, check for right or left deflection on OMNI needle

Note: When crossing over station, the OMNI flag may flicker in and out; this condition is normal.

XXXVI

J-4 COMPASS CHECK

1 - check in Mag and DG positions on a known heading (±2°)

2 - check in Mag and DG positions that when aircraft turns, compass card rotates freely and in proper direction; check in both directions

XXXVII

MAGNETIC (STANDBY) COMPASS

1 - ±7° of J-4 cardinal headings

XXXVIII

LOCALIZER AND GLIDE PATH (ARN-18) OPERATION

1 - these functions will be checked on the Palmdale ILS facilities

2 - trim airplane in level flight at 1,500 feet above terrain and at an IAS of 220-240 knots

Note: Extend landing gear at outer marker.

3 - set VHF frequency selector to 109.1

4 - the leg of the ILS shall be approached from approximately 8 to 14 miles out at an 89° angle or less

5 - when the localizer (vertical) needle on the course indicator starts to move from the stop, make a 35° bank into the localizer beam and assume proper heading; at this time, glide path needle will be riding at top of indicator; when approaching the outer marker the glide path needle should start moving toward center of indicator and be approximately centered at outer marker

6 - check aircraft altitude at outer (approximately 3,900 feet @PMD) and middle (approximately 2,800 feet @PMD) marker; also, check for alignment on runway

7 - with glide path needle centered, maintain the proper course and follow glide path

8 - the landing approach should be discontinued at 200 feet above terrain minimum altitude

XXXIX

EMERGENCY GEAR DROP

1 - this check, if satisfactory, need only be made once during company shakedown; this check to be conducted only if the normal extension and retraction of the landing gear was satisfactory; the following procedure will be used:

a - normal gear handle UP

b - pull emergency gear handle and note that clip, if applicable, holds handle out

c - gear extends and locks within 15 seconds

d - when gear is down and locked place normal handle in the down position

Note: (1) when gear is dropped by emergency, the nose wheel steering is inoperative

(2) should the gear not extend and lock, the following should be used:

(a) normal handle UP

(b) push in emergency gear handle

(c) allow 1 1/2 minutes for the air to bleed off; the gear will not retract immediately but will try to retract 5 or 6 times before they will indicate UP and locked

(d) after gear is UP and locked, lower gear by the normal hydraulic system; if the gear still does not indicate down and locked; pull the emergency gear handle; have a visual check made by the tower or other aircraft; alert emergency crews via tower and land

(e) if after landing the gear still indicates unsafe, stop on the runway and have landing gear pins installed

XL

RAM AIR TURBINE CHECK

1 - select PRIMARY hydraulic gage

2 - extend RAT during landing pattern; (maximum extension 345 knots)

3 - check that under no control movement the gage readings are above the following:

(1) 200 knots - 2450 pounds

(2) 180 knots - 1800 pounds

(3) 160 knots - 1350 pounds

(4) 140 knots - 1000 pounds

XLI

LANDING

1 - if gear is to be extended by emergency system, it is recommended that a simulated flame-out pattern be flown to allow more time for gear drop and RAT check

2 - after touch-down check the following:

a - deploy drag chute; (maximum deployment time - 4 seconds)

b - check jettisoning operation when clear of the runway

c - with speed brake switch IN, re-pull handle noting the secondary hydraulic system gage, the brakes should open; with handle pulled, operate speed brake switch; speed brakes should remain OPEN

XLII

POST LANDING CHECK

1 - Engine: if an engine re-check is required, obtain Palmdale Tower permission to stop on taxi strip for run-up

2 - Operate trigger salvo. There should be no door action due to nose wheel door microswitch which prevents firing when door is in open position

3 - boost pumps OFF

4 - VHF NAV - OFF

5 - radar OFF

6 - pressure - OFF

7 - canopy OPEN, if desired; maximum speed 50 knots

8 - master switch OFF; check that all electrical power is OFF the aircraft; turn back ON and re-set AC and DC generators

9 - Fuel System:

a - low level warning: at least once during company shakedown this system should be checked by running both No. 3 tanks down to low level indication; the warning lights should illuminate at 570 pounds ±160 pounds and ±4% of indicator reading

b - left and right tank operation: select left and right tank position and note normal engine operation and that fuel level in No. 3 tank on selected side drops; it is necessary to have fuel in No. 3 tanks left and right only in order to detect drop in No. 3 tank level

c - flow divider: select No. 3 tanks left and right; maximum differential in readings will be 300 pounds unless afterburner use was above average during the flight

d - fuel quantity: select No. 3 tanks left and right; the total of those two readings should be within 160 pounds and ±4% of indicator total reading

Note: If the fuel quantity indications are questionable, check on actual amount of fuel required for reservicing the aircraft.

XLIII

SHUTDOWN

1 - with external power applied and ground crew positioned, retard throttle OFF, monitor shutdown, check fuel flow zero, normal engine deceleration, and no rise in TT_7

Note: Engine cool-down period is 5 minutes. This time should have elapsed by the time aircraft is in the chocks.

2 - operate speed brakes; this will bleed the secondary hydraulic system below 800 psi prior to the primary system and give a reliable check on the hydraulic warning flasher unit

3 - work controls to work primary system below 800 psi; check hydraulic warning light goes steady at approximately 800 pounds

4 - speed brakes closed; hydraulic pressure zero; rpm zero; extend, rotate, and pull drag chute handle approximately 1/2 inch to check that pneumatic system will open speed brakes; push drag chute handle IN

5 - turn fuel selector OFF

6 - install seat pin

7 - disconnect personal leads and insure lap belt pin is clear

8 - check to see canopy fixture is installed prior to leaving aircraft

9 - make short visual check of aircraft for leaks, pressure readings, etc.

XLIV

POST FLIGHT

1 - Short Post Flight Briefing to Crew Chief for fuel, engine, hydraulics, gear, flight controls, and pneumatic discrepancies; this short briefing is necessary and important for proper ground disposition of the aircraft.

2 - Post Flight Meeting with Foreman and Liaison Engineers for further discussion and elaboration on flight discrepancies.

3 - Complete Convair Form 1860 in triplicate.

4 - No. 2 copy to Inspection; No. 3 copy to Shop; return original copy to Production Flight Office (Flight Dispatch).

5 - Complete aircraft folder.

THE PRODUCTION FLIGHT TEST PROCEDURES AND SPECIFICATIONS FOR THE F-102A APPLY TO THE TF-102A WITH THE FOLLOWING EXCEPTIONS:

LOADING

1 - Weight and Balance: cg limits 26.4% to 27.5% mac.

2 - Note limitations and engineering shortages.

EXTERNAL CHECK

1 - check side slip transducer probe for alignment and security.

POST START CHECK

Canopy: signal for the ground crew to remove canopy hold fixture and perform the following:

- a - rotate canopy latch handle counter-clockwise to vertical position and check counterbalance
- b - rotate canopy latch handle clockwise to horizontal with canopy in an intermediate position and note canopy will hold
- c - rotate canopy latch handle to vertical and close canopy; push in handle and note cinchdown during initial travel
- d - push latch handle full in; the canopy locks should not bind, both lock over-center indicators aligned, and canopy unlock light extinguished
- e - pull canopy handle out (approximately 8 inches) and note latches disengage and cinch-down is relieved
- f - taxi with canopy as desired

Note: Do not taxi with the canopy open unless canopy latch is in lock (horizontal) position.

IN-FLIGHT CHECKS

1 - all switches and controls will be functionally checked from both sides

2 - the emergency landing gear retract button cannot be checked under normal conditions

3 - note canopy buffet between 0.82 Mach and 0.87 Mach. Buffet will increase with higher angles of attack, ~~higher~~ lower altitudes, and accelerated maneuvers

4 - Vmax level at 35,000 feet will be approximately 0.92 Mach

5 - VNE Dive speed is 510 knots IAS or 1.3 Mach, whichever occurs first

6 - altitude specification is 48,500 feet

7 - maximum differential between airspeed indicates at 300 knots IAS is ±5 knots

8 - maximum differential between J-2 Compasses is ±2°

9 - maximum differential for altimeters is ±175 feet at 10,000 feet indicated altitude

SECTION III

RADAR FLIGHT TEST

SPECIFICATION CONFORMANCE

The following outlines MG-3 Radar Test Procedures which are conducted in a manner to afford compliance with Test Specification ZN-8-004 and ZN-8-005 as amended by AMC letter MCPHFC (RDZ-1SFG/2-6) dated 11 August 1955.

I

BEFORE START WITH POWER APPLIED TO AIRCRAFT

1 - armament switch to SAFE

2 - salvo selector switch to SNAKE

3 - elevation control to DETENT

4 - azimuth scan control to B

5 - radar power switch to OFF

6 - radar mode switch

  a - check light in 30 mile and 6 mile position

  b - check for no light in the ground map and beacon positions

  c - return switch to 30 mile position

7 - intensity controls left and right turn to mid position

8 - IF gain control full down (counter clockwise)

9 - radar hand control; check locked position; unlock hand control

10 - elevation roll switch to DETENT

11 - check electronic cooling light ON

Note: Forward instrument panel lights rheostat must be turned clockwise to illuminate the 6 mile and 30 mile mode lights.

II

AFTER ENGINE START

1 - AC power check 115 ± 3 volts

2 - electronic cooling light OFF

3 - radar power switch ON

4 - start clock: no presentation for 30 seconds, then the artificial horizon and the range sweep race will appear; the range sweep trace will sweep

5 - depress and hold the action switch on the hand control; adjust right intensity control counter clockwise until the range sweep trace disappears; then turn the RT intensity control clockwise until the range sweep trace is barely visible; while still holding the action switch, press the return to search button; release the action switch; release the return search button

6 - turn IF gain control full up (clockwise)

7 - check the delay timer: 3 minutes, 15 seconds to 4 minutes, 40 seconds for main bang and ground clutter to appear on the scope

8 - adjust left intensity control until the artificial horizon is at a comfortable level; hold horizon knob IN if necessary to erect the horizon, (± 10 degrees 90 seconds maximum); center the artificial horizon by rotating the horizon centering knob

9 - depress action switch and place range gate marker on main bang; check that attack display appears momentarily then goes to break away signal

Warning: If electronic cooling light does not go out do not turn on radar: abort the flight.
If electronic warning light illuminates: turn radar off and abort the flight.

III

IN FLIGHT

1 - check AC voltage 115 ± 3 volts

2 - rotate artificial horizon centering knob full clockwise and counter clockwise; check that the artificial horizon has approximately 2 inches of travel and that it passes through the center position of the scope; re-center the horizon

3 - set the azimuth scan switch successively to "L" "C" "B" and "R"; check that the range sweep trace paints corresponding areas on the scope

"L" 65°L to 10°R
"C" 35°L to 35°R
"B" 65°L to 65°R
"R" 10°L to 65°R

Reset azimuth scan switch to "B"

4 - vary elevation scan control up and down; ground echoes should appear at the lower antenna angles, and disappear at the higher antenna angles; return el-scan control to detent

5 - 30 mile mode - check that the 30 mile marking is illuminated

a - unlock hand control -- press the action switch; observe that the range gate marker appears on the range sweep trace at 7-1/2 miles

b - move the hand control forward and backward; check that the range gate marker moves from the 15 mile position down to the main bang

c - move the hand control so that the range gatemarker coincides with the altitude mark or the main bang; note that attack display appears on the scope

Warning: Do not attempt this with AFCS engaged.

d - while holding the action switch, move the hand control from side to side, and check that the range sweep trace follows the hand control

e - check the altitude mark appears on the range sweep trace at a position corresponding to aircraft altitude

f - note that characteristic ground or cloud echoes appear on the scope at correct range, azimuth, and elevation

6 - 6 mile mode - check the 6 mile marking is illuminated

a - press the action switch, move the hand control forward and backward

b - check that the range gate marker travels the full length of the range sweep trace

c - while holding the action switch, move the hand control from side to side and check that the range sweep trace follows movement of hand control

d - return to search; check that the altitude mark appears on the range sweep trace at a position corresponding to aircraft altitude

Note: This must be done below 36,000 feet indicated altitude.

7 - return to 30 mile mode --

While holding the action switch actuate the elevation roller switch on the hand control up and down. Check that the aspect of display varies.

8 - return to search

Note: If electronic cooling light illuminates: abort the flight.

IV

GROUND MAP

1 - turn radar mode switch to ground map; check that the 6 mile and 30 mile markings are not illuminated

2 - move the elevation control switch UP; check that the ground display disappears; move the elevation control switch down, check that the ground display reappears

3 - check that the strobes follow corresponding movement of the hand control

4 - fly a heading toward an identifiable terrain feature

5 - turn the elevation scan control to provide a radiation angle appropriate to an estimated range requirement; check accuracy of screen presentation by visual reference to terrain features

6 - move hand control until the lower strobe is just below the reference terrain feature; press the beacon expand switch and check that the expanded display appears on the scope

7 - antenna stabilization check:

a - return to search in ground map mode; move the elevation scan control to detent

b - perform a bank to the right and a bank to the left; place the aircraft in a 15° dive and a 15° climb; while these maneuvers are being performed, check that the orientation of map display on the scope remains constant

V

BEACON OPERATION

1 - set radar mode switch to beacon; check 6 mile and 30 mile light not illuminated

2 - fly a heading toward one of the following stations:

a - March AFB	2-2
b - El Toro	1-2-1
c - Downey	2
d - San Diego	1-2-1-1

3 - set the armament selector switch to 3 RAD or 24 rockets

4 - turn the elevation scan control to provide a radiation angle appropriate to the estimated range requirement; check that the coded beacon response appears on the scope

5 - depress the action switch, move the hand control until the lower strobe is just below the lowest horizontal bar of reply; brighten the reply to best response with the elevation roll switch; press the beacon expand switch and check that the expanded display appears on the scope

VI

ANTI-JAM CHECK

1 - system in auto search and 30 mile mode

2 - press anti-jam button and hold at 2 second intervals; check that targets and indicator trace remain on scope

Note: A slight variation in target definition and intensity may be noticed during this check.

THE INTERCEPTOR AND THE TARGET AIRPLANE WILL RENDEZVOUS OVER PALMDALE AIRPORT AT AN ASSIGNED ALTITUDE. AN IN-TRAIL POSITION WILL BE FLOWN BY THE INTERCEPTOR DURING SNAKE MODE CHECKS. THE FLIGHT PATTERN SHOWN IN FIGURE 1 WILL BE USED FOR SYSTEM CHECKS IN THE ATTACK MODES.

VII

SNAKE MODE OPERATION - Check in the 6 mile mode with armament switch to arm.

Set armament selector switch to SNAKE and arming switch to ARM. Follow and maintain a constant distance from the target airplane.

1 - press action switch on hand control and lock on to target airplane at approximately 5,000 yards; check that attack display appears on scope

2 - check range rate arc at zero position on indicator and range circle remaining at constant diameter; also check that steering dot is centered

3 - increase range to 6000 yards; check that range rate arc is moving in a counter-clockwise direction

4 - decrease range to 4000 yards; check that range rate arc is moving in a clockwise direction and that range circle has begun to shrink

5 - make a 50° turn to the right; check that indicator trace and steering dot move to the left and that system maintains lock-on

6 - make a 100° turn to the left; check that indicator trace and steering dot move to the right and that system maintains lock-on

7 - place aircraft in a 30° climb; check that dot and artificial horizon move down and that system maintains lock-on

8 - place aircraft in a 30° dive; check that artificial horizon and steering dot move up and that system maintains lock-on

9 - position interceptor so that a 30° tail attack can be made

10 - decrease range to 200 yards; check that pull-out presentation appears on scope without armament doors opening

Note: 2000 yards equals one radar mile.

Note: Reference circle will not shrink in the snake mode.

VIII

MISSILE ATTACK (Lead Collision Course)

Position interceptor so that a 90° lead collision course can be flown toward the target airplane (see figure 1). Set salvo selector switch to MISSILES 3 RAD.

1 - obtain lock-on and check for attack presentation

2 - check indicator trace for correct position relative to target

3 - during the attack run, check that range rate arc indicates correct rate of closure and that indicator trace is moving toward center of scope

4 - check that visual contact exists between target and interceptor during last 3 miles of attack run

5 - check that range circle is decreasing in diameter

6 - check that reference circle shrinks to 1/4 inch diameter during final phase of attack run, ("B" Time)

7 - check stability of steering dot during final attack phase for both pitch and yaw

8 - check for pull-out presentation on scope and observe location of indicator trace near center position

9 - visually observe relationship of target to interceptor at time pull-out signal appears

Note 1: During a normal 90° lead collision run, no maneuvering is required for interceptor to pass clear of target.

Note 2: The target ship will be responsible for avoiding collision with the interceptor.

Note 3: If lock-on is lost in the missile mode, the radar must be recycled by returning ACP switch to snake for 5 seconds.

WARNING: Failure of azimuth angle of target to decrease and of range sweep trace to move toward center of scope during final portion of attack run is an indication that a collision course with the target is being flown.

IX

ROCKET ATTACK (Lead Collision Course)

1 - set ACP switch to 24 rockets

2 - the rocket mode can be flown in the same manner as the missile mode with the following exception:

3 - check that range circle begins to shrink at 5000 yards and reference circle decreases to 1/4 inch in diameter; the reference circle decreases to 1/4 inch at 20 seconds to fire

Note: Other than Note No. 3, the same notes and warnings that applied to missile attack will apply to rocket attack.

X

HIGH ALTITUDE CHECK - 52,500 feet

1 - radar mode switch to 30 mile

2 - check range sweep trace normal - no appearance of arcing on the scope

3 - check that ground return appears at proper position on the scope

4 - make normal rocket run on the target airplane from this altitude

5 - make normal missile run on the target airplane from this altitude

WARNING ALL RADAR FLIGHTS: Turn radar power switch to STANDBY

- a - during accelerated maneuvers
- b - while making an ILAS approach
- c - before landing

XI

CROSS CHECKS IN THE EVENT OF A FAILURE

1 - Loss of Video:

- a - check for presence of main bang

b - depress action switch; check that range gate appears on the range sweep trace

c - check if video appears back on the scope when going from STANDBY to ON, or from WARM to ON

d - check the range sweep trace; note if it brightens momentarily every four seconds

2 - Loss of B Scan:

a - actuate anti-jam button for two seconds, then go to STANDBY and back to ON

b - check AC power

c - check cabin pressurization

d - 1 - turn radar power switch to OFF

2 - turn the right intensity control to mid-position

3 - turn IF gain control full counter clockwise

4 - turn radar power switch to ON

5 - adjust the right intensity control and the IF gain control as listed in After Engine Start Procedure

3 - erratic steering dot - check if the range sweep trace is steady

4 - weak video - check ground map return

5 - no beacon return, go to STANDBY and back to ON

ARMAMENT SYSTEMS

TO ACCOMPLISH THE FOLLOWING CHECKS, IT IS NECESSARY THAT A "T" TO "g" JUMPER BE INSERTED IN THE R/H AFT 5401 PLUG, THE COCKING MECHANISM ON THE 3 FORWARD MISSILE LAUNCHER RAILS BE PULLED BACK AND LOCKED, (THIS WILL NOT BE NECESSARY IF BALLAST IS ON THE RAILS), TO SIMULATE THAT THE MISSILES HAVE NOT LEFT THE LAUNCHERS AT THE FIRE SIGNAL, AND THE ROCKET INTERVALOMETER OUT-PUT PLUG IS TIED BACK AND STOWED.

I

TRIGGER SALVO

Operation to be conducted in a clear area with target ship in formation to check the operation:

1 - ROCKET MODE

a - radar ON

b - arm safe switch to ARM

c - ACP switch to rocket trigger salvo

d - squeeze trigger to first detent - (missile bay doors should open)

e - squeeze trigger all the way - (missile bay doors should close)

f - release trigger

Note: Radar ON is not necessary for any trigger salvo cycling, but this will put it in readiness for later radar cycling. Arm safe switch is to remain in the armed position for the entire armament check except to get out of missile misfire.

2 - MISSILE MODE

a - ACP switch to missile trigger salvo

b - squeeze trigger to the first detent (missile bay doors should open and the aft launchers should extend)

c - squeeze trigger all the way back (aft launchers should retract, the front launchers should extend, retract and the missile bay doors should close)

II

RADAR TIE-IN

1 - ROCKET MODE

a - arm safe switch to ARM

b - reset intervalometer by returning ACP switch to the rocket trigger salvo position for 10 seconds

c - check that the radar is not locked on; (to prevent inadvertent cycling of the armament system); then turn ACP to 24 or 48 rockets

d - lock-on to target and make normal radar pass; (the doors should open at approximately 6.5 seconds before fire-time); the doors should close immediately after FIRE

Note: It is necessary to stop momentarily (5 seconds) at the snake position in any radar operation to clear radar. Radar will cease to scan when switching the ACP switch through either the ROCKET OPT or the missile 3 OPT positions. To return to search you must depress the AUTO SEARCH button on the left stick grip.

To repeat radar rocket run, clear radar by going back to SNAKE momentarily and reset intervalometer.

It is possible when re-cycling the intervalometers from the missile trigger salvo position for the missile intervalometer to reset before the rocket intervalometer. In this event place the arm safe switch to ARM, clear radar by going to "Snake" for 5 seconds, and re-cycle intervalometer.

2 - MISSILE MODE

a - arm safe switch to ARM for minimum of 2 minutes; (time needed for 004 box to time out)

b - ACP switch to SNAKE (5 seconds)

c - ACP Switch to ALL

d - lock-on to target and make a normal missile pass; (the doors open and the aft launchers extend at D time, no more than 3 seconds before E time); at E time the aft launchers will retract and the forward launchers will extend; at this time the misfire light on the ACP panel will illuminate indicating that a misfire has occurred e - depress the push to retract button and hold until the forward launchers retract and the doors close

Note: Releasing button before sequence is complete will stop cycle in an intermediate position. Launchers will fully retract. Doors can be closed by actuating push-to-re-tract button again.

To repeat radar missile runs - re-cycle intervalometers.

It is possible when re-cycling the intervalometers from the rocket trigger salvo position for the rocket intervalometer to reset before the missile intervalometer.

In this event place the arm safe switch to ARM. Clear radar by going to snake for 5 seconds. Re-cycle intervalometer. Clear radar by going back to snake.

THREE POINTS TO REMEMBER

1 - RE-CYCLE RADAR

2 - RE-CYCLE INTERVALOMETER WITH ARM SAFE SWITCH TO ARM

3 - ALWAYS BE IN "SEARCH" BEFORE SELECTING ARMAMENT POSITION

(SEE FINAL PAGE SECTION II FOR EXCEPTIONS APPLICABLE TO TF-102A)

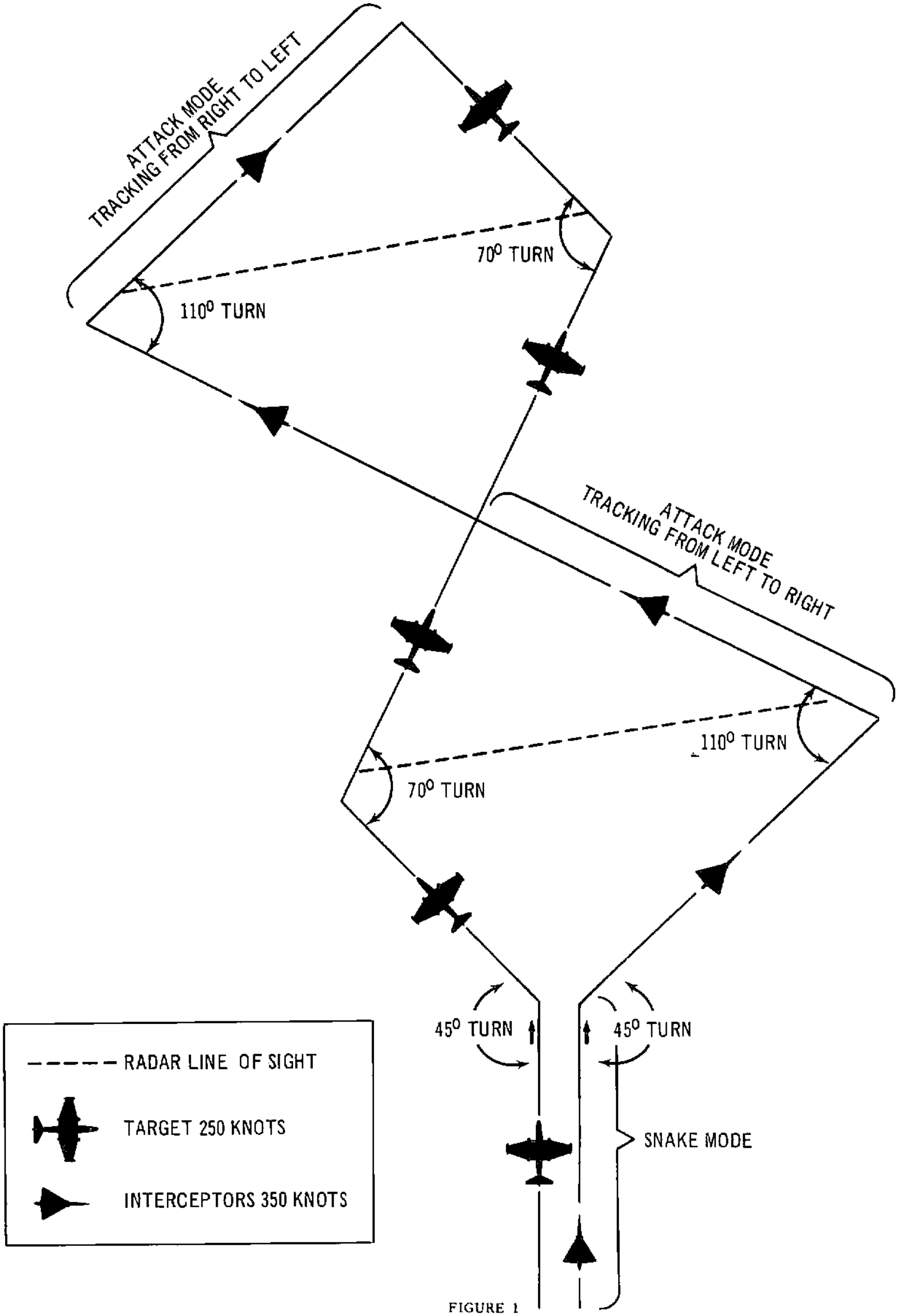

FIGURE 1
F/TF-102A INTERCEPTORS PRODUCTION FLIGHT TEST
PROCEDURE WITH MG-3 RADAR SYSTEM

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F/TF-102A

PRODUCTION FLIGHT TEST PROCEDURES

SECTIONS II & III

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322-1A

JANUARY 1957

CONVAIR
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49

UNCLASSIFIED

F/TF-102A

PRODUCTION FLIGHT TEST PROCEDURES

SECTIONS II & III

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[redacted]

CONVAIR-PALMDALE

17 DECEMBER 1956

PRODUCTION FLIGHT DEPARTMENT

1. THE BASIC F-102A PRODUCTION FLIGHT TEST PROCEDURE DATED 1 NOVEMBER 1955 WAS APPROVED BY AMC LETTER MCPHFC (RDZSFG/19A) WES/eh DATED 28 MAY 1956.

2. THE FOLLOWING PROCEDURE MODERNIZES THE BASIC PROCEDURE IN ELABORATION OF TOLERANCES AND TECHNIQUES GAINED THROUGH ONE YEAR OF EXPERIENCE. IT DOES NOT IN ANY MANNER CHANGE THE INTENT OR OBLIGATION INCURRED BY PARAGRAPH 1 ABOVE.

3. THE PURPOSE OF THIS PROCEDURE IS TO PROVIDE STANDARD GUIDANCE FOR PRODUCTION FLIGHT TESTS IN DETERMINING SPECIFICATION CONFORMANCE OF THE AIRCRAFT AND ITS SYSTEMS. IT DOES NOT SUPPLANT ANY PUBLISHED TECHNICAL ORDERS PERTAINING TO THE F/TF-102A.

R. W. WARDEN, JR.
CHIEF PILOT

FOREWORD

The F/TF-102A Production Flight Test Procedures are divided into three sections:

Section I - Preflight Inspection consisting of all checks, inspections, and operations performed by the Flight Test Engineer prior to the release of the interceptor for flight from Convair San Diego.

Section II - Functional Flight Test including all functional flight checks performed on taxi check, initial flight out of Convair San Diego, and subsequent flights.

Section III - Radar Flight Test scheduled upon completion of the functional flight test in conjunction with a target aircraft. Radar Flight Test includes complete check of the fire control and armament systems. Mechanical discrepancies may be carried forward from Section II but must be checked and cleared on the radar flights.

Generally an aircraft shall have performed satisfactorily during Section I of the Production Flight Test Procedures before being released for Sections II and III. Minor discrepancies may be waived at the discretion of the pilot providing they are cleared before completion of the final section - Radar Flight Test.

These procedures cover Sections II and III only. Section I is published separately.

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SECTION II

FUNCTIONAL FLIGHT TEST

I

LOADING

1 - Release: sign and note aircraft servicing, special limitations, and shortages.

2 - Weight and Balance: check center of gravity (cg) limits 26.5% to 30.5% mean aerodynamic chord (mac).

3 - Engineering Limitations: read and note limitations for this particular aircraft.

Note: in addition to these items the preflight inspection form and engine-run sheet will be reviewed prior to acceptance of a "green" aircraft for ferry flight to Palmdale. The ferry pilot will retain a copy of all five (5) aforementioned forms and turn them in to the Palmdale Flight Dispatch upon arrival.

II

EXTERNAL CHECK

1 - left-hand intake and boundary layer ducts clear

2 - left-hand duct extension secure

3 - left-hand electronic bay door secure

4 - rocket jump angle vane for security and torque paint

5 - left-hand static ports clear

6 - radome condition and security

7 - pitot mast and head secure, pitot cover removed, and static ports clear

8 - right-hand static ports clear

9 - right-hand electronic bay door secure

10 - nose wheel door and well area - no leaks, door seal, taxi light, circuit breakers, battery, dust curtains, microswitch, oleo inflation (should measure 5 inches between scissors axle bolt), tire inflation, tire condition, slippage marks, steering damper, and gear down lock pin removed

11 - right-hand intake and boundary layer ducts clear

12 - right-hand duct extension secure

13 - lower electronic door secure

14 - right-hand missile doors, hinges, and contour

15 - right-hand wing leading edge and wing fences

16 - ram air turbine compartment - primary and secondary fluid level - reservoir caps, primary and secondary accumulator pressure 750 pounds per square inch (psi), leakage, general condition, and security; foreign materials - (ram air turbine (RAT.) kit installed if required); ram air turbine turns freely (clockwise); rat. door will be closed by Inspector after leak check

17 - right-hand main gear well area, tire condition, slippage mark, brake and lines, fairing doors, landing light, gear microswitch under roller, rate gyros warm and turning (external power ON). All circuit breakers IN, strut extension between axle scissors bolt 5-5/8 inch. Static trailing wire grounded

18 - right-hand engine compartment - leaks - fire warning conduit, flight controls

19 - right wing (lower side) fuel panels for leaks, elevon actuator for leaks, and fairing binding

20 - right-hand fillets for elevon rubbing

21 - vertical fin condition and drag chute installation

22 - rudder area for leakage

23 - afterburner nozzles for closure and warpage

24 - afterburner flame holder and general condition of tail pipe

25 - left-hand fillets for elevon rub

26 - left wing (lower side) fuel panels for leaks, elevon actuator for leaks, and fairing binding

27 - left-hand engine compartment - leaks, condition and security of all lines, Shelly unit pin (flush), and throttle linkage

28 - left-hand main gear well area - tire condition, slippage mark, brake and lines, fairing doors, landing light, gear microswitch under roller, pitch gyro warm and turning (external power ON), strut extension between axle scissors bolt 5-5/8 inch static trailing wire grounded, high pressure pneumatic system gage 3000 psi, ±200 psi, combustion starter selector switch for external air start and pre-charge missile bay doors (circuit breakers IN, depress switches and receive amber and green lights)

29 - left wing leading edge and wing fences

30 - left missile doors, hinges, and contour

31 - fuselage external canopy jettison lanyard cover for security

32 - oxygen filler cover on and secure

III

BEFORE ENTERING COCKPIT

1 - wheel chocks in place

2 - bridle installed (ground run only)

3 - area clear (blast area and forward intake areas)

4 - fire truck standing by

5 - canopy hold safety bar installed

IV

ON ENTERING COCKPIT

1 - while on ladder, check entire upper surface of aircraft for damage, loose plates, "Q" intake ports and general condition

2 - mechanical linkage from canopy to arm ejection seat for security and connections

3 - canopy and windshield glass for damage, i.e., scratches, distortion, discoloration, delamination, etc.

4 - initiator pins all removed

5 - arm rests down and locked

6 - right arm rest safety pin installed

7 - canopy jettison handle in detent

V

PRE-START COCKPIT CHECK

1 - external AC and DC power applied

2 - adjust rudder pedals and seat. Check pedals aligned and seat for excessive play (1/4-inch maximum)

3 - fasten shoulder harness and lap belt:

  a - check inertia reel in locked and unlocked position

  b - check force required to open lap belt, should not be less than 5 pounds or more than 15 pounds

4 - personal equipment leads fastened and checked, (oxygen hose, "G" connection if applicable, radio lead, etc.)

VI

COCKPIT LEFT-RIGHT CHECK

1 - left aft circuit breaker panel

2 - spare lamp supply

3 - map case

4 - "G" suit control valve

5 - mask de-fog set as desired

6 - Fuel Control:

  a - tank selector - both tanks to ENGINE, selector knob must be pushed in slightly to turn; also check security of the knob

  b - boost pumps - check one at a time and leave the left forward pump ON to recharge the combustion starter fuel flask for next start

  c - push to test fuel quantity. If totalizer needle moves this indicates system working normal. If button is held the system is grounded and will indicate approximately 200 pounds below the empty mark

  d - check fuel quantity reading against actual fuel on board (total in gallons X fuel specific weight = total pounds on gage reading) tolerance is ±160 pounds and ±4% of indicator reading. Fuel specific weight is 6.5 pounds per gallon @59°F.)

  e - check left and right position readings for each No. 3 tank. (No. 3 tank capacity is 159 gallons of which 157 gallons are usable. To find the correct reading in pounds multiply 157 gallons by fuel specific weight. Tolerance is the sum of ±160 pounds and ±4% of indicator reading.)

7 - ARC-34 - OFF

8 - ram air turbine handle - IN

9 - throttle - OFF

10 - speed brakes switch - NEUTRAL

11 - emergency fuel (turn ON and check for light on master warning panel). Return switch to NORMAL guard down

12 - Oxygen Equipment (MB-2 Regulator):

  a - check pressure (1800 pounds)

  b - check ON - OFF lever for freedom of movement. Set to ON

  c - set 100% or NORMAL

    d - press to test EMERGENCY. Check for pressure flow

    e - check tear drop blinkers

13 - landing gear control handle DOWN. Gear indicators down and locked. Press test button - warning light in handle should illuminate

14 - Landing - Taxi Light Switch:

    a - check taxi light (ground man to check)

    b - check landing light (ground man to check)

    c - switch to neutral - (OFF)

15 - lower left circuit breaker panel (all IN)

16 - emergency gear extension handle IN. Safety clip installed - if applicable

17 - Cabin Pressure Switch:

    a - check installation, should have to pull spring loaded switch out to place in RAM position

    b - turn switch to OFF position

    c - note that cabin altimeter reads approximately field elevation

18 - drag chute handle IN

19 - needle and ball (reads aircraft attitude)

20 - clock set

21 - altimeter set (field elevation)

22 - rate of climb needle (reads zero)

23 - MM-2 attitude indicator. (Check that adjust knob changes reference and turns freely.)

24 - air-speed indicator (reads zero and index knob turns freely)

25 - Omni course setting knob turns freely. (VHF Nav power switch OFF. Flags should be showing.)

26 - Engine Instruments:

a - tachometer - zero - check limit markings

(1) GREEN band - 90%-98%

(2) RED mark - 102%

b - exhaust temperature - zero - check limit markings

(1) GREEN band - 470°-610°

(2) RED mark at 660° and 670°

c - ratiometer. Knob turns freely and index moves in the right direction. Check cruise and takeoff position

d - hydraulic pressure gage: select primary and secondary. Red light should be illuminated, if not, check circuit breaker. Check limit markings

(1) GREEN band - 2500 psi and 3000 psi

(2) RED mark - 1000 psi and 3100 psi

e - fuel flow - zero

27 - check fire warning and overheat circuit, FLASH in over-heat position. STEADY in fire position

28 - radar intensity controls

29 - turn IF gain control DOWN, fully counter-clockwise

30 - anti-clutter switch OUT

31 - storm lights - ON - check lights then turn OFF

Note: with storm lights ON flight instrument and all warning lights will be at full brilliance.

32 - canopy lock handle OUT. (Light illuminated CANOPY UN-LOCK)

33 - Master Warning Panel:

a - check TEST position, all positions should illuminate plus master warning light

Note: there are two light bulbs in each panel, if one is burned out only half the labeling will be readable.

b - check RE-SET position

34 - check electrical panel:

a - AC generator - OFF

b - DC generator - OFF

c - battery - OFF

d - AC bus - Normal (guard down)

e - master - Normal (guard down)

35 - check VHF Nav:

a - power switch OFF

b - selector knobs turn freely

36 - J-4 Compass Control:

a - N on hemisphere selector

b - set latitude dial (34° @ PMD and 33° @ SAN)

c - select DG position and turn compass card by the SET ± control. Turn off heading 10-40°

d - select mag and note that the compass card fast slaves to the approximate aircraft heading

37 - check list (security and legibility)

38 - check Identification Friend or Foe (IFF) control for selection, installation, then to STANDBY

39 - check right circuit panel (all breakers IN)

40 - Armament Panel:

a - arm switch SAFE - (guard down)

b - arm selector SNAKE

41 - Automatic FLIGHT Control System (AFCS) Control Panel:

- a - direct manual (yaw damper/turn coordinator) - OFF
- b - altitude hold - OFF
- c - AFCS - OFF
- d - AILAS - OFF

42 - Radar:

- a - elevation scan - NEUTRAL
- b - azimuth scan - BROAD
- c - power - OFF
- d - 30 mile search

43 - Center Pedestal:

- a - pitch damper - OFF
- b - trim servo - OFF
- c - Nesa - NORMAL (must pull to place OFF)
- d - temperature control - check auto and manual positions
- e - exterior lights - OFF
- f - anti-ice - OFF
- g - de-fog - OFF
- h - rain clear - OFF
- i - pitot heater - OFF
- j - all light rheostats ON for approximately 15 minutes. Check for shorts, etc. (may be turned OFF during flight)
- k - make general quality check of cockpit (splitter screen, paint, glare shield, scope alignment, etc.)

VII

PRE-START CHECK

1 - emergency fuel OFF (warning light OUT)

2 - fuel selector (both tanks to ENGINE)

3 - left and right forward pumps ON

4 - master switch ON

5 - battery switch ON - (to assure start if external DC power fails)

6 - throttle - OFF

7 - signal for external air to be turned ON

Note: One start will be accomplished utilizing the aircraft's high pressure pneumatic system during Convair shakedown.

VIII

START

Depress ignition button and hold. Move throttle outboard. When combustion starter fires, move throttle inboard and up to IDLE in one continuous motion. Continue holding ignition button until starter cuts out at approximately 30%. START may be aborted at any time by releasing ignition button and placing throttle in OFF position.

Check:

1 - idle rpm 55-65% N_2

2 - fuel flow 800 pph minimum

3 - TT_7 340° maximum stabilized IDLE

4 - oil pressure warning light - OUT

5 - hydraulic pressures 3000 ± 150 psi

IX

POST START CHECK

1 - signal for external starter air to be removed

2 - AC generator switch - re-set and ON

3 - DC generator switch - re-set and ON

4 - signal for external electrical power to be removed

Note: Check master warning panel. If AC or DC warning light illuminates, re-set generators.

5 - Check electrical system as follows:

a - place DC generator switch OFF. Check that master warning panel reads DC power failure. (If not, check external power removed). If warning lights remain illuminated, battery is connected. Generator switch back to ON position.

b - place battery switch to OFF. If warning lights remain illuminated the DC generator is operating.

Note: If at any time all DC power is lost the armament system must be recharged.

c - AC generator switch OFF. Master warning panel should read AC POWER FAILURE. (If not, check external power removed).

d - place AC bus switch to EMERGENCY. Check voltmeter selector. Should read 103-140 volts on all phases except 26 volts which should read 26 volts. Note boost pump lights ON. (Non-essential bus).

e - AC generator switch to ON. Then turn AC bus switch to NORMAL. (Check AC warning light OUT). Check all phases, should read 115 volts ±3 on all phases, 26 volts position should read 26 volts.

6 - electronic cooling light, OUT. (If not OUT advance throttle slightly and return to idle, light should go out).

7 - Radar to STANDBY or ON

8 - ARC-34 ON

9 - boost pumps ON individually and note AC voltage. (Aft pumps first and note lights OUT).

10 - Flight Control Checks:

- a - move rudder pedals full throw and check for binding, pedal force, correct rudder travel, and centering
- b - deflect ailerons fully (7°) both directions and check correct travel, centering, and the following forces:
  - (1) break-out force (2 pounds ±1/2 maximum)
  - (2) 5° stop (force is 13 pounds ± 1 1/2)
  - (3) 7° stop (8 pounds increase in force)
- c - deflect elevons fully both directions (up 25°, down 8°) and check correct travel, centering, and the following forces:
  - (1) break-out force (2 pounds ±1/2 maximum)
  - (2) up and down stops (force under static conditions is approximately 40 pounds ±4 up and 12 pounds ±3 down)

    Note: Control break-out forces check. By use of fish scale check the force necessary to start control movement. It will be necessary to station a man at the elevon to signal when movement is started.

11 - Hydraulic Check:

- a - select primary system on the gage and from neutral move the stick full forward, full aft, and back to neutral, in one continuous motion. The pressure should not fall below 1200 psi and hydraulic warning light should not illuminate.
- b - select secondary system and check the same as above, except the pressure should not fall below 1000 psi.
- c - engage trim-servo; operate speed brakes through two complete cycles and check gage for normal fluctuation, trim servo tie in and restrictor action.
- d - during all stick movements the hydraulic pressure should stabilize at 3000 pounds ±150 pounds within two seconds after control movement

is stopped. (Recovery rate is the most important check of the hydraulic systems).

12 - Damper Check:

- a - engage manual mode (yaw damper and trim coordinator)
- b - engage pitch damper
- c - engage trim servo
- d - repeat controls full throw check

  Note: Rapid lateral movements of stick may induce rudder kicks. This condition normal during ground checks only.

- e - press Emergency Direct Manual on right-hand grip. Manual mode, pitch damper, and trim servo should disengage.

13 - Trim Rate Checks:

- a - rudder: authority is 7 1/2° L and R. Rate is 1.5°/seconds or total throw (15°) in 10 ±4 seconds.
- b - aileron: authority is 3° up or down. Rate is .6°/seconds or total throw (6°) in 10 ±4 seconds.
- c - elevator: authority is 2°down and 9° up. Rate is 1.1°/ seconds or total throw (11°) in 10 ±4 seconds.

14 - Take-Off Trim

Check from maximum deflection each way for dead spots, i.e., right aileron, forward elevator and right rudder. Surfaces should center in the following order: aileron, elevator, rudder, and take-off trim light should illuminate. Take-off trim limits are elevator 5° up ±1/2°, rudder 0° ±1/2°, and ailerons equal. Visual reference to these surfaces may vary due to manufacturing tolerances within the elevon and wing tip installation.

15 - Automatic Flight Control System (AFCS) Check:

Note: AFCS may be ground checked and operated in flight with radar in STANDBY or ON.

- a - fast erect radar horizon upon completion of 4 1/2 minute warmup cycle.

b - engage manual mode, pitch damper, and AFCS in this order.

Note: Engage AFCS with momentary interrupt switch depressed.

c - release MI, check AFCS remains ON, controls remain stable, and that lockout valves engage after 1 1/4 second time delay.

d - depress MI and move controls laterally and longitudinally. Control movement should be normal and AFCS will disengage during lateral movement as ailerons reach limit switch at approximately 3° left and right.

e - beep trim laterally left and right. AFCS and pitch damper will disengage as ailerons reach limit switches at approximately 3° left and right. Pitch damper may not disengage due to collapse of lockout valves causing premature AFCS disengagement via lockout switches. Premature disengagement is not considered a discrepancy as long as 3° aileron movement is not exceeded.

f - reengage AFCS and beep trim longitudinally. Trim will be normal through full trim authority and AFCS will remain ON. Stick will continue to move approximately 1.5 seconds after release of beep button due to smoothing circuitry. Trim follow-up may also be checked at this time by turning AFCS switch OFF. If trim follow-up is satisfactory, controls will hold the beeped trim position. A slight stick transient is allowable (0.4° maximum). Do not beep trim beyond normal trim limits.

g - manually over-power AFCS by moving controls laterally and again AFCS and pitch damper should disengage up on reaching aileron limit switches (10-15 pounds stick force). Pitch damper may again not disengage due to reasons explained in step e - above.

Note: Abrupt up elevator should disengage AFCS and abrupt down elevator may or may not disengage AFCS. This is a by-product of low force lockouts and lockout switches. Such disengagement is not necessary for protection of the aircraft since the pitch "G" limiter provides this safety feature.

h - reengage AFCS. Hit EMERGENCY DISCONNECT on stick. This should disengage manual mode, pitch damper, trim servo, and AFCS.

i - re-set take-off trim, check elevons visually.

16 - Nesa Control: NORMAL, note that glass warms and that no excessive distortion, discoloration, delamination, and/or cracks appear.

17 - Anti-Ice: place switch in the MANUAL position at stabilized idle and note rpm drop. Place switch directly to AUTO and note that the rpm recovers. The master warning panel light should be OFF in manual and auto positions. Do not attempt flight if engine rpm cannot be regained with switch in OFF position. While making this check no glycol flow should be noted as this is rendered inoperative by main gear door close switch.

18 - mask de-fog: set at approximately 12 o'clock for pressure suit flight

19 - Radar Pre-flight check. (See Section III).

20 - ARC-34: make radio check on channel 10 and obtain mean ratiometer setting

21 - ratiometer: set ratiometer (T. O. window).

22 - Canopy: signal for the ground crew to remove canopy hold fixture and perform the following:

a - depress canopy hold button and check counter balance. The force required to move the canopy up or down should not exceed 15 pounds.

b - release button with the canopy in an intermediate position and note that the canopy will hold. (If taxi hold position is inoperative, have crew chief check external release button on upper left duct).

c - close canopy and allow canopy to cinch down - the force required to initiate cinch down action should not exceed 15 pounds. Canopy should cinch down in approximately 4 seconds from 2 inches to 6 inches above longeron

d - push lock handle in - the canopy locks should not bind on the rollers, the handle should move freely, and the light should extinguish.

e - pull canopy handle out and note that cinch down pressure is relieved and that canopy will open freely

f - taxi with canopy as desired

X

TAXI CHECK

During taxi electronic cooling will cease above 75% N_2 rpm and a noise level change will be noted.

1 - Nose Wheel Steering:

a - check directional control and centering by depressing Nose Wheel Steering (NWS) button and operating rudders left and right.

Note: Turning of nose wheel by steering button is limited to 50° left or right of center, at which point nose wheel is in free wheeling, and operation of the NWS switch is overridden by braking action. Braking action must be applied to bring the nose back to the 50° range, while at the same time holding the NWS switch depressed, in order to regain power steering.

2 - Brakes:

make functional check of the brakes for pedal force, uneven response, and drag

3 - Nose Wheel Strut:

should not bottom during normal brake application

XI

TAKE-OFF CHECK

1 - Emergency Fuel System:

a - with engine at idle, select fuel control switch to EMERGENCY. The emergency fuel ON light should illuminate, and fuel flow should read 750 to 1050 pph.

b - advance throttle to full military power and check fuel flow. Minimum is 6400 pph at San Diego (sea level) and 6000 pph at Palmdale (elev. 2550 ').

Note: TT_7 below 640°.

c - retard throttle to 80% N_2 rpm and reselect NORMAL fuel system. (Note emergency fuel warning light OUT).

2 - Power Check:

a - advance throttle to military power noting any binding or restriction in throttle movement b - check ratiometer against calculated value. (May take 30 seconds to peak).

Note: If this reading is questionable, stabilize engine at full military for 3 minutes and re-check.

c - check N_2 rpm. (Max. 102%)

d - check that brakes hold in full military power

e - check fuel flow (6000 min.)

f - check exhaust gas temperature. 670° maximum on acceleration and 640° maximum stabilized

XII

TAKE-OFF

1 - engage nose wheel steering button

2 - release brakes and assure NWS is engaged

3 - light afterburner by moving power lever outboard. If exhaust nozzle fails to open when afterburning starts there will be a rapid rise in temperature and loss of approximately 4% in rpm. If evidence of either of these conditions exists terminate afterburning immediately by moving power lever inboard

4 - check fuel flow, TT_7, and N_2 rpm during initial takeoff roll

5 - raise gear when safely airborne and check indicators for UP indication

XIII

CLIMB

1 - continue afterburning climb maintaining airspeed below 300 knots indicated air speed (IAS) on green airplanes; keep close surveillance on all engine instruments and be prepared to select the emergency fuel system if a drop in N_2 rpm and fuel flow is indicated

2 - during climb-out trim aircraft in pitch only, then at 300 knots IAS release stick and note lateral roll-off and/or directional trim shift; if lateral roll-off exceeds 30° per second or directional trim shift exceeds 1/2 ball width record the amount and direction as a flight squawk

3 - shutoff afterburner by moving the throttle inboard; check ratiometer to determine if nozzle closed

4 - relight afterburner by moving the throttle outboard; check time to light (2 seconds maximum) and engine instruments to determine if nozzle opened

5 - retard throttle to aft stop in afterburning range; check continuance of afterburning in power range; (min. 2% N_2 rpm)

XIV

GEAR WARNING LIGHT

Note: The landing gear warning system has 3 switches incorporated to illuminate the warning light under all landing situations in the event the landing gear is not in the down and locked position. One microswitch is located at the military stop on the power quadrant, one meletron switch is actuated by air-speed and one meletron switch is actuated by altitude.

1 - moving the power lever to full military will put out the light, as will increasing the airspeed above 321 knots or altitude above 13,500 feet ±1000 feet, or at any time the gear is down and locked

2 - during descent the light will come on at 10,000 feet ±500 feet if the airspeed is below 250 ±15 knots, providing the throttle is aft of the military stop

XV

COCKPIT PRESSURIZATION (Level Off 15,000 feet)

1 - check cockpit altimeter should read 10,000 feet ±320 feet

2 - check for fumes in cockpit

3 - select OFF. (Pressure should bleed slowly to approximately 12,000 feet) with no in-flow to the cockpit).

4 - select RAM; the dump valve should operate; some ram pressure will be noted depending on airspeed. The cabin altimeter will read approximately 13,500. If suspected that the cabin dump valve did not operate, re-pressurize, select switch straight to RAM and pressure should dump immediately.

5 - re-select PRESS and note that the cabin altimeter returns to 10,000 feet ± 320 feet within 45 seconds

XVI

COCKPIT TEMPERATURE CONTROL

1 - select AUTO-HOT and note temperature change (this may require 3 minutes)

2 - select AUTO-COLD, note that inflow air cools (this may also require 3 minutes)

3 - select MANUAL-HOT, the temperature change should be apparent within approximately 30 seconds

4 - select MANUAL-COLD (cool air should be felt almost immediately)

5 - select comfortable level

Note: The rheostat knob must be held in the detent position for MANUAL HOT and COLD.

XVII

SPEED BRAKES

1 - operate speed brakes through a complete cycle; note that the opening and closing is symmetrical

2 - check pitch response; pitch up will be noted on extension and slight pitch down on retraction if restrictors installed

3 - re-check with trim servo ON; this should compensate for pitch changes

XVIII

LANDING GEAR

1 - cycle gear at 240 knots IAS; gear should extend and lock in 6 ± 1 seconds

2 - observe secondary hydraulic system, aircraft response, and noise level

3 - at least once during shakedown, the gear should be retracted by the EMERGENCY RETRACT BUTTON; depress the button with the gear handle in the down position and the gear should retract in 6 ± 1 second. The red light will remain on in the gear handle; regain control of the normal system by placing the gear handle in the UP position

Note: The EMERGENCY RETRACT BUTTON does not have to be held in for gear retraction when airborne.

XIX

TRIM

1 - check rudder trim, left and right

2 - aileron trim, left and right

3 - elevator trim, nose up and down

4 - actuate takeoff trim-button; there should be no aircraft response

XX

CLIMB TO 26,500 FEET CHECK

1 - time turns: check turn and bank indicator during climb; hold 1 needle-width turn; aircraft should turn 45° in 30 seconds (±5 seconds); check both left and right

2 - check airspeed indicator against Mach indicator; when climbing through 22,500 feet at 300 knots IAS, Mach indicator should read .68 ± . 01.

Note: This is only an instrument check as the Mach meter and the airspeed both obtain information from the same source.

XXI

LEVEL AT 26,500 FEET ALTITUDE CHECK

1 - Cabin Altimeter: should read 10,000 feet (±320 feet) at full differential pressure (5.0 ±0.1 psi)

2 - Electrical Check: check a-c voltages; all phases should read 115 (±3) volts, except 26 volts which should read 26V; check d-c power by testing any warning light.

3 - Emergency Fuel System:

a - at 80% N_2 rpm, switch to the EMERGENCY fuel system; advance throttle to full military; check N_2 rpm, fuel flow and TT_7

b - cycle afterburner, check re-light, aft stop for blowout, and afterburner range (2% N_2 rpm)

c - retard throttle to 80% N_2 rpm, re-select the NORMAL system

XXII

CLIMB TO 35,000 FEET ALTITUDE AND ESTABLISH LEVEL FLIGHT

1 - trim aircraft hands off to 0.8 Mach

2 - engage manual mode; check engage transient. (1/4 ball maximum)

3 - manual mode OFF; check dis-engage transient

4 - push rudder and note natural aircraft damping

5 - manual mode ON; push rudder as above and note damping to be approximately dead beat

6 - engage pitch damper; check for engage transient; (any pitch transient is unacceptable)

7 - pitch damper OFF; check disengage transient

8 - induce pitch oscillation and note natural damping

9 - pitch damper ON; induce pitch oscillation and note damping to be approximately dead beat

XXIII

VMAX LEVEL AT 35,000 FEET

Check the following:

1 - Manual Mode:

a - engage manual mode

b - accelerate to Vmax level; this will normally be approximately 1.16 Mach at standard temperature (-55°C at 35,000 feet). For each degree off standard add or subtract .005 Mach.

c - check that all short period oscillations are damped through entire speed range

Note: Aileron forces should be equal to those of lower speeds. Elevator and rudder forces should increase.

2 - Trim:

a - during acceleration note lateral or directional trim changes; directional trim changes displacing more than 1/2 ball are unacceptable; lateral roll-off that requires aileron re-trimming in such an amount as to reduce level Vmax is unacceptable; if either of these conditions exist repeat check in direct manual mode

Note: In direct manual mode lateral/directional short period oscillations will persist during acceleration or deceleration through 0.9 Mach to 1.1 Mach; a loss of elevon effectivity will be noted at approximately 0.92 Mach with trim servo OFF causing tuck down during acceleration and pitch up during deceleration.

b - if any unacceptable engage transients or trim changes exist as stated above, then prior to landing, trim aircraft hands off level, 300 knots IAS at 10,000 feet altitude; do not retrim for remainder of flight; advise maintenance crews of this set trim condition before leaving aircraft

3 - Trim Servo:

a - check that during acceleration trim servo induces a nose-up trim from 0.8 to 1.3 Mach

b - check that during deceleration trim servo action is nose-down

Note: The normal trim should always be operational and over-ride the trim servo when engaged.

XXIV

AFCS CHECK

Note: As of this writing only heading hold and roll hold modes are operational.

1 - airborne engagement of AFCS is the same as the ground procedure; however, it is recommended that the aircraft be in trimmed straight and level flight with the radar horizon properly erected to wings level ±3° prior to engagement

2 - upon releasing MI switch aircraft should hold attitude corresponding to radar horizon; aircraft response should be stable; a 1 1/4 second delay will be noted after releasing MI before AFCS engages

3 - depress MI and fly to various attitudes, release MI, note 1 1/4 time delay, and aircraft stability in new attitude

Note: Roll hold will not be operational for angles of bank less than 5° or more than 71°. Attitude should be established prior to release of MI. AFCS will not engage at pitch angles in excess of 77°.

4 - a check of heading hold may be made by establishing a desired heading; the aircraft should maintain this heading under stable conditions and within the allowable limits of the J-4 compass; heading hold should be checked in both MAG and DG position of the J-4 compass

5 - speed and altitude changes should not adversely effect AFCS operation; do not exceed 480 knots IAS below 12,000 feet with AFCS engaged

6 - check longitudinal beep trim for correct aircraft response and trim follow-up

7 - check lateral beep trim for correct aircraft response; a .3 second delay will ensue upon release of the trim button within either heading hold or roll hold; a 1 1/4 second delay will ensue when trimming from one mode to another

8 - the recommended procedure for checking the pitch G limiter is to depress MI and enter a wind-up turn at approximately 450 knots IAS; AFCS should disengage at a positive 4.5 G's or a negative 1.5 G; however, due to accelerometer error, and varied pilot technique a tolerance of ±1 G will be acceptable; check AFCS switch OFF and that lockout valves have released

XXV

CLIMB TO 45,000 FEET

1 - check N_2 rpm

2 - check exhaust gas temperature (660° maximum above 35,000 feet, 670° for 2 minutes on acceleration)

3 - check cabin altimeter at 40,000 feet; should read 16,000 feet ±1,000 feet

4 - check partial afterburner power

XXVI

LEVEL AT 45,000 FEET ALTITUDE CHECK

1 - engage manual mode

2 - accelerate to Vmax level; (this will normally be approximately 1.1 Mach at standard temperature (-55° C.)

3 - check N_2 rpm; (due to temperature N_2 rpm will drop approximately 2%)

4 - check exhaust gas temperature; (660° maximum continuous, 670° allowable for 2 minutes on acceleration)

5 - check aircraft trim and flight controls

XXVII

VNE DIVE

1 - trim servo OFF

2 - manual mode engaged and Pitch Damper engaged

3 - obtain Vmax level at 45,000 feet or above

4 - dive to Vne of Mach 1.5 or 655 knots IAS, whichever occurs first

5 - check noise level at Vne

6 - check stability and control

7 - check anti-G suit equipment in dive recovery

8 - reduce power to military, extend speed brakes, and check aircraft response in slowing to subsonic speed

Note: Do not exceed 5.6 G in symmetrical pull-out or 4.0 G in an asymmetrical pull-out. Zero G is limited to 3 seconds.

XXVIII

HIGH ALTITUDE CHECK

Once during shakedown processing the aircraft will be flown in-excess of 50,000 feet for the following checks:

1 - engine performance

2 - afterburner blowout

3 - radar operation

4 - cabin pressurization

Note: The MC-1 partial pressure suit will be worn for all flights above 45,000 feet.

XXIX

ESTABLISH DESCENT

1 - check Nesa effectivity; distortion and discolorization not excessive

2 - check canopy de-fog ON; air should blow out canopy upper center de-fog outlets; cockpit noise level will increase slightly

3 - at 35,000 feet retard throttle rapidly to stabilized idle, then advance rapidly to full military; if compressor stall occurs, use the following procedure:

a - throttle - IDLE

b - reduce to subsonic speed and decrease altitude

c - if stall continues and engine starts to over-temperature, throttle - OFF; make a normal air start

XXX

ROLL

1 - perform aileron rolls left and right; check for aileron centering and normal aircraft response; roll rate is unrestricted above 20,000 feet and restricted 1/2 (3.5°) aileron throw below 20,000 feet in excess of 450 knots IAS

Note: Do not perform maximum deflection aileron rolls between 0.88 Mach and 1.0 Mach without operative stability augmentation. Radar should be in standby position during maneuvers. This condition prevents gyro damage and maintains the antenna in a rigid position.

XXXI

LOW ALTITUDE SPEED RUN

1 - attain 0.95 Mach maximum between 5,000 feet and 10,000 feet above terrain

2 - check stability augmentation performance

3 - check aircraft trim

4 - check flight controls for centering, forces, and response

XXXII

ARC-34 RADIO

1 - while in flight check communications on several channels through the frequency range of the set: (225.0 to 339.9 mc)

2 - check operation for receiver squelch; the squelch should function to keep receivers quiet except during reception of signals above the preset level

3 - check for presence of sidetone in the headsets

XXXIII

IFF CHECK

1 - select Channel 12 (364.2) "Duly" or "Swanee" and check operation on all modes

XXXIV

ARN-12 MARKER BEACON RECEIVER

1 - check operation of the marker beacon receiving set as aircraft is flown at 1,000 feet and 25,000 feet over a fan-type marker of CAA instrument landing markers

2 - check at outer and inner markers during ILS approach

XXXV

ARN-14 OMNI-DIRECTIONAL RANGE SYSTEM

1 - General communication, omni-directional range signals and course indicator should constantly indicate direction of omni-range station from aircraft, as measured from magnetic North. The compass card, controlled

by the remote indicating magnetic compass, should rotate so that magnetic heading of aircraft is indicated against reference marker at top of indicator face. Vertical pointer of the course indicator should indicate lateral positional deviation from the selected omnirange station. Warning flag should prevent false on-course indications by obscuring part of center-pointing vertical needle of course indicator when airborne equipment is not receiving the proper omnirange signal. Complete operational check as follows:

a - turn on VHF NAV system and set VHF frequency selector to 112.2

b - check for station identification

c - the vertical pointer of the course indicator should give "on-course", "fly-left" or "fly-right" indications

d - check volume control and set at desired level

e - establish a known inbound heading to station by varying aircraft heading so that the No. 2 needle on RMI falls under top indice

f - hold above heading and set same heading on OMNI indicator with course indicator knob; the OMNI needle should be centered and the RMI reading should equal ID249 window reading ±2 degrees (no wind condition)

g - establish drift correction, if necessary, and cross-check that OMNI deviation needle reading corresponds to RMI card needle reading ±2°; check that OMNI needle deflection is correct directionally

h - when flying over station check that RMI No. 2 needle swings 180° and that the TO-FROM indication is correct; also, check for right or left deflection on OMNI needle

Note: When crossing over station, the OMNI flag may flicker in and out; this condition is normal.

XXXVI

J-4 COMPASS CHECK

1 - check in Mag and DG positions on a known heading (±2°)

2 - check in Mag and DG positions that when aircraft turns, compass card rotates freely and in proper direction; check in both directions

XXXVII

MAGNETIC (STANDBY) COMPASS

1 - ±7° of J-4 cardinal headings

XXXVIII

LOCALIZER AND GLIDE PATH (ARN-18) OPERATION

1 - these functions will be checked on the Palmdale ILS facilities

2 - trim airplane in level flight at 1,500 feet above terrain and at an IAS of 220-240 knots

Note: Extend landing gear at outer marker.

3 - set VHF frequency selector to 109.1

4 - the leg of the ILS shall be approached from approximately 8 to 14 miles out at an 89° angle or less

5 - when the localizer (vertical) needle on the course indicator starts to move from the stop, make a 35° bank into the localizer beam and assume proper heading; at this time, glide path needle will be riding at top of indicator; when approaching the outer marker the glide path needle should start moving toward center of indicator and be approximately centered at outer marker

6 - check aircraft altitude at outer (approximately 3,900 feet @PMD) and middle (approximately 2,800 feet @PMD) marker; also, check for alignment on runway

7 - with glide path needle centered, maintain the proper course and follow glide path

8 - the landing approach should be discontinued at 200 feet above terrain minimum altitude

XXXIX

EMERGENCY GEAR DROP

1 - this check, if satisfactory, need only be made once during company shakedown; this check to be conducted only if the normal extension and retraction of the landing gear was satisfactory; the following procedure will be used:

a - normal gear handle UP

b - pull emergency gear handle and note that clip, if applicable, holds handle out

c - gear extends and locks within 15 seconds

d - when gear is down and locked place normal handle in the down position

Note: (1) when gear is dropped by emergency, the nose wheel steering is inoperative

(2) should the gear not extend and lock, the following should be used:

(a) normal handle UP

(b) push in emergency gear handle

(c) allow 1 1/2 minutes for the air to bleed off; the gear will not retract immediately but will try to retract 5 or 6 times before they will indicate UP and locked

(d) after gear is UP and locked, lower gear by the normal hydraulic system; if the gear still does not indicate down and locked; pull the emergency gear handle; have a visual check made by the tower or other aircraft; alert emergency crews via tower and land

(e) if after landing the gear still indicates unsafe, stop on the runway and have landing gear pins installed

XL

RAM AIR TURBINE CHECK

1 - select PRIMARY hydraulic gage

2 - extend RAT during landing pattern; (maximum extension 345 knots)

3 - check that under no control movement the gage readings are above the following:

(1) 200 knots - 2450 pounds

(2) 180 knots - 1800 pounds

(3) 160 knots - 1350 pounds

(4) 140 knots - 1000 pounds

XLI

LANDING

1 - if gear is to be extended by emergency system, it is recommended that a simulated flame-out pattern be flown to allow more time for gear drop and RAT check

2 - after touch-down check the following:

  a - deploy drag chute; (maximum deployment time - 4 seconds)

  b - check jettisoning operation when clear of the runway

  c - with speed brake switch IN, re-pull handle noting the secondary hydraulic system gage, the brakes should open; with handle pulled, operate speed brake switch; speed brakes should remain OPEN

XLII

POST LANDING CHECK

1 - Engine: if an engine re-check is required, obtain Palmdale Tower permission to stop on taxi strip for run-up

2 - Operate trigger salvo. There should be no door action due to nose wheel door microswitch which prevents firing when door is in open position

3 - boost pumps OFF

4 - VHF NAV - OFF

5 - radar OFF

6 - pressure - OFF

7 - canopy OPEN, if desired; maximum speed 50 knots

8 - master switch OFF; check that all electrical power is OFF the aircraft; turn back ON and re-set AC and DC generators

9 - Fuel System:

a - low level warning: at least once during company shakedown this system should be checked by running both No. 3 tanks down to low level indication; the warning lights should illuminate at 570 pounds ±160 pounds and ±4% of indicator reading

b - left and right tank operation: select left and right tank position and note normal engine operation and that fuel level in No. 3 tank on selected side drops; it is necessary to have fuel in No. 3 tanks left and right only in order to detect drop in No. 3 tank level

c - flow divider: select No. 3 tanks left and right; maximum differential in readings will be 300 pounds unless afterburner use was above average during the flight

d - fuel quantity: select No. 3 tanks left and right; the total of those two readings should be within 160 pounds and ±4% of indicator total reading

Note: If the fuel quantity indications are questionable, check on actual amount of fuel required for reservicing the aircraft.

XLIII

SHUTDOWN

1 - with external power applied and ground crew positioned, retard throttle OFF, monitor shutdown, check fuel flow zero, normal engine deceleration, and no rise in TT_7

Note: Engine cool-down period is 5 minutes. This time should have elapsed by the time aircraft is in the chocks.

2 - operate speed brakes; this will bleed the secondary hydraulic system below 800 psi prior to the primary system and give a reliable check on the hydraulic warning flasher unit

3 - work controls to work primary system below 800 psi; check hydraulic warning light goes steady at approximately 800 pounds

4 - speed brakes closed; hydraulic pressure zero; rpm zero; extend, rotate, and pull drag chute handle approximately 1/2 inch to check that pneumatic system will open speed brakes; push drag chute handle IN

5 - turn fuel selector OFF

6 - install seat pin

7 - disconnect personal leads and insure lap belt pin is clear

8 - check to see canopy fixture is installed prior to leaving aircraft

9 - make short visual check of aircraft for leaks, pressure readings, etc.

XLIV

POST FLIGHT

1 - Short Post Flight Briefing to Crew Chief for fuel, engine, hydraulics, gear, flight controls, and pneumatic discrepancies; this short briefing is necessary and important for proper ground disposition of the aircraft.

2 - Post Flight Meeting with Foreman and Liaison Engineers for further discussion and elaboration on flight discrepancies.

3 - Complete Convair Form 1860 in triplicate.

4 - No. 2 copy to Inspection; No. 3 copy to Shop; return original copy to Production Flight Office (Flight Dispatch).

5 - Complete aircraft folder.

THE PRODUCTION FLIGHT TEST PROCEDURES AND SPECIFICATIONS FOR THE F-102A APPLY TO THE TF-102A WITH THE FOLLOWING EXCEPTIONS:

LOADING

1 - Weight and Balance: cg limits 26.4% to 27.5% mac.

2 - Note limitations and engineering shortages.

EXTERNAL CHECK

1 - check side slip transducer probe for alignment and security.

POST START CHECK

Canopy: signal for the ground crew to remove canopy hold fixture and perform the following:

a - rotate canopy latch handle counter-clockwise to vertical position and check counterbalance

b - rotate canopy latch handle clockwise to horizontal with canopy in an intermediate position and note canopy will hold

c - rotate canopy latch handle to vertical and close canopy; push in handle and note cinchdown during initial travel

d - push latch handle full in; the canopy locks should not bind, both lock over-center indicators aligned, and canopy unlock light extinguished

e - pull canopy handle out (approximately 8 inches) and note latches disengage and cinch-down is relieved

f - taxi with canopy as desired

Note: Do not taxi with the canopy open unless canopy latch is in lock (horizontal) position.

IN-FLIGHT CHECKS

1 - all switches and controls will be functionally checked from both sides

2 - the emergency landing gear retract button ~~must be held until landing gear indicates up and locked~~ cannot be checked under normal conditions

3 - note canopy buffet between 0.82 Mach and 0.87 Mach. Buffet will increase with higher angles of attack, ~~higher~~ lower altitudes, and accelerated maneuvers

4 - Vmax level at 35,000 feet will be approximately 0.92 Mach

5 - VNE Dive speed is 510 knots IAS or 1.3 Mach, whichever occurs first

6 - altitude specification is 48,500 feet

7 - maximum differential between airspeed indicates at 300 knots IAS is ±5 knots

8 - maximum differential between J-2 Compasses is ±2°

9 - maximum differential for altimeters is ±175 feet at 10,000 feet indicated altitude

SECTION III

RADAR FLIGHT TEST

SPECIFICATION CONFORMANCE

The following outlines MG-3 Radar Test Procedures which are conducted in a manner to afford compliance with Test Specification ZN-8-004 and ZN-8-005 as amended by AMC letter MCPHFC (RDZ-1SFG/2-6) dated 11 August 1955.

I

BEFORE START WITH POWER APPLIED TO AIRCRAFT

1 - armament switch to SAFE

2 - salvo selector switch to SNAKE

3 - elevation control to DETENT

4 - azimuth scan control to B

5 - radar power switch to OFF

6 - radar mode switch

  a - check light in 30 mile and 6 mile position

  b - check for no light in the ground map and beacon positions

  c - return switch to 30 mile position

7 - intensity controls left and right turn to mid position

8 - IF gain control full down (counter clockwise)

9 - radar hand control; check locked position; unlock hand control

10 - elevation roll switch to DETENT

11 - check electronic cooling light ON

Note: Forward instrument panel lights rheostat must be turned clockwise to illuminate the 6 mile and 30 mile mode lights.

II

AFTER ENGINE START

1 - AC power check 115 ± 3 volts

2 - electronic cooling light OFF

3 - radar power switch ON

4 - start clock: no presentation for 30 seconds, then the artificial horizon and the range sweep race will appear; the range sweep trace will sweep

5 - depress and hold the action switch on the hand control; adjust right intensity control counter clockwise until the range sweep trace disappears; then turn the RT intensity control clockwise until the range sweep trace is barely visible; while still holding the action switch, press the return to search button; release the action switch; release the return search button

6 - turn IF gain control full up (clockwise)

7 - check the delay timer: 3 minutes, 15 seconds to 4 minutes, 40 seconds for main bang and ground clutter to appear on the scope

8 - adjust left intensity control until the artificial horizon is at a comfortable level; hold horizon knob IN if necessary to erect the horizon, (± 10 degrees 90 seconds maximum); center the artificial horizon by rotating the horizon centering knob

9 - depress action switch and place range gate marker on main bang; check that attack display appears momentarily then goes to break away signal

Warning: If electronic cooling light does not go out do not turn on radar: abort the flight.
If electronic warning light illuminates: turn radar off and abort the flight.

III

IN FLIGHT

1 - check AC voltage 115 ± 3 volts

2 - rotate artificial horizon centering knob full clockwise and counter clockwise; check that the artificial horizon has approximately 2 inches of travel and that it passes through the center position of the scope; re-center the horizon

3 - set the azimuth scan switch successively to "L" "C" "B" and "R"; check that the range sweep trace paints corresponding areas on the scope

"L" 65°L to 10°R
"C" 35°L to 35°R
"B" 65°L to 65°R
"R" 10°L to 65°R

Reset azimuth scan switch to "B"

4 - vary elevation scan control up and down; ground echoes should appear at the lower antenna angles, and disappear at the higher antenna angles; return el-scan control to detent

5 - 30 mile mode - check that the 30 mile marking is illuminated

a - unlock hand control -- press the action switch; observe that the range gate marker appears on the range sweep trace at 7-1/2 miles

b - move the hand control forward and backward; check that the range gate marker moves from the 15 mile position down to the main bang

c - move the hand control so that the range gatemarker coincides with the altitude mark or the main bang; note that attack display appears on the scope

Warning: Do not attempt this with AFCS engaged.

d - while holding the action switch, move the hand control from side to side, and check that the range sweep trace follows the hand control

e - check the altitude mark appears on the range sweep trace at a position corresponding to aircraft altitude

f - note that characteristic ground or cloud echoes appear on the scope at correct range, azimuth, and elevation

6 - 6 mile mode - check the 6 mile marking is illuminated

- a - press the action switch, move the hand control forward and backward
- b - check that the range gate marker travels the full length of the range sweep trace
- c - while holding the action switch, move the hand control from side to side and check that the range sweep trace follows movement of hand control
- d - return to search; check that the altitude mark appears on the range sweep trace at a position corresponding to aircraft altitude

Note: This must be done below 36,000 feet indicated altitude.

7 - return to 30 mile mode --

While holding the action switch actuate the elevation roller switch on the hand control up and down. Check that the aspect of display varies.

8 - return to search

Note: If electronic cooling light illuminates: abort the flight.

IV

GROUND MAP

1 - turn radar mode switch to ground map; check that the 6 mile and 30 mile markings are not illuminated

2 - move the elevation control switch UP; check that the ground display disappears; move the elevation control switch down, check that the ground display reappears

3 - check that the strobes follow corresponding movement of the hand control

4 - fly a heading toward an identifiable terrain feature

5 - turn the elevation scan control to provide a radiation angle appropriate to an estimated range requirement; check accuracy of screen presentation by visual reference to terrain features

6 - move hand control until the lower strobe is just below the reference terrain feature; press the beacon expand switch and check that the expanded display appears on the scope

7 - antenna stabilization check:

- a - return to search in ground map mode; move the elevation scan control to detent
- b - perform a bank to the right and a bank to the left; place the aircraft in a 15° dive and a 15° climb; while these maneuvers are being performed, check that the orientation of map display on the scope remains constant

V

BEACON OPERATION

1 - set radar mode switch to beacon; check 6 mile and 30 mile light not illuminated

2 - fly a heading toward one of the following stations:

a - March AFB	2-2
b - El Toro	1-2-1
c - Downey	2
d - San Diego	1-2-1-1

3 - set the armament selector switch to 3 RAD or 24 rockets

4 - turn the elevation scan control to provide a radiation angle appropriate to the estimated range requirement; check that the coded beacon response appears on the scope

5 - depress the action switch, move the hand control until the lower strobe is just below the lowest horizontal bar of reply; brighten the reply to best response with the elevation roll switch; press the beacon expand switch and check that the expanded display appears on the scope

VI

ANTI-JAM CHECK

1 - system in auto search and 30 mile mode

2 - press anti-jam button and hold at 2 second intervals; check that targets and indicator trace remain on scope

Note: A slight variation in target definition and intensity may be noticed during this check.

THE INTERCEPTOR AND THE TARGET AIRPLANE WILL RENDEZVOUS OVER PALMDALE AIRPORT AT AN ASSIGNED ALTITUDE. AN IN-TRAIL POSITION WILL BE FLOWN BY THE INTERCEPTOR DURING SNAKE MODE CHECKS. THE FLIGHT PATTERN SHOWN IN FIGURE 1 WILL BE USED FOR SYSTEM CHECKS IN THE ATTACK MODES.

VII

SNAKE MODE OPERATION - Check in the 6 mile mode with armament switch to arm.

Set armament selector switch to SNAKE and arming switch to ARM. Follow and maintain a constant distance from the target airplane.

1 - press action switch on hand control and lock on to target airplane at approximately 5,000 yards; check that attack display appears on scope

2 - check range rate arc at zero position on indicator and range circle remaining at constant diameter; also check that steering dot is centered

3 - increase range to 6000 yards; check that range rate arc is moving in a counter-clockwise direction

4 - decrease range to 4000 yards; check that range rate arc is moving in a clockwise direction and that range circle has begun to shrink

5 - make a 50° turn to the right; check that indicator trace and steering dot move to the left and that system maintains lock-on

6 - make a 100° turn to the left; check that indicator trace and steering dot move to the right and that system maintains lock-on

7 - place aircraft in a 30° climb; check that dot and artificial horizon move down and that system maintains lock-on

8 - place aircraft in a 30° dive; check that artificial horizon and steering dot move up and that system maintains lock-on

9 - position interceptor so that a 30° tail attack can be made

10 - decrease range to 200 yards; check that pull-out presentation appears on scope without armament doors opening

Note: 2000 yards equals one radar mile.

Note: Reference circle will not shrink in the snake mode.

VIII

MISSILE ATTACK (Lead Collision Course)

Position interceptor so that a 90° lead collision course can be flown toward the target airplane (see figure 1). Set salvo selector switch to MISSILES 3 RAD.

1 - obtain lock-on and check for attack presentation

2 - check indicator trace for correct position relative to target

3 - during the attack run, check that range rate arc indicates correct rate of closure and that indicator trace is moving toward center of scope

4 - check that visual contact exists between target and interceptor during last 3 miles of attack run

5 - check that range circle is decreasing in diameter

6 - check that reference circle shrinks to 1/4 inch diameter during final phase of attack run, ("B" Time)

7 - check stability of steering dot during final attack phase for both pitch and yaw

8 - check for pull-out presentation on scope and observe location of indicator trace near center position

9 - visually observe relationship of target to interceptor at time pull-out signal appears

Note 1: During a normal 90° lead collision run, no maneuvering is required for interceptor to pass clear of target.

Note 2: The target ship will be responsible for avoiding collision with the interceptor.

Note 3: If lock-on is lost in the missile mode, the radar must be recycled by returning ACP switch to snake for 5 seconds.

WARNING: Failure of azimuth angle of target to decrease and of range sweep trace to move toward center of scope during final portion of attack run is an indication that a collision course with the target is being flown.

IX

ROCKET ATTACK (Lead Collision Course)

1 - set ACP switch to 24 rockets

2 - the rocket mode can be flown in the same manner as the missile mode with the following exception:

3 - check that range circle begins to shrink at 5000 yards and reference circle decreases to 1/4 inch in diameter; the reference circle decreases to 1/4 inch at 20 seconds to fire

Note: Other than Note No. 3, the same notes and warnings that applied to missile attack will apply to rocket attack.

X

HIGH ALTITUDE CHECK - 52,500 feet

1 - radar mode switch to 30 mile

2 - check range sweep trace normal - no appearance of arcing on the scope

3 - check that ground return appears at proper position on the scope

4 - make normal rocket run on the target airplane from this altitude

5 - make normal missile run on the target airplane from this altitude

WARNING ALL RADAR FLIGHTS: Turn radar power switch to STANDBY

a - during accelerated maneuvers

b - while making an ILAS approach

c - before landing

XI

CROSS CHECKS IN THE EVENT OF A FAILURE

1 - Loss of Video:

a - check for presence of main bang

b - depress action switch; check that range gate appears on the range sweep trace

c - check if video appears back on the scope when going from STANDBY to ON, or from WARM to ON

d - check the range sweep trace; note if it brightens momentarily every four seconds

2 - Loss of B Scan:

a - actuate anti-jam button for two seconds, then go to STANDBY and back to ON

b - check AC power

c - check cabin pressurization

d - 1 - turn radar power switch to OFF

2 - turn the right intensity control to mid-position

3 - turn IF gain control full counter clockwise

4 - turn radar power switch to ON

5 - adjust the right intensity control and the IF gain control as listed in After Engine Start Procedure

3 - erratic steering dot - check if the range sweep trace is steady

4 - weak video - check ground map return

5 - no beacon return, go to STANDBY and back to ON

ARMAMENT SYSTEMS

TO ACCOMPLISH THE FOLLOWING CHECKS, IT IS NECESSARY THAT A "T" TO "g" JUMPER BE INSERTED IN THE R/H AFT 5401 PLUG, THE COCKING MECHANISM ON THE 3 FORWARD MISSILE LAUNCHER RAILS BE PULLED BACK AND LOCKED, (THIS WILL NOT BE NECESSARY IF BALLAST IS ON THE RAILS), TO SIMULATE THAT THE MISSILES HAVE NOT LEFT THE LAUNCHERS AT THE FIRE SIGNAL, AND THE ROCKET INTERVALOMETER OUT-PUT PLUG IS TIED BACK AND STOWED.

I

TRIGGER SALVO

Operation to be conducted in a clear area with target ship in formation to check the operation:

1 - ROCKET MODE

- a - radar ON
- b - arm safe switch to ARM
- c - ACP switch to rocket trigger salvo
- d - squeeze trigger to first detent - (missile bay doors should open)
- e - squeeze trigger all the way - (missile bay doors should close)
- f - release trigger

Note: Radar ON is not necessary for any trigger salvo cycling, but this will put it in readiness for later radar cycling. Arm safe switch is to remain in the armed position for the entire armament check except to get out of missile misfire.

2 - MISSILE MODE

- a - ACP switch to missile trigger salvo
- b - squeeze trigger to the first detent (missile bay doors should open and the aft launchers should extend)
- c - squeeze trigger all the way back (aft launchers should retract, the front launchers should extend, retract and the missile bay doors should close)

II

RADAR TIE-IN

1 - ROCKET MODE

a - arm safe switch to ARM

b - reset intervalometer by returning ACP switch to the rocket trigger salvo position for 10 seconds

c - check that the radar is not locked on; (to prevent inadvertent cycling of the armament system); then turn ACP to 24 or 48 rockets

d - lock-on to target and make normal radar pass; (the doors should open at approximately 6.5 seconds before fire-time); the doors should close immediately after FIRE

Note: It is necessary to stop momentarily (5 seconds) at the snake position in any radar operation to clear radar. Radar will cease to scan when switching the ACP switch through either the ROCKET OPT or the missile 3 OPT positions. To return to search you must depress the AUTO SEARCH button on the left stick grip.

To repeat radar rocket run, clear radar by going back to SNAKE momentarily and reset intervalometer.

It is possible when re-cycling the intervalometers from the missile trigger salvo position for the missile intervalometer to reset before the rocket intervalometer. In this event place the arm safe switch to ARM, clear radar by going to "Snake" for 5 seconds, and re-cycle intervalometer.

2 - MISSILE MODE

a - arm safe switch to ARM for minimum of 2 minutes; (time needed for 004 box to time out)

b - ACP switch to SNAKE (5 seconds)

c - ACP Switch to ALL

d - lock-on to target and make a normal missile pass; (the doors open and the aft launchers extend at D time, no more than 3 seconds before E time); at E time the aft launchers will retract and the forward launchers will extend; at this time the misfire light on the ACP panel will illuminate indicating that a misfire has occurred e - depress the push to retract button and hold until the forward launchers retract and the doors close

Note: Releasing button before sequence is complete will stop cycle in an intermediate position. Launchers will fully retract. Doors can be closed by actuating push-to-re-tract button again.

To repeat radar missile runs - re-cycle intervalometers.

It is possible when re-cycling the intervalometers from the rocket trigger salvo position for the rocket intervalometer to reset before the missile intervalometer.

In this event place the arm safe switch to ARM. Clear radar by going to snake for 5 seconds. Re-cycle intervalometer. Clear radar by going back to snake.

THREE POINTS TO REMEMBER

1 - RE-CYCLE RADAR

2 - RE-CYCLE INTERVALOMETER WITH ARM SAFE SWITCH TO ARM

3 - ALWAYS BE IN "SEARCH" BEFORE SELECTING ARMAMENT POSITION

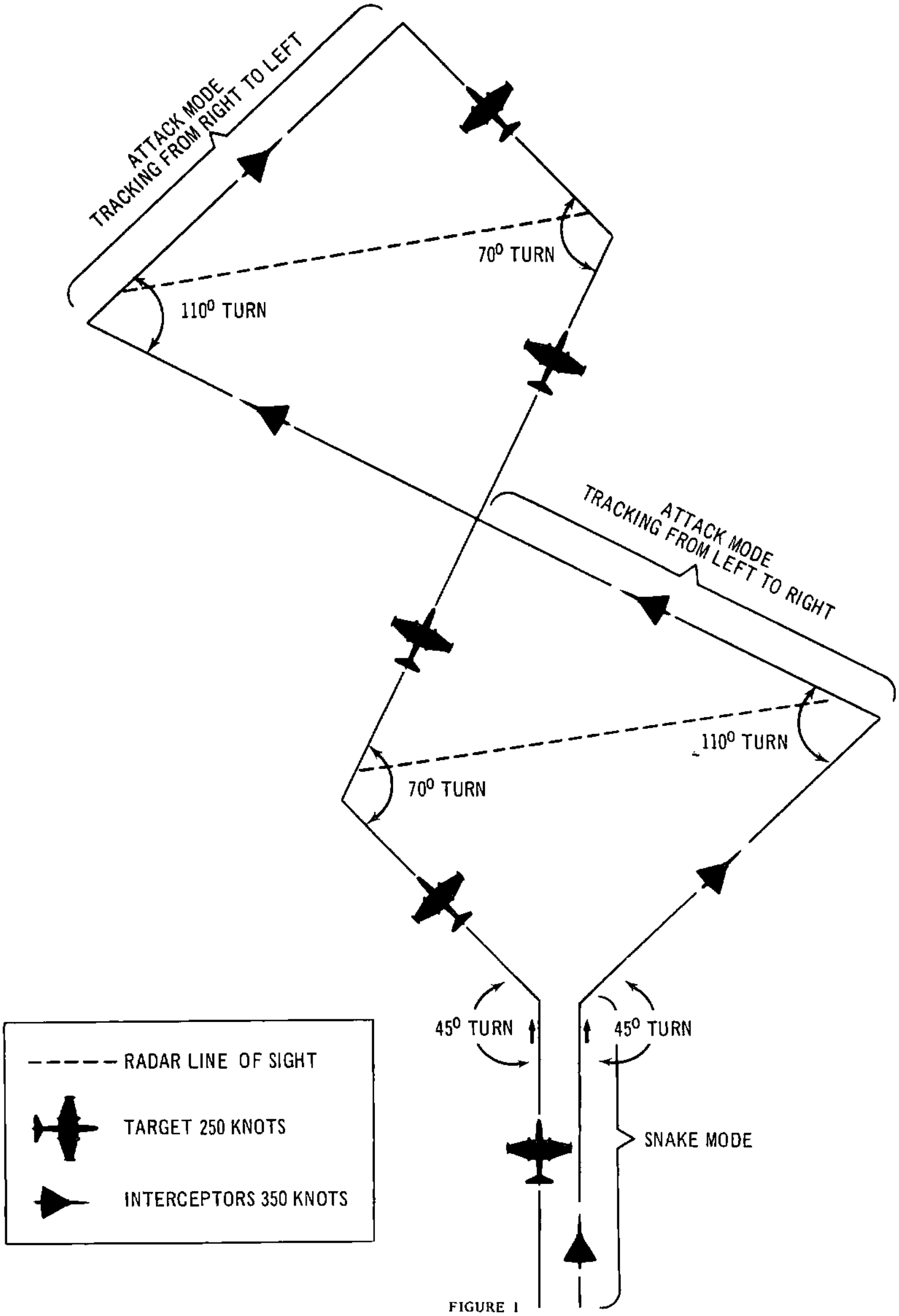


FIGURE 1
F/TF-102A INTERCEPTORS PRODUCTION FLIGHT TEST
PROCEDURE WITH MG-3 RADAR SYSTEM

Addendum I to
AFFTC-TR-56-23
JULY 1957
ASTIA DOCUMENT
No. AD-124105

~~H W Fox~~
~~Carlson~~
Daniel
Applegate
Bull

[redacted]

UNCLASSIFIED

Title Unclassified

F-102A

PHASE IV
PERFORMANCE TEST

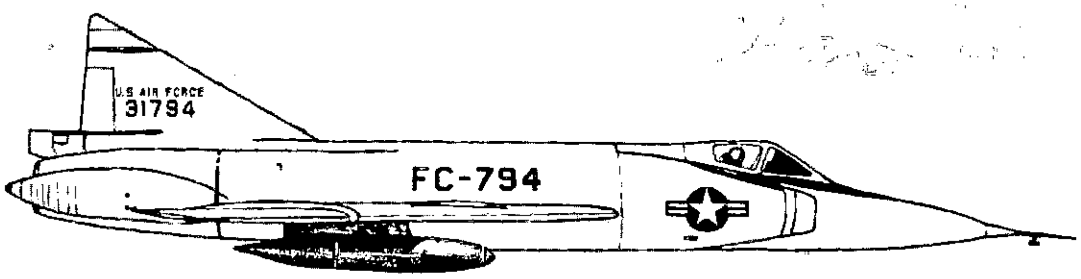
WITH TWO STANDARD CLASS II 230-GALLON PYLON TANKS

57-F-1374

LAWRENCE F. McNAMAR
PROJECT ENGINEER

LOREN W. DAVIS
CAPTAIN, USAF
PROJECT PILOT

UNCLASSIFIED



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EDWARDS AIR FORCE BASE, CALIFORNIA
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE

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115

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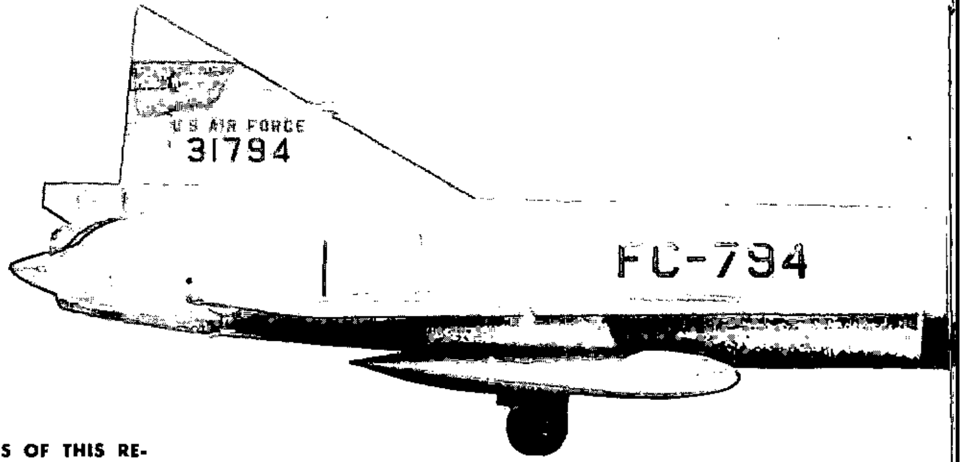

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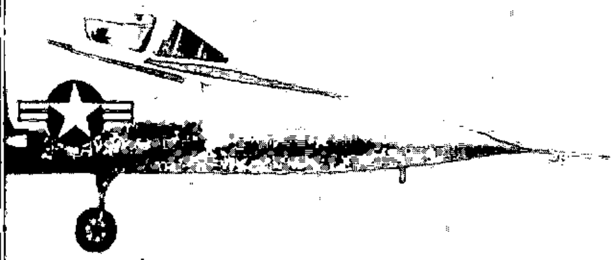
(Title Unclassified)

F-102A PHASE IV PERFORMANCE TEST



WITH TWO STANDARD CLASS II 230-GALLON PYLON TANKS

ADDENDUM I TO AFFTC-TR-56-23 JULY 1957
ASTIA DOCUMENT NO. AD-124105
LAWRENCE F. McNAMAR, Project Engineer
LOREN W. DAVIS, Captain, USAF, Project Pilot

[redacted]

F.M.S. Studios, Pasadena, California, 155 Copies, 28 July 1957

SUMMARY

The test F-102A airplane was equipped with two 230-gallon pylon tanks to extend the range for ferrying purposes. With the tanks installed, the aircraft is limited to 450 knots IAS or .95 true Mach number, and to a load factor of 2.4g with the tanks full. Fuel sequencing of the production installation first uses fuel from the pylon tanks, then the internal tanks. The test airplane was equipped with selector switches mounted in the cockpit to control the sequencing of fuel from the pylon and internal tanks.

A total of 22 hours and 6 minutes were flown at the Air Force Flight Test Center, Edwards Air Force Base, California, to obtain Phase IV quantitive performance data for the Pilot's Handbook and to obtain qualitative handling characteristics of the tank equipped airplane.

Take-off distance from brake release to an altitude of 50 feet is increased 500 feet with maximum power and 1150 feet with military power. Although the tanks increase drag appreciably, the tank limit speed can be attained or exceeded with military power at all altitudes below 35,000 feet. The tanks reduce climb performance, lowering the maximum power combat ceiling from 56,500 to 50,000 feet, and reducing the military power ceiling from 45,000 to 42,000 feet. The additional fuel provided by the tanks increases the ferry range to 1435 nautical air miles, an addition of approximately 250 miles. If the tanks are dropped when empty, the range is increased to 1685 nautical air miles.

Handling characteristics of the production version F-102A with pylon tanks installed are unsatisfactory due to a neutral stability condition which exists during take-off and initial climb. With full internal and external fuel, the center of gravity is slightly aft of the limit. Fuel is used from the external tanks from engine start, maintaining full internal fuel until the tanks are empty. The fuel sloshing to the rear of the partially empty external tanks during take-off acceleration aggravates the aft cg condition, causing the aircraft to approach the neutral stability region. This deficiency was overcome in the test aircraft by using fuel from only the internal tanks during start and taxi, thus keeping the external tanks full. Since the cg shifts forward rapidly when internal fuel is used, the cg is within limits at take-off. In other respects the handling qualities of the tank equipped airplane are generally the same as the unequipped airplane except that a lateral-directional oscillation is encountered at speeds above 440 knots IAS. Although this oscillation is beyond the authority of the damper system, it is not considered a safety of flight problem.

Jettison of the tanks is satisfactory when the landing gear is retracted, but may endanger the landing gear when it is extended. Cycling of the armament doors is prohibited with pylon tanks installed necessitating tank jettison prior to combat, and prohibiting armament training missions with tanks installed.

This report has been reviewed and is approved / 15 JULY 1957

H. A. Hanes
H. A. HANES
Colonel, USAF
Director, Flight Test

Marcus F. Cooper
MARCUS F. COOPER
Brigadier General, USAF
Commander

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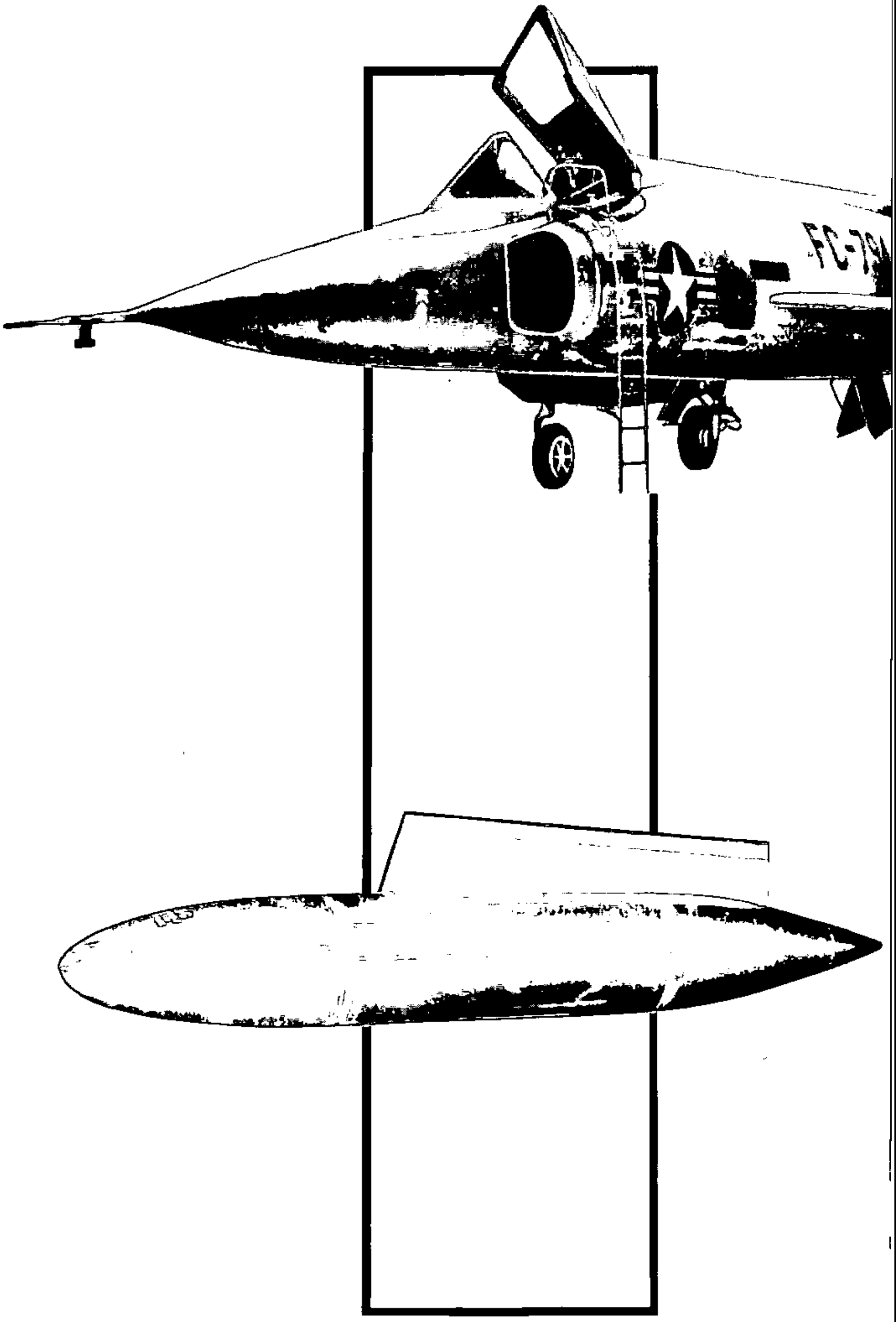
TEST DATA CORRECTED FOR INSTRUMENT ERROR

*NOTE:
Due to the limited requirement for the information contained in this Appendix, it has been published in a separate volume. Copies of this Appendix can be obtained by writing to the Armed Services Technical Information Agency (see inside cover).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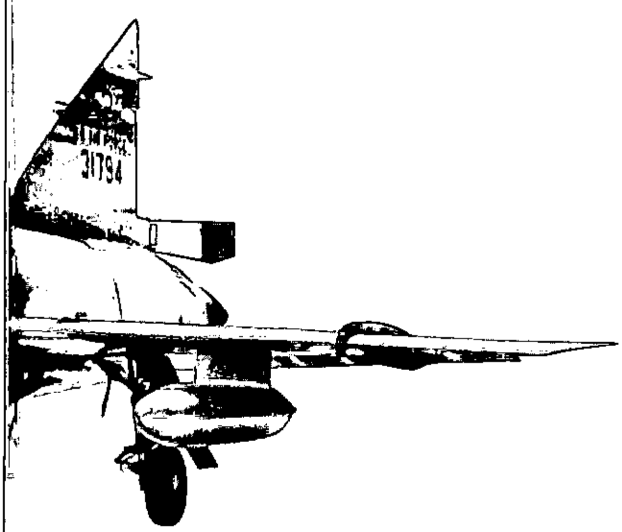


INTRODUCTION

This report presents the results of Phase IV performance flight tests of the F-102A airplane, S/N 53-1791, equipped with jettisonable pylon fuel tanks. These results are compared to those obtained in the previous Phase IV test on the same airplane not equipped with external stores.

The tests were flown at the Air Force Flight Test Center, Edwards Air Force Base, California, during the period from 2 January to 1 February 1957. Fifteen flights were flown; 2 without pylon tanks and 13 with the tanks installed. The total flying time was 22 hours and 6 minutes.

The only significant change over the previous Phase IV test aircraft is the installation of standard Class II, 230-gallon, jettisonable pylon tanks. The basic weight of these pylons and tanks is 289 pounds. When installed on the aircraft and in a level position, the tanks hold 448 gallons of fuel. In level flight attitude there are approximately 3 gallons of unusable fuel remaining in each tank when the "empty tank" signal comes on in the cockpit.

TEST RESULTS

taxiing and ground handling

The taxiing and ground handling qualities of the pylon tank equipped airplane are the same as those of the airplane not equipped with external tanks except for the requirement for more power to initiate the ground roll due to the increase in weight. The nose wheel steering is difficult to use (with or without external tanks) because of high rudder forces and the excessive force required to hold the nose wheel steering button in the engage position. Taxiing is further compromised because of unsatisfactory brakes, which are discussed under a separate heading.

take-off and initial climb

On the test airplane, the use of fuel from the external tanks is controlled by electrically operated valves. A switch for each tank is mounted on the instrument panel, giving the pilot complete control over the use of fuel from either or both tanks. However, there is no provision for such control in the production tank modification plans. Present plans are to have external tanks feed fuel as soon as the engine is started, maintaining full internal fuel until the pylon tanks are empty. This arrangement can cause a serious handling problem due to neutral stability caused by the aft cg location which will exist during take-off and initial climb. With full internal and external fuel, the cg is 0.5 percent MAC aft of the present aft limit of 30.5 percent MAC. The forward shift of the center of gravity as fuel is used from the external tanks is very slow. However, the forward shift of the center of gravity with the consumption of fuel from the number 1 (aft) tank is quite rapid, requiring the consumption of only 300 pounds of fuel to bring the cg within limits. Since this quantity of fuel was normally required for taxi during this test, the cg was held within limits for take-off by use of fuel from the aft tanks only. This procedure can be used only when tank control switches are available to the pilot. In Air Defense Command use, where the aircraft is hangared at the end of the runway, less than 300 pounds of fuel would be required. However, if fuel were used from the aft tank only, take-off handling characteristics would be satisfactory despite a slightly aft cg condition.

30.8

One take-off was made simulating the production configuration by turning the external tank switches on before engine start which allowed fuel to transfer from engine start until the tanks were empty. The internal fuel reading was 7000 pounds before engine start and 7400 pounds immediately prior to take-off, which indicated an internal overfilling condition. This 400 pounds plus approximately 300 pounds required to taxi meant that approximately 350 pounds of fuel had been transferred from each external tank. This partially evacuated the tanks, allowing the fuel to slosh to the rear during the take-off acceleration which aggravated the aft cg condition. The control response characteristics under these conditions showed that the aircraft approached the neutral stability region.

All of the test flights with the exception of the one described above were made using the tank switches to control feeding, burning 300 pounds of internal fuel before take-off and turning the external tanks on after becoming airborne. The handling characteristics were found to be the same as those of the airplane without external tanks so long as the cg was maintained within limits. The increased weight required a 10 knot increase in the recommended take-off speed.

The increased weight and drag of the pylons and tanks plus the weight of the additional fuel has increased the take-off distance approximately 500 feet in maximum power and 1150 feet in military power. Take-off performance is presented in Figure 1, Appendix I and summarized in the following table:

TAKE-OFF PERFORMANCE
Standard Day—Sea Level

	Loading	Weight lbs.	IAS at Take-Off kts.	Ground Roll ft.	IAS at 50 Feet kts.	Total Distance at 50 Feet ft.
Maximum Power (14,900 Pounds Thrust)	No Stores*	27,600	145	2250	160	3000
	Two Full 230-Gallon Tanks	30,600	145	2500	160	3500
	Two Full 230-Gallon Tanks	30,600	165	3100	180	4300
Military Power (10,100 Pounds Thrust)	No Stores*	27,600	150	3900	160	5250
	Two Full 230-Gallon Tanks	30,600	150	4700	160	6400
	Two Full 230-Gallon Tanks	30,600	155	4850	175	7200

* Previous Phase IV data at a standard thrust of 14,800 pounds maximum and 9,800 pounds military power has been corrected to 14,900 pounds and 10,100 pounds maximum and military power respectively. Figure 2 of Appendix I presents the previous data corrected to the higher thrust values.

continuous climb performance

The climb performance of the F-102A airplane is reduced by the installation of pylon tanks. The combat ceiling at maximum power is decreased from about 56,500 feet to 50,000 feet and from 45,000 to 42,000 feet at military power. At maximum power the time to climb to optimum cruise altitude (35,000 feet) is increased from 2.25 minutes to 2.9 minutes and at military power from 7.7 minutes to 10.5 minutes.

During the climb at maximum power at indicated airspeeds above 440 knots, a lateral and directional oscillation is experienced. This oscillation is of a magnitude beyond the authority of the damper system but is not of a severity to be considered a safety of flight problem. The climb attitude and the handling characteristics below 440 knots are approximately the same as the airplane not equipped with pylon tanks.

Climb performance is presented in Figures 3, 4 and 5 of Appendix I, and is compared to the previous Phase IV data in the following table.

CLIMB PERFORMANCE

	Two 230-Gallon Pylon Tanks				No Tanks (Reference AFFTC-TR-56-23 F-102A Phase IV Performance Test)			
	Altitude ft.	Weight lbs.	Rate of Climb ft/min.	Time to Climb min.	Altitude ft.	Weight lbs.	Rate of Climb ft/min.	Time to Climb min.
Maximum Power	Sea level	30,300	13,600	0	Sea level	27,300	17,500	0
	20,000	29,650	12,900	1.5	20,000	26,550	17,300	1.1
	30,000	29,200	11,600	2.3	30,000	26,250	12,300	1.8
	40,000	28,900	6,300	3.4	40,000	25,950	7,300	2.8
	50,000	28,400	800	6.7	50,000	25,550	3,500	4.7
Military Power	Sea level	30,400	5,600	0	Sea level	27,300	5,500	0
	20,000	29,850	3,500	4.5	20,000	26,800	5,100	3.7
	30,000	29,550	2,400	8.2	30,000	26,500	3,300	6.0
	40,000	29,200	700	14.6	40,000	26,200	1,600	10.0
Normal Rated Power	Sea level	30,400	4,500	0	—	—	—	—
	10,000	30,100	3,600	2.6	—	—	—	—
	20,000	29,800	2,600	5.9	—	—	—	—
	30,000	29,500	1,600	12.8	—	—	—	—
	35,000	29,300	1,200	14.3	—	—	—	—

At maximum power the optimum climb speed is limited by the contractor's restrictions on the pylon tank equipped airplane (450 knots IAS or .95 true Mach number). Below 15,000 feet the schedule is limited to about 440 knots because of the previously mentioned oscillations.

At military power the climb schedule should be flown about 0.05 Mach number slower than the optimum climb schedule for the unequipped airplane. Optimum climb speed schedules for the tank equipped airplane determined from level flight accelerations are presented in Figures 6 through 10 of Appendix I.

level flight performance

Installation of the pylon tanks increases the minimum drag coefficient to 0.0132 from 0.0122. The drag rise (increase in drag due to Mach number effects) occurs at a lower Mach number as well, yielding a larger drag differential at high Mach numbers. Maximum level flight speeds, however, are dictated by the pylon tank limitation (450 knots IAS or .95 true Mach number) rather than power limitations. The tank limit speed can be attained or exceeded with military power at all altitudes below 35,000 feet.

Best cruising speed is not affected by the addition of tanks; however, the increase in weight and drag results in a reduction of cruise altitude from 40,000 to 38,000 feet at the start of a cruise climb, and the specific range at the optimum conditions has been lowered approximately 19 percent (to 0.195 from 0.24 nautical air miles per pound of fuel). A more complete discussion of range capabilities is included in the paragraph on "Range Missions". Cruise data is presented in Figure 13 of Appendix I and summarized in the following table:

CRUISE PERFORMANCE

Two 230-Gallon Pylon Tanks Installed

Altitude ft.	Weight lbs.	Mach No.	Specific Range NAMPP	Engine Pressure Ratio P_{t_7}/P_{t_0}
12,000	27,800	.60	.12	1.60
28,000	26,600	.725	.17	1.94
33,000	26,800	.80	.184	2.12
38,000	26,200	.85	.195	2.32

Cruise capabilities without external stores have been checked at 15,000 and 30,000 feet to resolve difficulties encountered during the previous Phase IV program. This data shows an increase in specific range and a decrease in fuel flow required from that of the earlier tests. Figure 12 of Appendix I presents the cruise data, and a discussion of the problem is included in Appendix I.

Complete level flight performance for both the tank and no store loadings are presented in Figures 12 through 25 of Appendix I.

range mission

The ferry range capability of the F-102A (equipped with pylon tanks) is increased approximately 250 nautical air miles to 1435 miles when tanks are retained throughout the flight and using military power for take-off and climb-out, a cruise climb starting at 35,000 feet, and arriving over the destination with 1,000 pounds of fuel remaining. This range can be increased 250 additional nautical air miles by jettisoning the tanks when they are empty. This mission does not include combat. An actual mission of this type was flown to demonstrate the range capability. The mission resulted in 1430 nautical miles traveled and 2,000 pounds of reserve fuel at 35,000 feet altitude. The additional 1,000 pounds of fuel reserve penalized the actual mission approximately 220 nautical miles. The specific range encountered during the mission checked within 2 percent of that obtained from level-flight cruise data. Results of the actual range mission are presented in Figure 26 of Appendix I and summarized in the following table. A computed mission is included for comparison.

ACTUAL RANGE MISSION

2100 Pounds Fuel Remaining at Beginning of Descent—
Pylon Tanks Dropped When Empty

	Take-Off	Military Power Climb	Cruise With Tanks	Cruise Without Tanks	Totals
Time — min.	1	11	35	130	177
Fuel used — lbs.	500	1000	1,420	4,940	7860
Mach No.	—	—	.8	.84	—
NAMT	—	80	280	1,073	1433
NAMPP	—	—	.197	.217	—
Altitude — ft.	—	—	35,000	35,000 to 37,000	—

COMPUTED RANGE MISSION

1000 Pounds Fuel Remaining at Beginning of Descent—
Pylon Tanks Dropped When Empty

Time — min.	1	10.5	32	165*	209
Fuel used — lbs.	500	1000	1,425	6,045*	8970
Mach No.	—	—	.825	.84*	—
NAMT — miles	—	77	278	1,330*	1685
NAMPP	—	—	.195	.22*	—
Altitude — ft.	—	—	35,000	35,000 to 37,000*	—

*Reference AFFTC-TR-56-23
—Phase IV Performance
Test F-102A.

maneuvering capabilities

Maneuvering with pylon tanks installed is restricted because of the tank limitations of 2.4g when full and 4.25g when empty. The tanks do not alter the maneuvering capability of the aircraft except for the load factor limit. Conditions at onset of buffet yield approximately the same lift coefficients as noted in previous tests without external stores. In these tests, buffet was first noted at the conditions tabluated below and presented in Figure 27 of Appendix I.

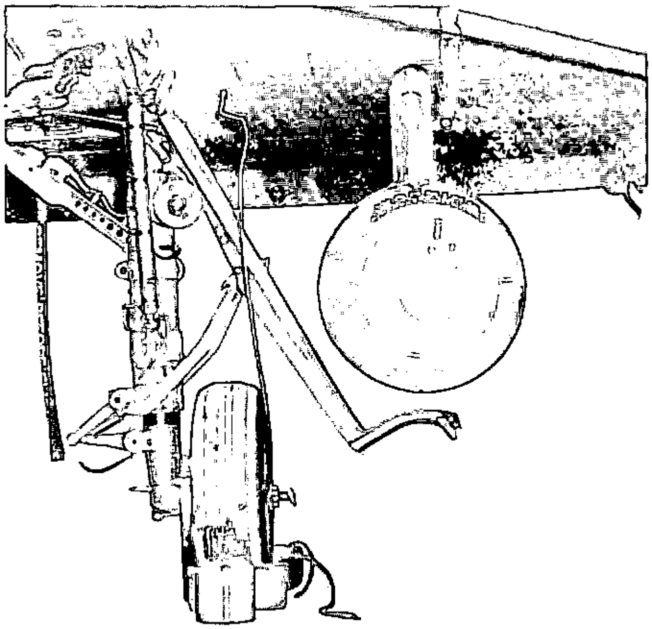
ONSET OF BUFFET

(Pylon Tanks Installed)

10,000 Feet, IAS—kts.	26,900 Pounds Load Factor	30,000 Feet, IAS—kts.	23,700 Pounds Load Factor
172	1	158	1
200	1.5	198	1.5
230	2.0	256	2.0

At the time the test program was flown the aircraft was restricted to 2g load limit. Due to this fact buffet boundary was not investigated at load factors above those listed.

The intensity of the buffet increases with an increase in load factor and/or with a decrease in airspeed. High rates of sink develop at these conditions; at 139 knots IAS a rate of sink of 1030 feet per minute was encountered with an airspeed bleed off of 8 knots per second. However, this rate of sink would be much higher if the airspeed bleed off were accomplished at ½ knot per second. Control of the aircraft is good in roll, pitch and yaw down to the minimum permissible speed of 120 knots IAS. Lower indicated speeds are considered unstable, however, because of the heavy buffet and high rates of sink encountered.



tank jettison

The contractor's tank jettison demonstration indicated that tanks can be jettisoned satisfactorily with the landing gear retracted. No jettisons were attempted with the gear down because of the possible danger of damaging the landing gear door fairings and the tires. A requirement exists for investigating the emergency jettisoning of the external tanks with the gear extended, and should be demonstrated by the contractor prior to tactical use of the tanks.

During the limited Phase IV program the tanks were jettisoned empty once near the cruising conditions of 0.8 Mach number and 35,000 feet. The tanks and pylons cleared the aircraft satisfactorily even though one of the jettisoning charges did not fire.

armament door operation

Cycling of armament doors is prohibited with pylon tanks installed. The specification for development of the pylon tank capability spelled out a requirement for ferry flights only without missile firing capability. The possibility does exist that the doors may be cycled, and missiles fired with tanks installed without undesirable characteristics. This capability has not been demonstrated by the contractor because of lack of requirement. Application of the F-102A would be extended if combat and training missions could be accomplished with tanks installed. The contractor should demonstrate the cycling of the armament doors while tanks are installed to determine if there are any adverse aerodynamic effects present in this configuration.

approach and landing

Installation of the pylon tanks does not affect the F-102A in landing except for a requirement for higher speeds when landing with additional fuel aboard. When less than 2000 pounds of fuel remain, a final approach should be flown at 160 knots IAS with a touchdown at 140 knots. With fuel in excess of 2000 pounds, the final should be accomplished with approximately 170 knots IAS and the touchdown made at 150 knots.

The landing performance results are presented in the following table. The values shown are an average of several landings. The maximum performance landing data is inconclusive, because of poor braking action and a resultant blown tire. All landings were made without the use of a drag chute.

LANDING PERFORMANCE

Standard Day—No-Wind Conditions

	Average Test Weight—lbs.	Average IAS at 50 Feet—kts.	Average Total Distance—ft.	Average IAS at Touchdown kts.	Average Ground Distance ft.
Normal Braking — No Drag Chute	23,000	160	9690	142	6755
	26,000	174	9880	148	7500
Maximum Performance — No Drag Chute	25,600	155	6060	135	4050

braking

The braking system in the F-102A is extremely unsatisfactory. The most serious deficiency is the time delay between deflection of the pedal and the application of braking action. This time delay is approximately 1½ seconds and is present both on application and release of the brakes. Also, the brake pedal deflection is excessive, there is very little "feel" to the system, and the pedal forces do not increase linearly with the degree of deflection.

During landing roll or an aborted take-off the combination of all of these deficiencies make it very easy to over-control the braking action and induce skidding. Once skidding has started the time delay does not allow the brake to release soon enough to prevent a blown tire.

asymmetric loading

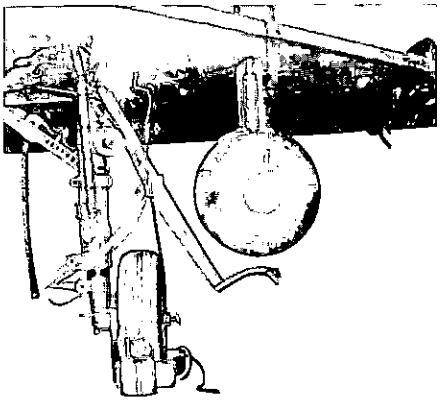
The airplane is easily controlled with an asymmetric load, such as when one pylon tank fails to feed. Differential elevon required to trim out this unbalanced load is in the region of 1 degree (0.5 degrees each elevon) in the cruise configuration. One take-off and one landing were made with one tank full and one empty using normal trim settings. The unbalanced load was easily controlled with a slight amount of lateral control deflection.

thrust calibration

Three static thrust calibrations were made, two at the beginning and one at the end of the program. The nonafterburning calibration agrees very closely with the previous Phase IV test while the calibration in afterburning indicates a lower thrust for the same pressure ratio. Static trim of the engine during these tests was 10,100 pounds in military power and 14,900 pounds in maximum power for a sea level standard day. The military power thrust was approximately 300 pounds greater than during the previous Phase IV test. The inflight bias curve (Figure 11 of Appendix I) agrees very well with that in the previous test (reference AFFTC-TR-56-23, Figure 26 of Appendix I).

airspeed calibration

The airspeed calibrations accomplished during this test substantiate the previous Phase IV calibrations and indicate no effect from the pylons and tanks. The airspeed system used the same test boom as was used previously in the Phase IV program. A comparison of these results is presented in Figure 31 of Appendix I.



CONCLUSIONS

Addition of two 230-gallon pylon tanks increases the ferry range of the F-102A to 1435 nautical miles, an addition of approximately 250 nautical miles. This range is further extended to 1685 nautical miles by dropping the tanks when empty. The increase in weight and drag resulting from the tank installation causes an increase in take-off requirements of 500 feet with maximum power and 1150 feet with military power. Though the parasite drag coefficient has been increased 8 percent and the drag rise is initiated at a lower speed, the limitations on the tanks (450 knots IAS or .95 true Mach number) dictate the maximum speed capability of the aircraft.

The F-102A aircraft with two 230-gallon tanks installed is unsatisfactory in the production configuration because of the aft cg and neutral stability conditions caused prior to and during take-off by uncontrolled fuel scheduling. This situation could be corrected by modifying the system to include selector switches to permit take-offs and initial climbs on internal fuel. The installation of pylon tanks lowers the climb performance by reducing the maximum power combat ceiling from 56,000 to 50,000 feet; and the military power ceiling from 45,000 to 42,000 feet.

Handling characteristics are not changed by installation of the tanks except for the unsafe condition at take-off caused by the rearward center of gravity position at take-off and a lateral-directional oscillation at indicated speeds above 440 knots.

Jettison of pylon tanks is satisfactory with the landing gear retracted, but may endanger the landing gear when extended.

RECOMMENDATIONS

It is recommended that the following be accomplished:

[illegible] → 1. Incorporate pylon tank fuel selector switches in F-102A aircraft prior to installing pylon tanks.

WARD MIT. → 2. Require full pylon tank fuel through take-off, using internal fuel only until the climb speed schedule is obtained (whenever pylon tanks are used).

[illegible] } 3. The contractor demonstrate jettison capabilities with the landing gear extended.

4. The contractor investigate the effects of armament door cycling with tanks installed.

5. Improve the braking system to give the pilot better "feel" and a rapid response to pedal action with reduced pedal deflection for maximum braking.

6. Improve the nose wheel steering by reducing rudder pedal forces required.

7. Reduce the force required to hold the nose wheel steering button engaged.

APPENDIX I

data analysis methods

thrust calibration

The thrust calibration data was corrected to standard sea level conditions as outlined in AF Technical Report No. 6273, pages 3-22 and 3-23, using equations 3.811, 3.812 and 3.816.

The engine manufacturer's C_g versus P_{t7}/P_{o} curve was used as a basis for extrapolating measured thrust data obtained on the thrust stand. Engine air flow for ram drag computations was obtained from Pratt and Whitney's curve No. 17145. A ram efficiency of .96 was assumed for this airframe in the subsonic region.

take-offs

Take-off data was corrected to standard day, no wind conditions as outlined in AFFTC Technical Note R-12 with the use of equations (4-3a) and (4-5a) on pages 7 and 8 respectively. The standard day thrust of 10,100 pounds at military power and 14,900 pounds at maximum power was obtained from the static thrust calibration (Figure 29) and the take-off engine trim curves.

Because the brakes will not hold the airplane in maximum power prior to take-off, the engine pressure ratio was read at military power just before brake release. Using this test P_{t7}/P_{t0} the test thrust was obtained from the thrust calibration curve (Figure 29) of this Appendix.

The take-off data of the previous Phase IV test was corrected to the standard thrust used in this program to aid in the comparison of the two tests.

continuous climbs

Continuous climbs were corrected to standard conditions at test gross weight as outlined in AF Technical Report No. 6273, pages 5-1 through 5-17 and using equations (5.203) and (5.206).

An engine pressure ratio bias curve (P_{t7}/P_{t0} versus t_{IC}) was used to determine power available with military or maximum power on a standard day. Standard thrust was determined by entering the bais curve with standard day temperature to obtain standard day P_{t7}/P_{t0}. Thus using test and standard day values of P_{t7}/P_{t0}, test and standard thrust is obtained as discussed in the paragraph on thrust calibration.

The in-flight bias curve Figure 11 agrees very well with the bias curve used for climbs in the previous Phase IV test. (See Figure 26, AFFTC-TR-56-23.)

■ accelerations

Level flight accelerations, used to determine a climb speed schedule, were reduced by the following method:

$$R/C = \frac{19362\ \theta as\ \Delta(M^2) + \Delta Hc}{\Delta t}$$

where the following notations apply

R/C	rate of climb
θas	standard ambient temperature divided by 288
$\Delta (M^2)$	difference in the square of Mach numbers at time 1 and time 2.
Δ Hc	pressure altitude increment
Δ t	time increment

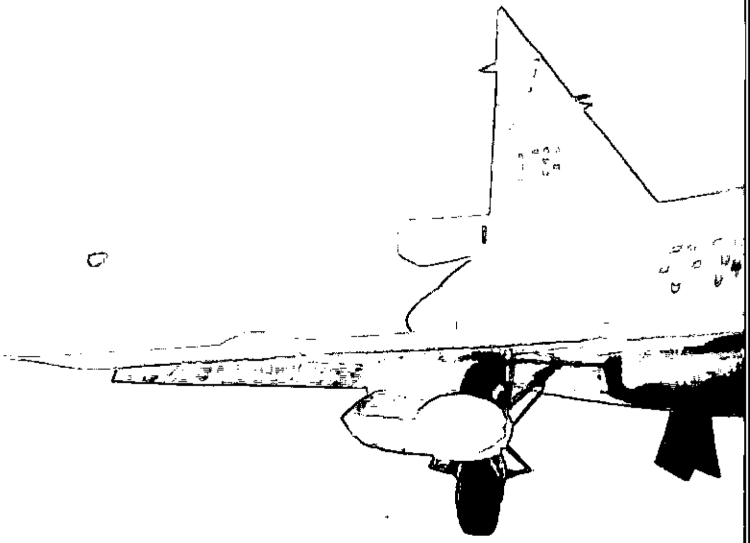
The power available is determined from the bias curve as described in the paragraph on thrust calibration.

■ level flight

The fuel flow was corrected to standard day using inlet conditions and is presented in Figure 28 of this Appendix. The thrust was obtained as described in the paragraph on thrust calibration and weight corrections were made using the drag polars.

Disagreement exists in the P_{t7}/P_{t0} versus Mach number and fuel flow data between the data obtained without external stores and that obtained in the previous Phase IV test. This disagreement arose from misinterpretation resulting from insufficient data at 30,000 feet and at the low speed end at 15,000 feet. The present data appears to be consistent and to agree with the drag polar.

There is a slight difference between the faired fuel flow curves from the two test programs (Figure 28 of Appendix I of this report and Figure 27 of Appendix I, AFFTC-TR-56-23). However, data from the earlier tests is more scattered than that from the present series and the test data is consistent with the faired curve in the present report. The scatter of the previous test results from the state of the art in fuel flow measurement. (Electrical noise was registered with the signal from the fuel flow transmitter causing the errors in the data.) The signal from the fuel flow transmitter in the later tests was recorded on an oscillograph where discrimination could be used in separating noise from the data.

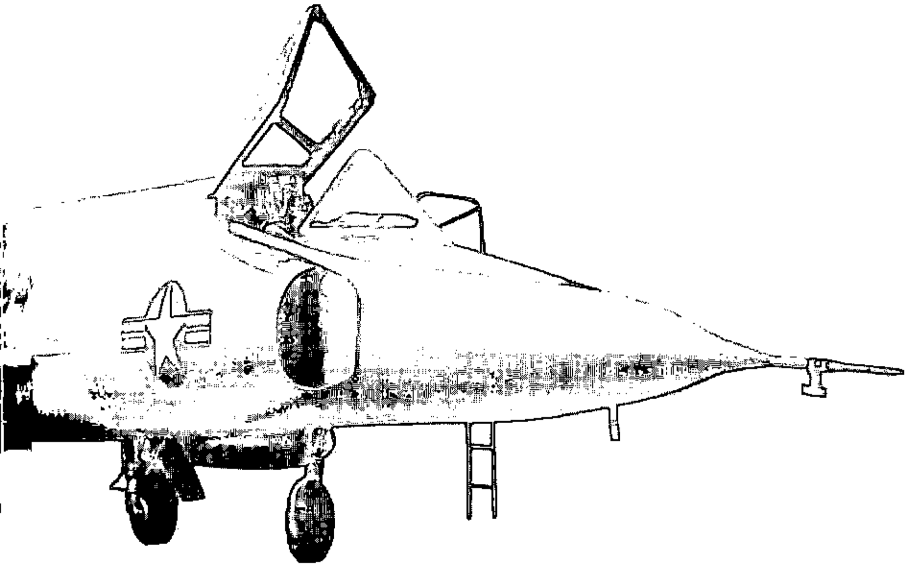


■ landings

The landing data was corrected to standard conditions as outlined in AF Technical Report No. 6273 pages 6-17 through 6-19 using equations (6.407) and (6.408).

■ temperature

Temperature used in the majority of the computations was obtained from weather balloon data taken at approximately the same time as the test flights. When test temperature data was used from the instrumentation in the airplane, a recovery factor of 0.96 was used.



performance plots

Figure No.

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3 Climb Performance 14
11 Pressure Ratio Bias Curve 25
12 Level Flight Performance 26
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27 Buffet Boundaries 42
28 Fuel Consumption 43
29 Static Thrust 44
31 Airspeed Calibration 46

FIGURE NO. 1

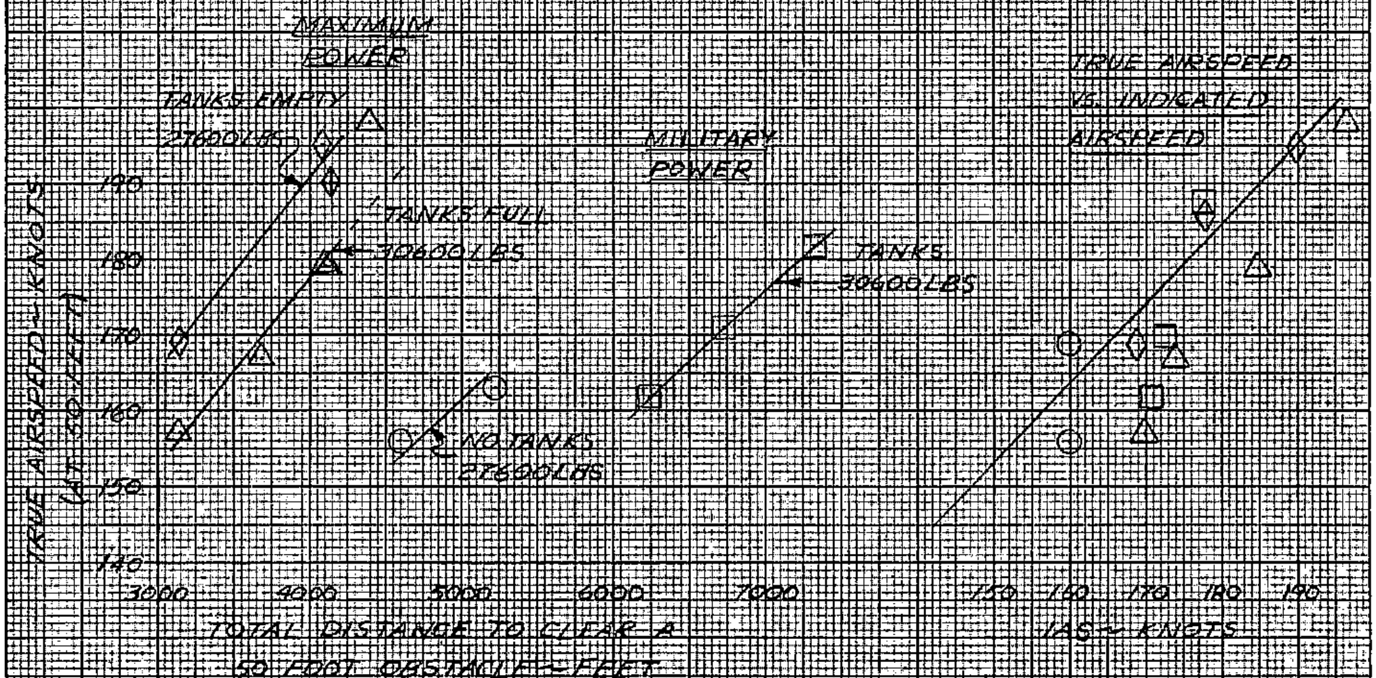
TAKE-OFF PERFORMANCE

F-102A USAF S/N 53-1791

J-57-P-23 ENGINE S/N 604713

FLAP NOZZLE AFTERBURNER S/N 6170039

SYMBOL	POWER	LOADING	WEIGHT	STD. THRUST
○	MILITARY	NO TANKS	27600	10100 LBS
□	MILITARY	FULL TANKS	30600	10100 LBS
△	MAXIMUM	FULL TANKS	30600	14900 LBS
◇	MAXIMUM	EMPTY TANKS	27600	14900 LBS
◈	MAXIMUM	ONE FULL TANK	27600	14900 LBS



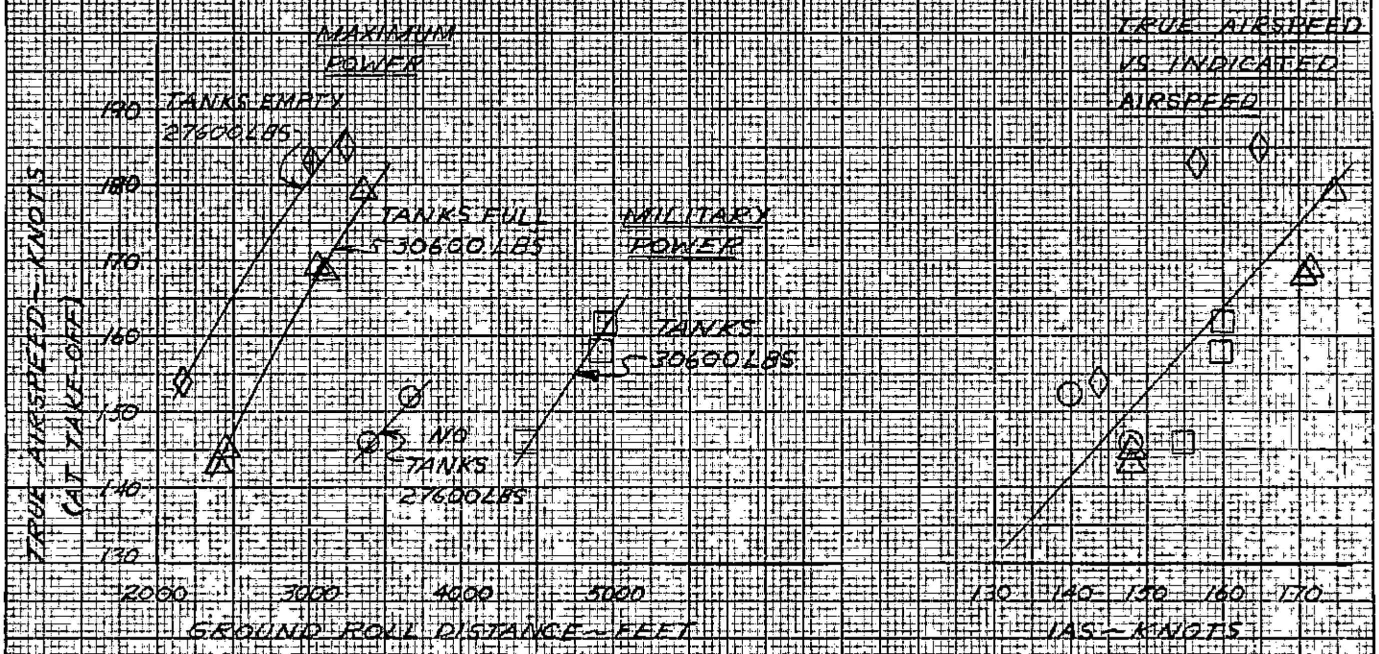
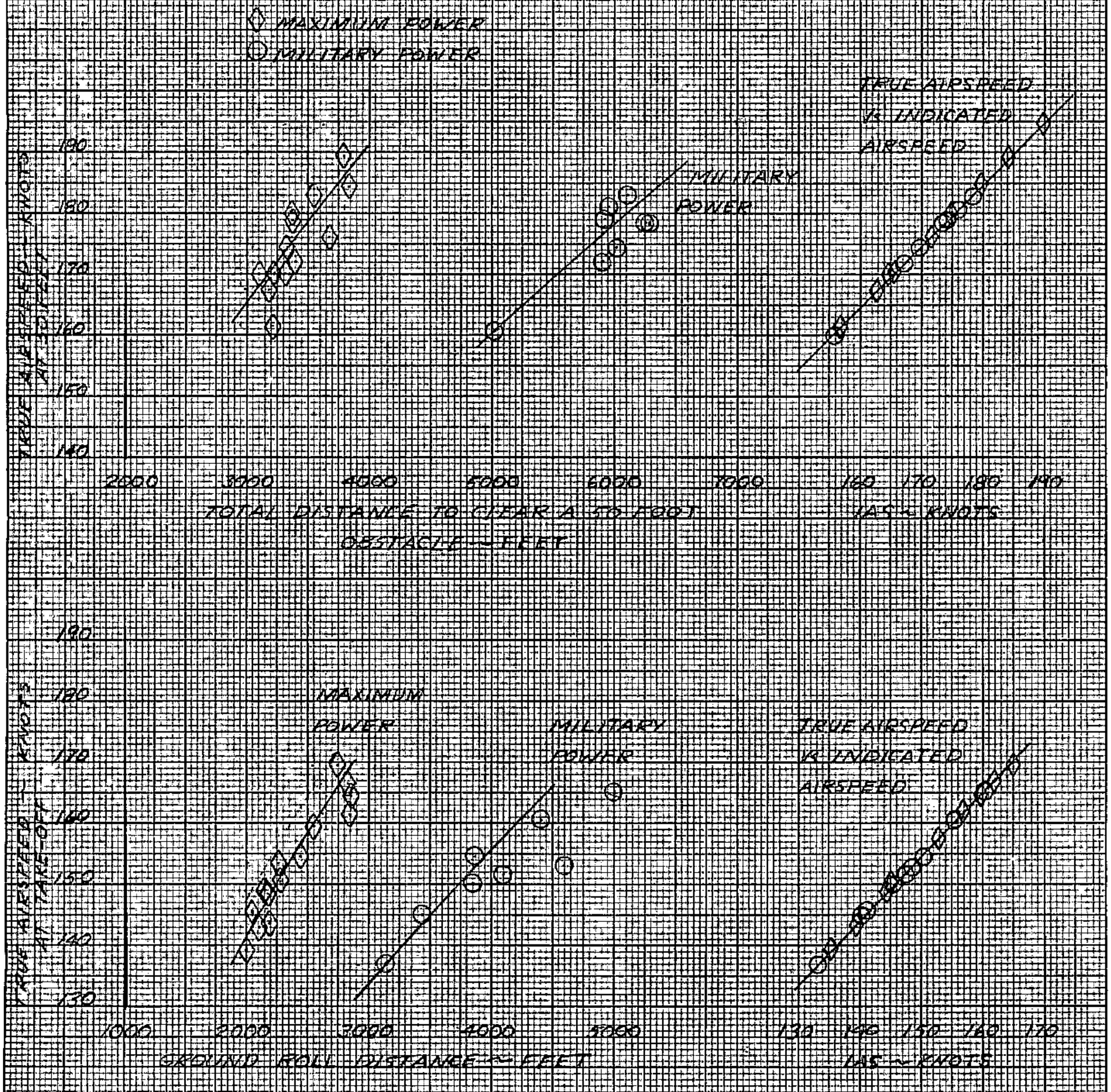


FIGURE NO. 2
TAKE OFF PERFORMANCE
F-102A USAF S/N 53-1791
SEA LEVEL STANDARD CONDITIONS
GROSS WEIGHT 27180 POUNDS

NOTE:
PREVIOUS PHASE IV DATA CORRECTED
TO STANDARD THRUST VALUES USED
IN TANK TEST PROGRAM
F_s MILITARY POWER = 10100 POUNDS
F_s MAXIMUM POWER = 14900 POUNDS



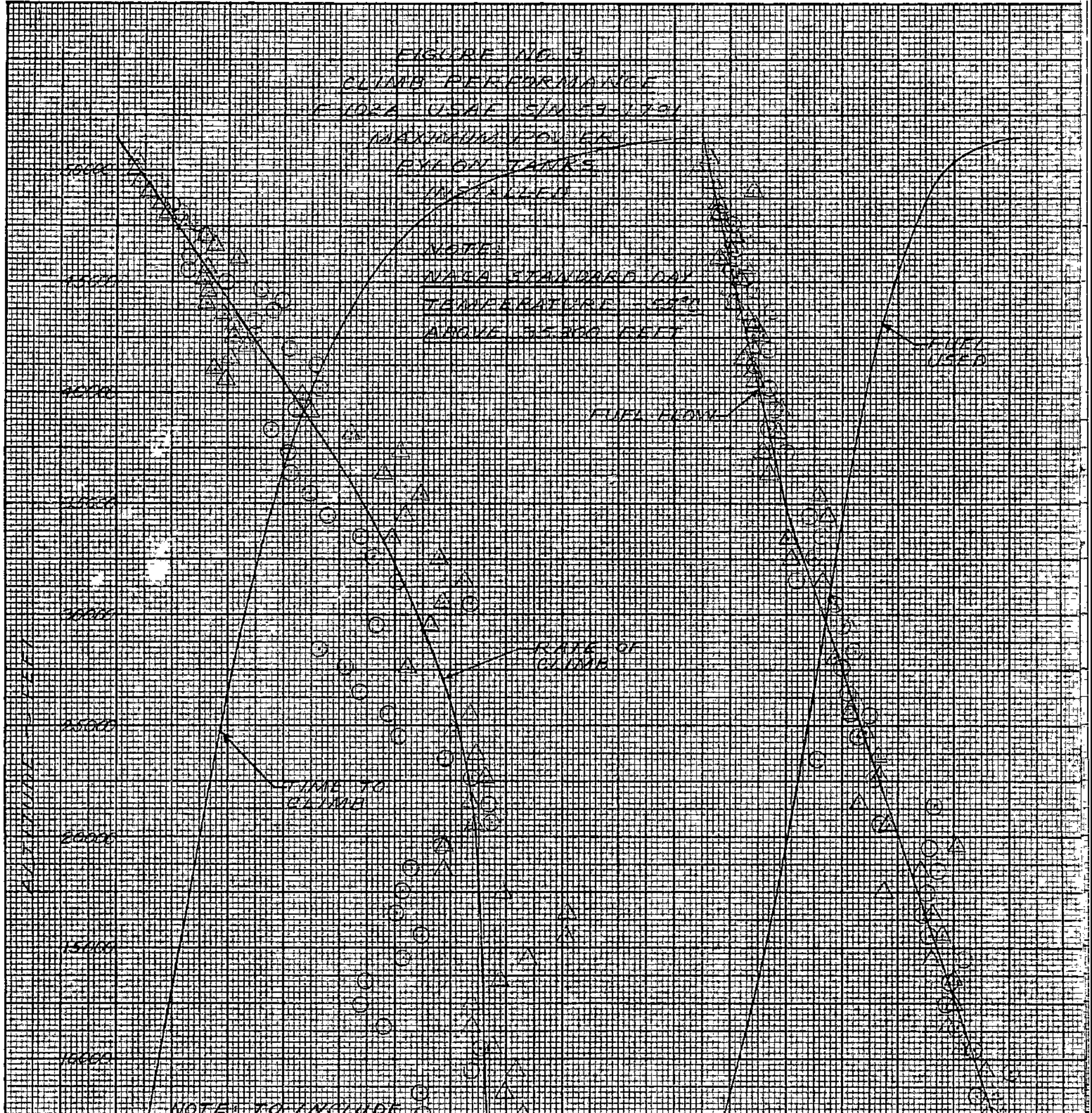


FIGURE NO. 3
CLIMB PERFORMANCE
F-102A USAF S/N 53-1791
MAXIMUM POWER
PYLON TANKS
INSTALLED

NOTE:
NACA STANDARD DAY
TEMPERATURE -55°C
ABOVE 35,300 FEET

NOTE: TO INCLUDE
ACCELERATION TO
CLIMB SPEED ADD
1-MINUTE TO TIME TO
CLIMB AND 500 LBS
TO FUEL USED

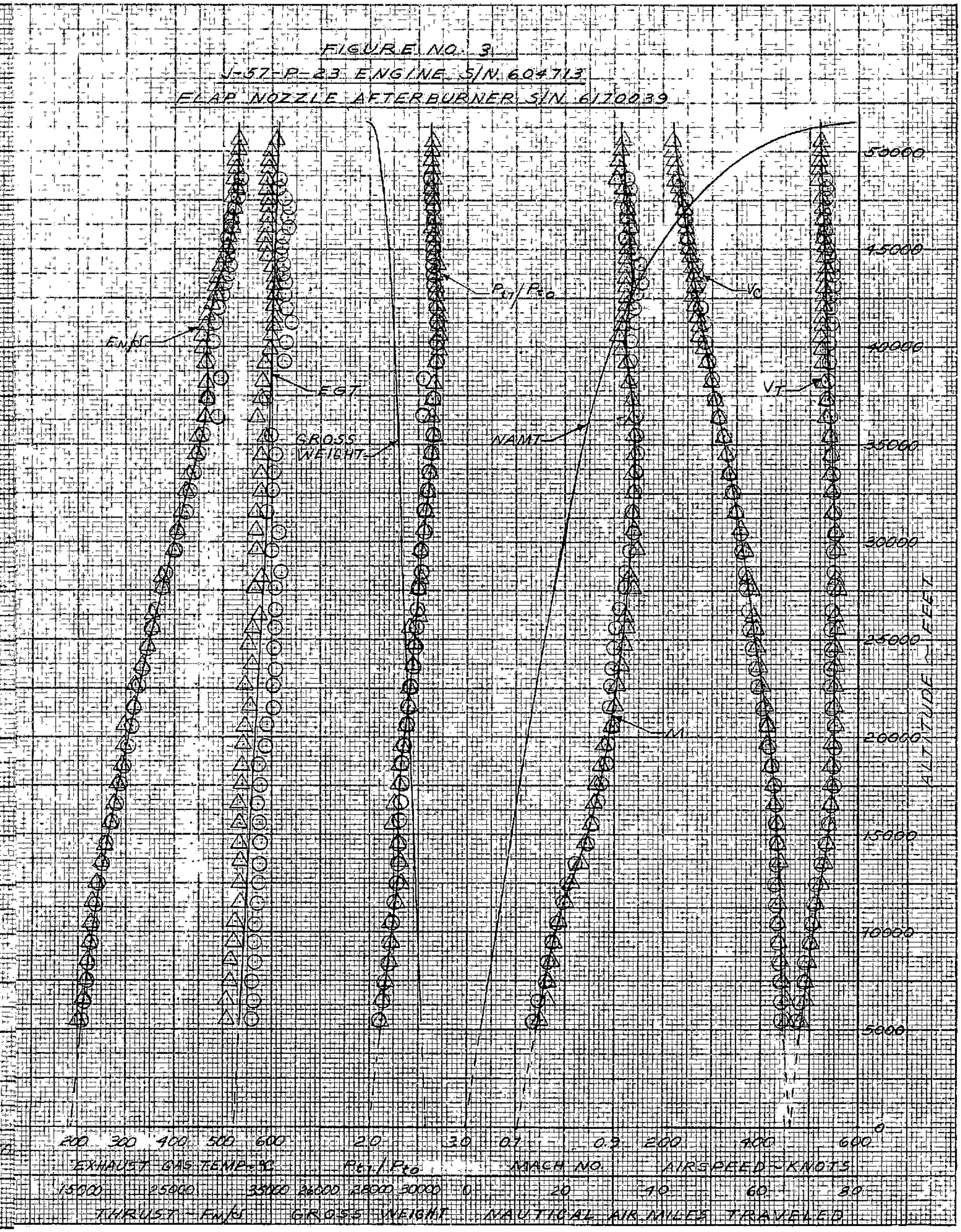


FIGURE NO. 3
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039

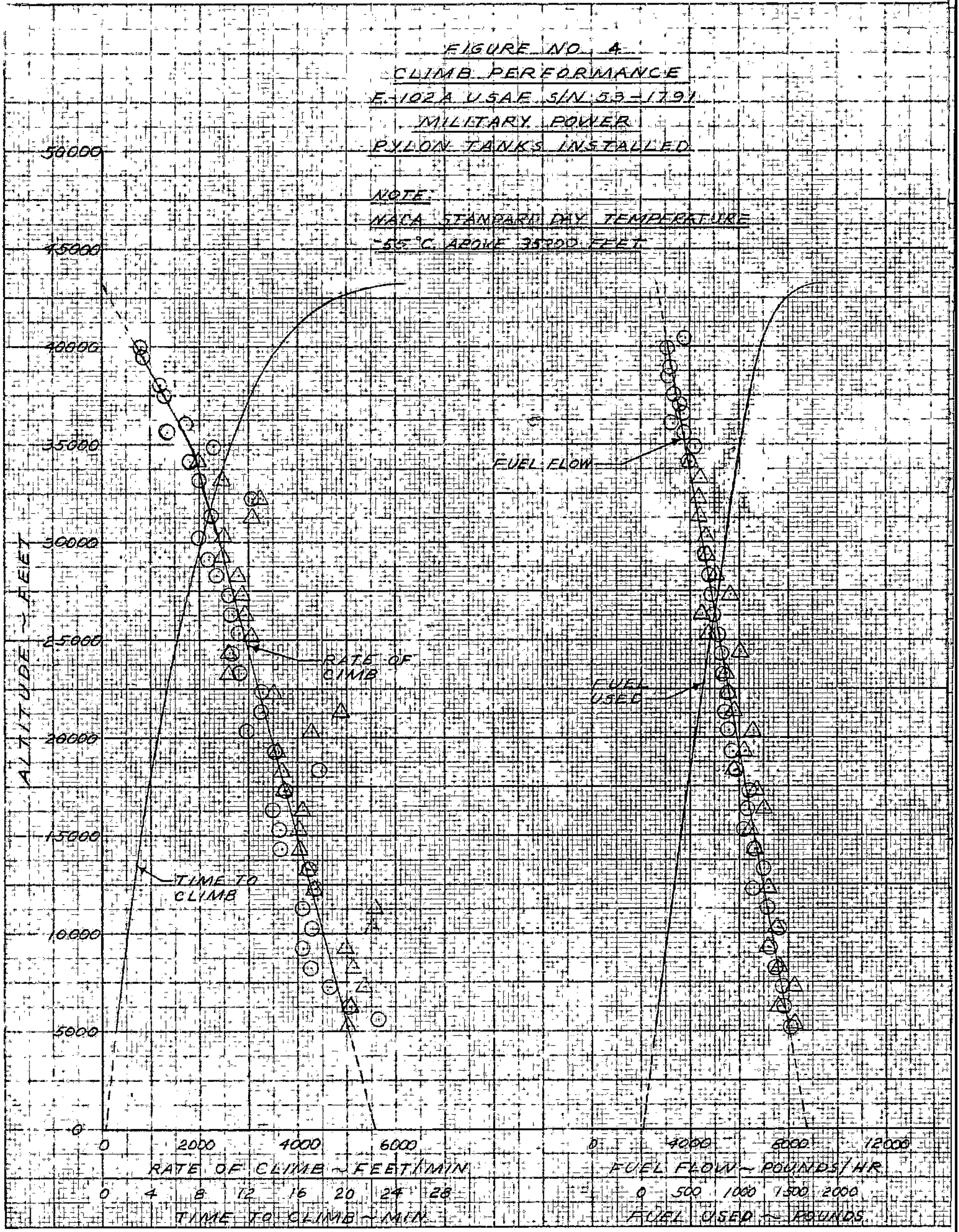


FIGURE NO. 4
CLIMB PERFORMANCE
F-102A USAF S/N 53-1791
MILITARY POWER
PYLON TANKS INSTALLED

NOTE:
NACA STANDARD DAY TEMPERATURE
-55°C. ABOVE 35000 FEET

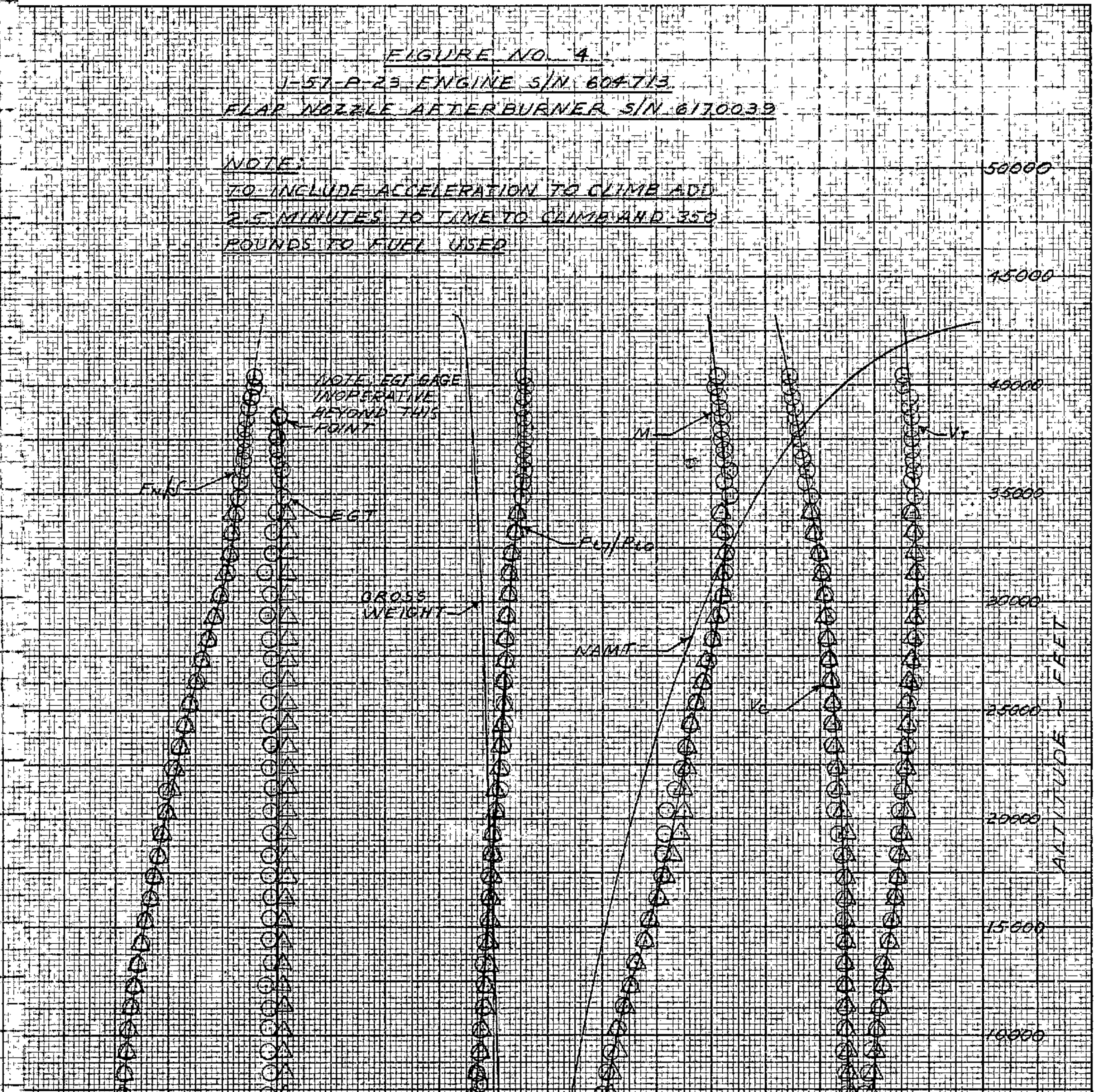


FIGURE NO. 4.

J-57-P-23 ENGINE S/N 604713

FLAP NOZZLE AFTERBURNER S/N 6170039

NOTE:
TO INCLUDE ACCELERATION TO CLIMB ADD
2.5 MINUTES TO TIME TO CLIMB AND 350
POUNDS TO FUEL USED

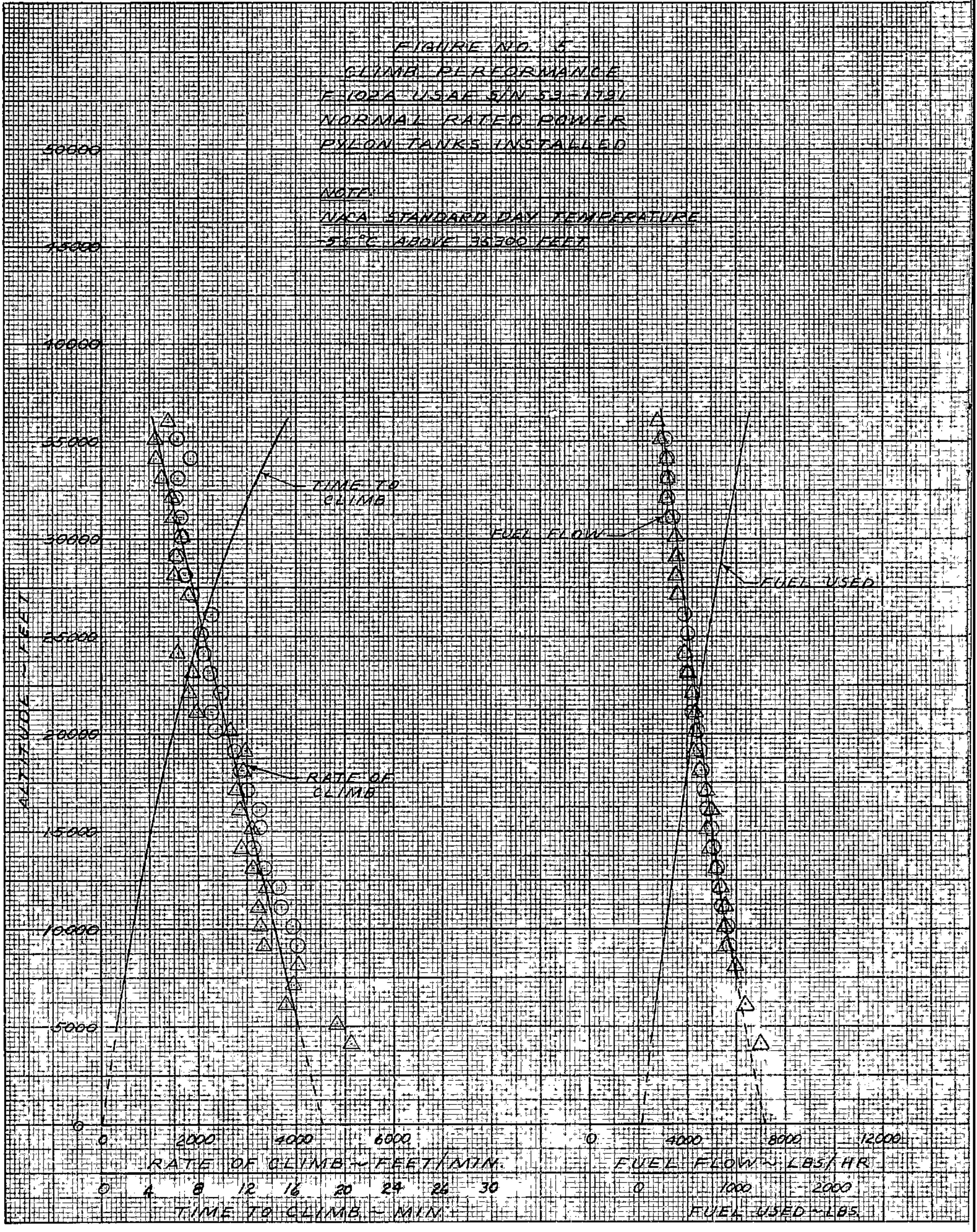


FIGURE NO. 5
CLIMB PERFORMANCE
F-102A USAF S/N 53-1791
NORMAL RATED POWER
PYLON TANKS INSTALLED

NOTE:
NACA STANDARD DAY TEMPERATURE
-56°C ABOVE 35300 FEET

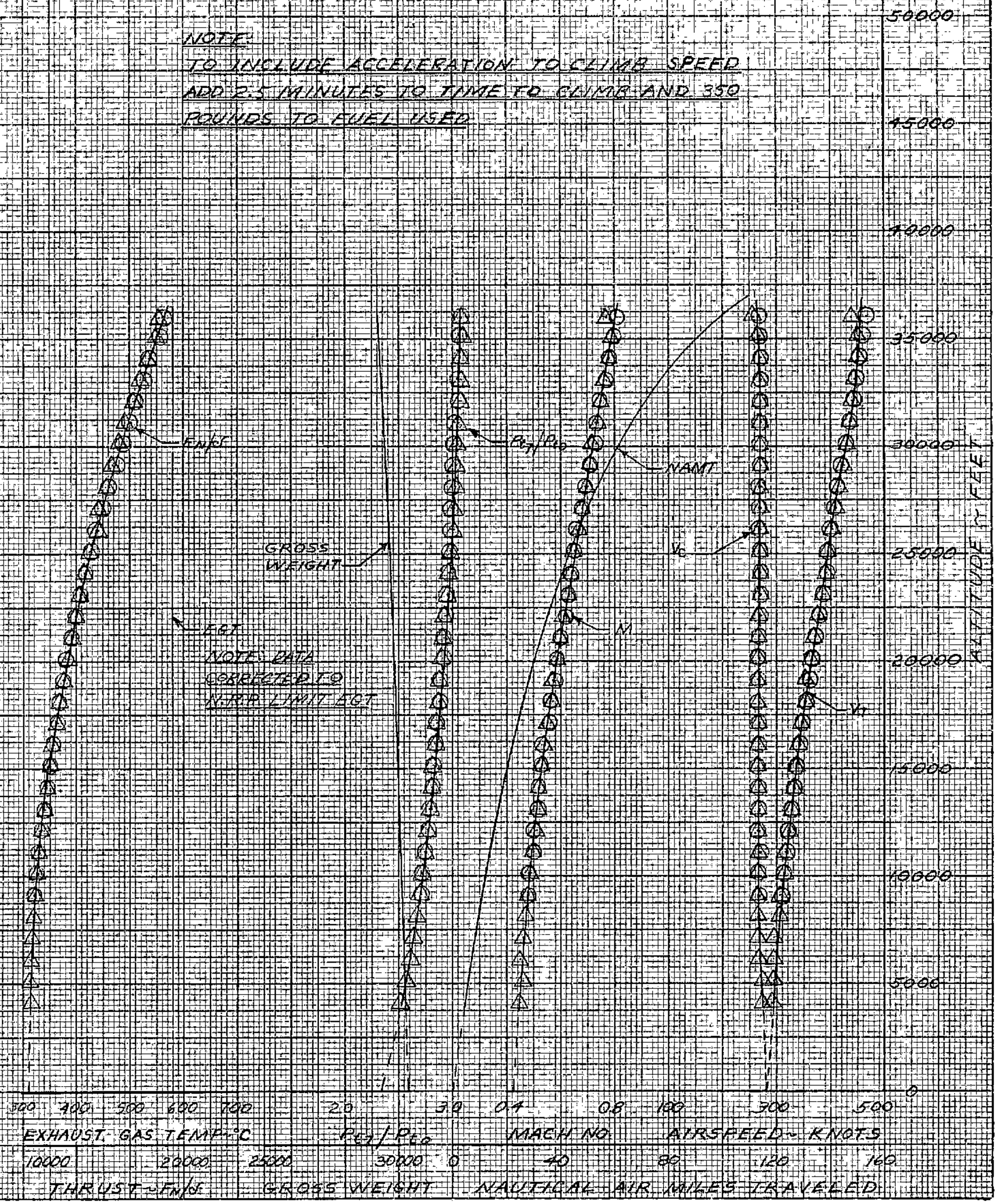
FIGURE NO. 5

J-57-P-23 ENGINE S/N 604713

FLAP NOZZLE AFTERBURNER S/N 6170039

NOTE:

TO INCLUDE ACCELERATION TO CLIMB SPEED ADD 2.5 MINUTES TO TIME TO CLIMB AND 350 POUNDS TO FUEL USED



NOTE: DATA CORRECTED TO N.R.P. LIMIT EGT

FIGURE NO. 6
CLIMB POTENTIAL
F-102A USAF S/N 53-1791
MAXIMUM POWER
J-57 P-23 ENGINE S/N 609713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

ALTITUDE WEIGHT
10 000 FT 30000 LB

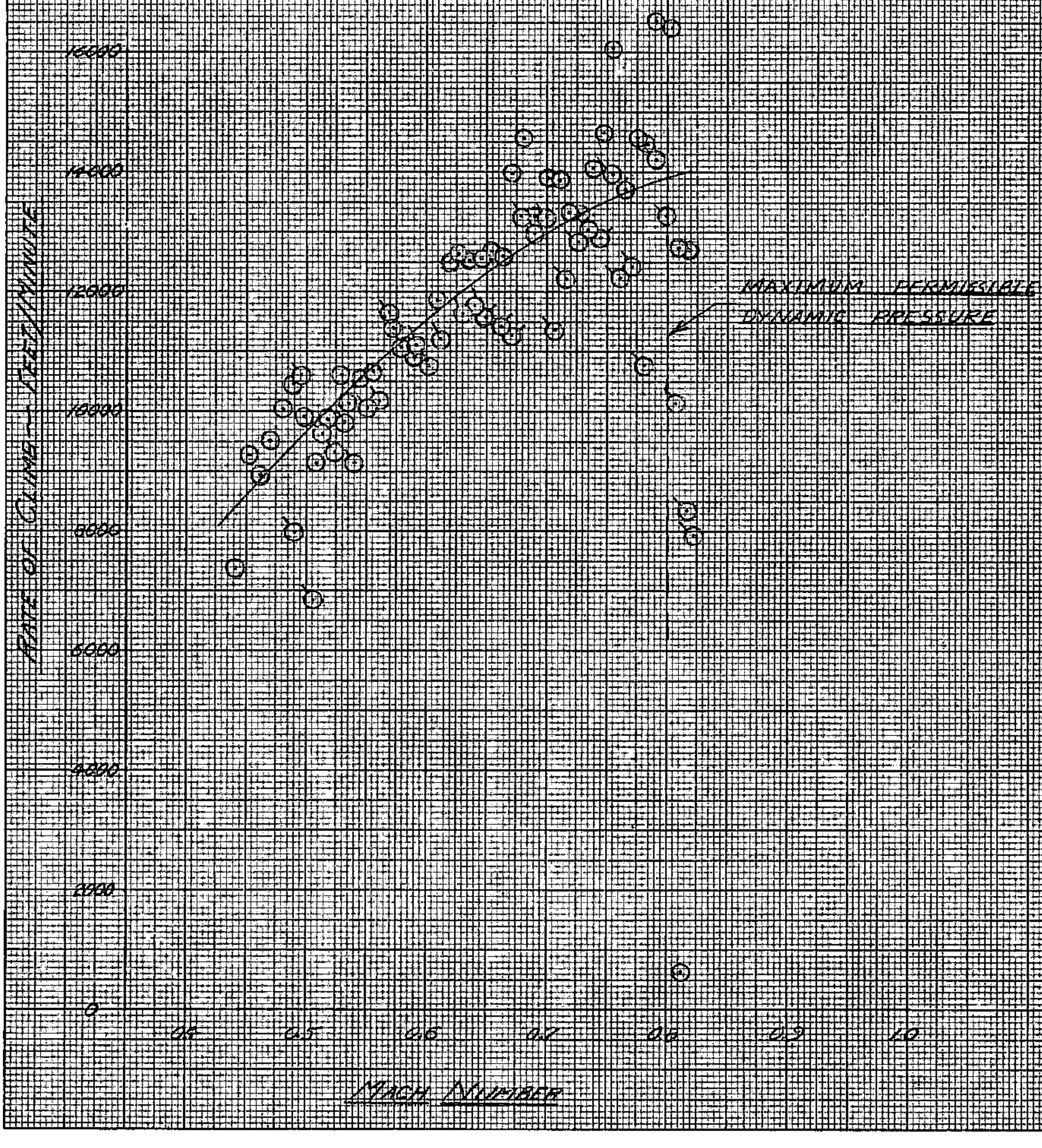


FIGURE NO. 7
CLIMB POTENTIAL
F-102A USAF S/N 53-1791
MAXIMUM POWER
J-57 P-23 ENGINE S/N 609713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

ALTITUDE	WEIGHT
25000 FT	29300 LB

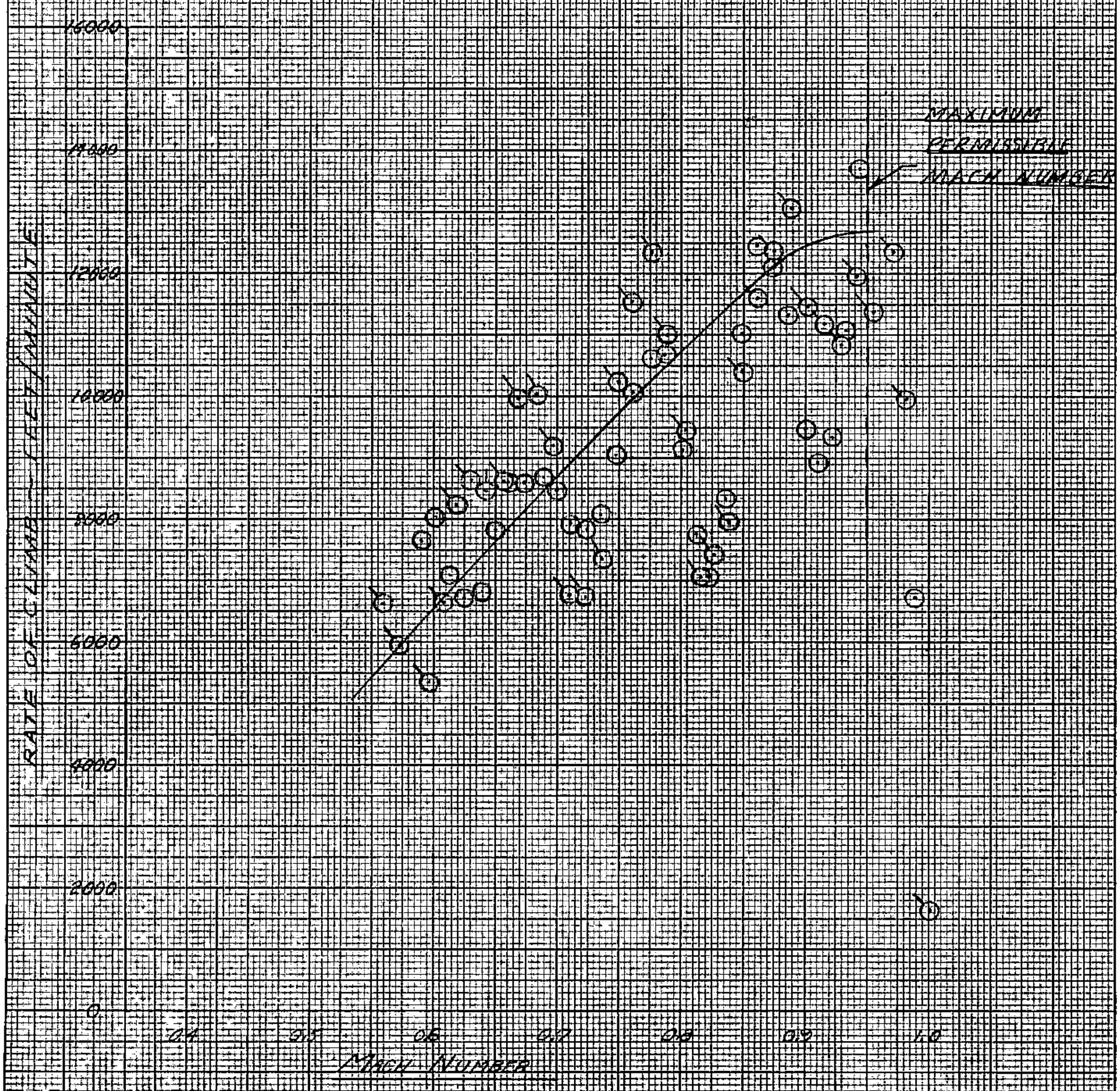


FIGURE NO. 8
CLIMB POTENTIAL
F-102A USAF S/N 53-1791
MAXIMUM POWER
J-57 P-23 ENGINE S/N 609713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

ALTITUDE	WEIGHT
35000 FT	28500 LB

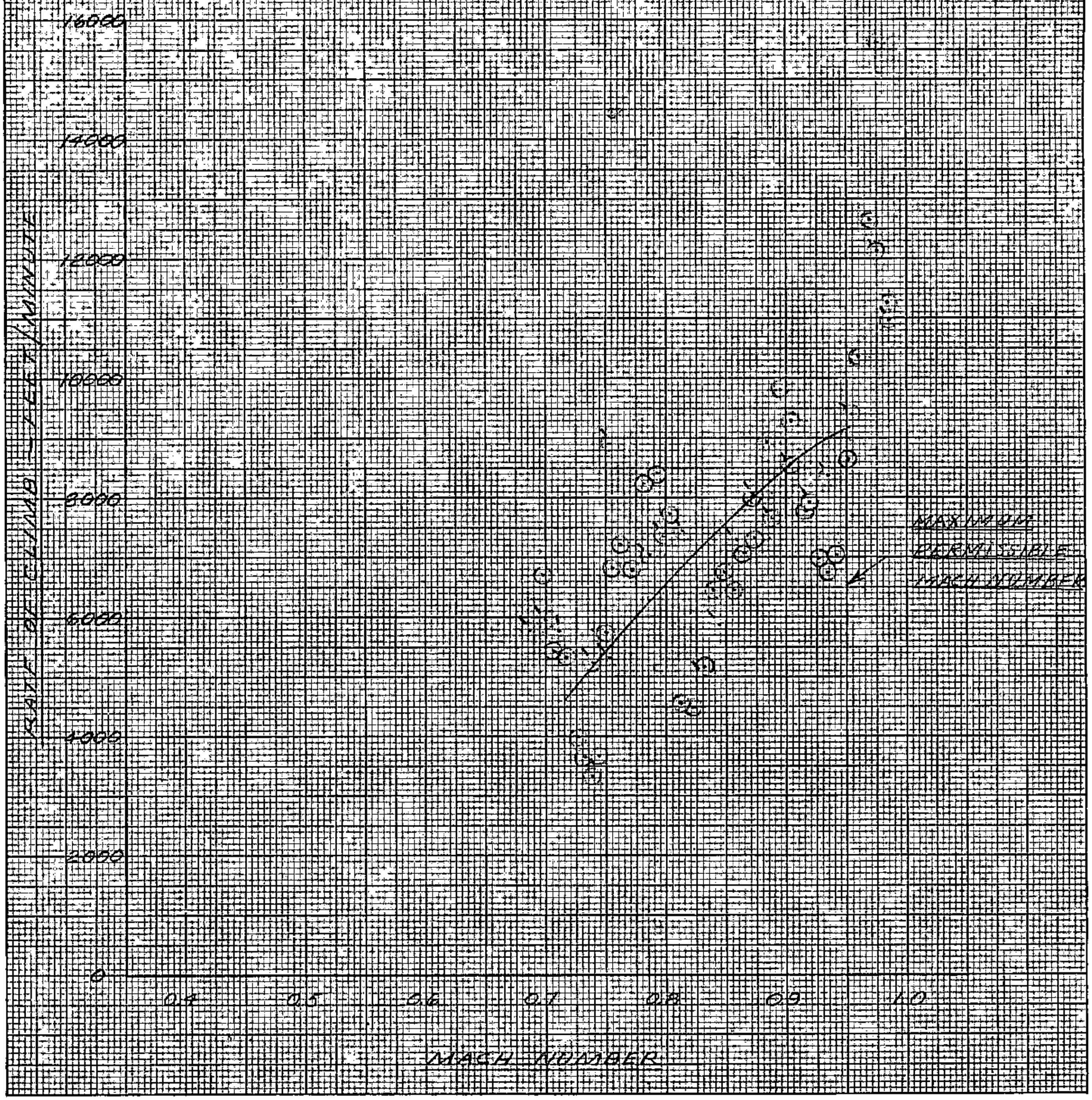


FIGURE NO. 9

CLIMB POTENTIAL

F-102A USAF S/N 53-1791

MILITARY POWER

J-57 P-23 ENGINE S/N 609713

FLAP NOZZLE AFTERBURNER S/N 6170039

PYLON TANKS INSTALLED

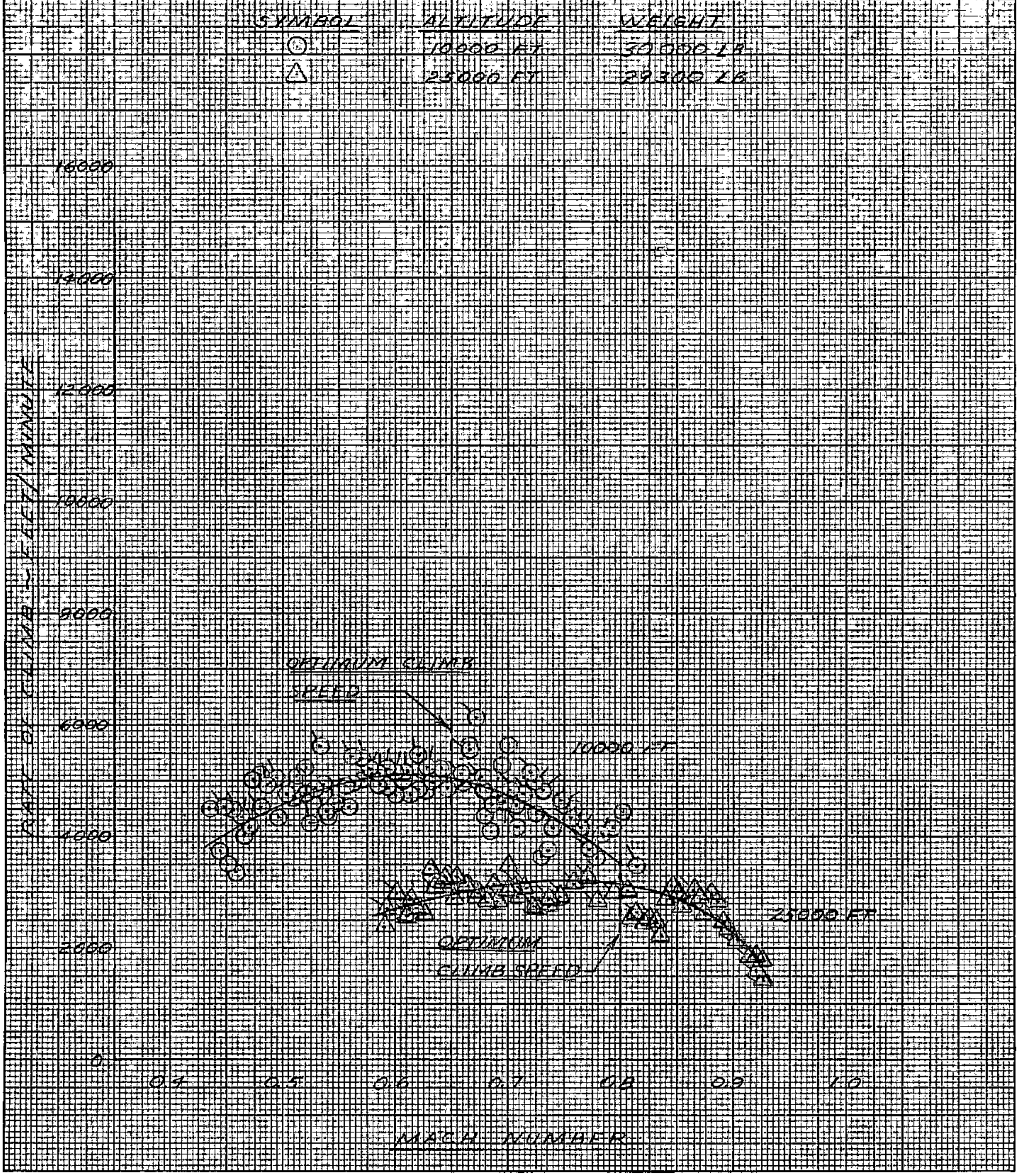


FIGURE NO. 10
CLIMB POTENTIAL
F-102A USAF S/N 53-1791
NORMAL RATED POWER
J-57 P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

SYMBOL	ALTITUDE	WEIGHT
⊙	10000 FT	30000 LB
△	25000 FT	29300 LB

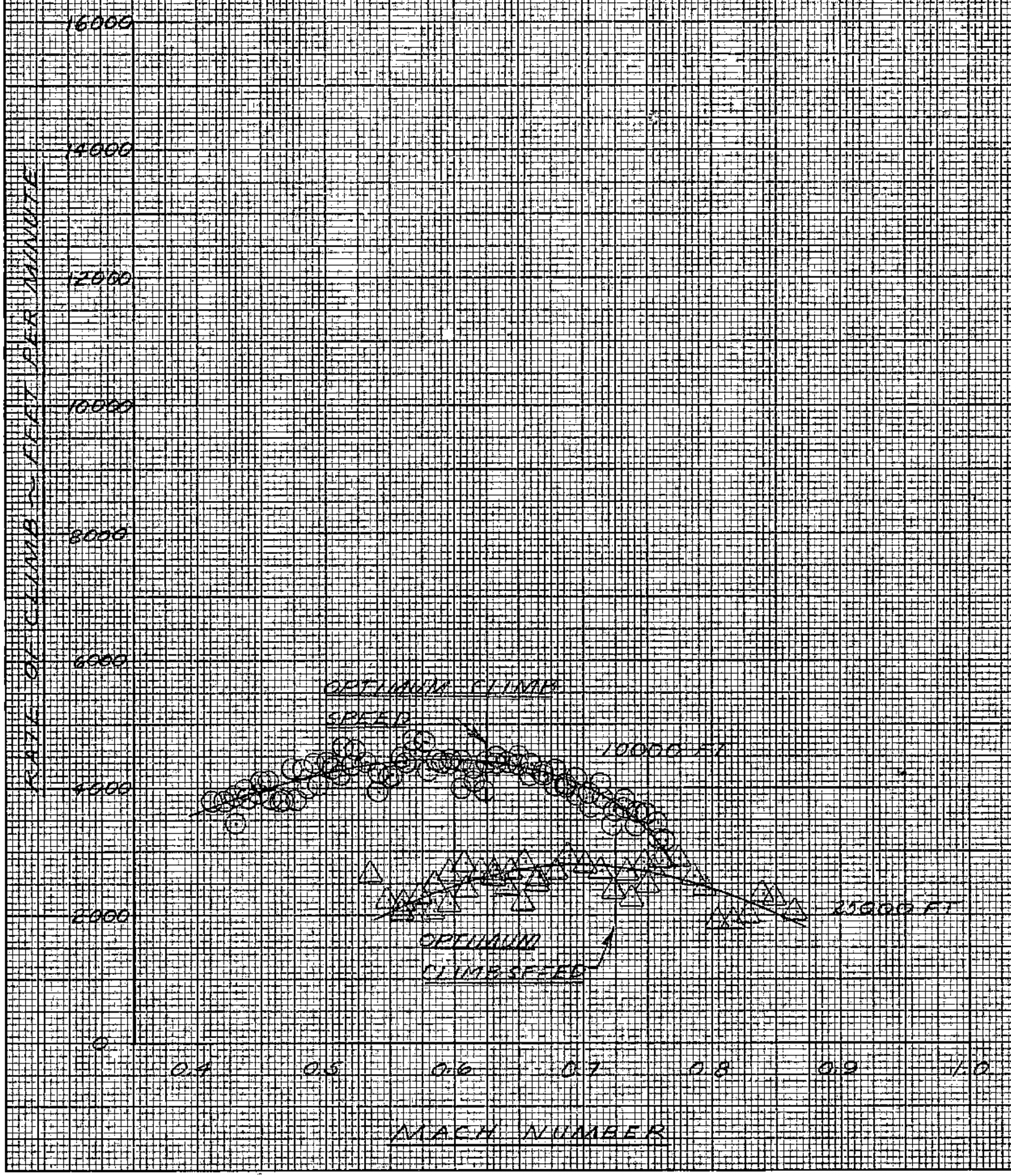


FIGURE NO. 11
PRESSURE RATIO BIAS CURVE
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

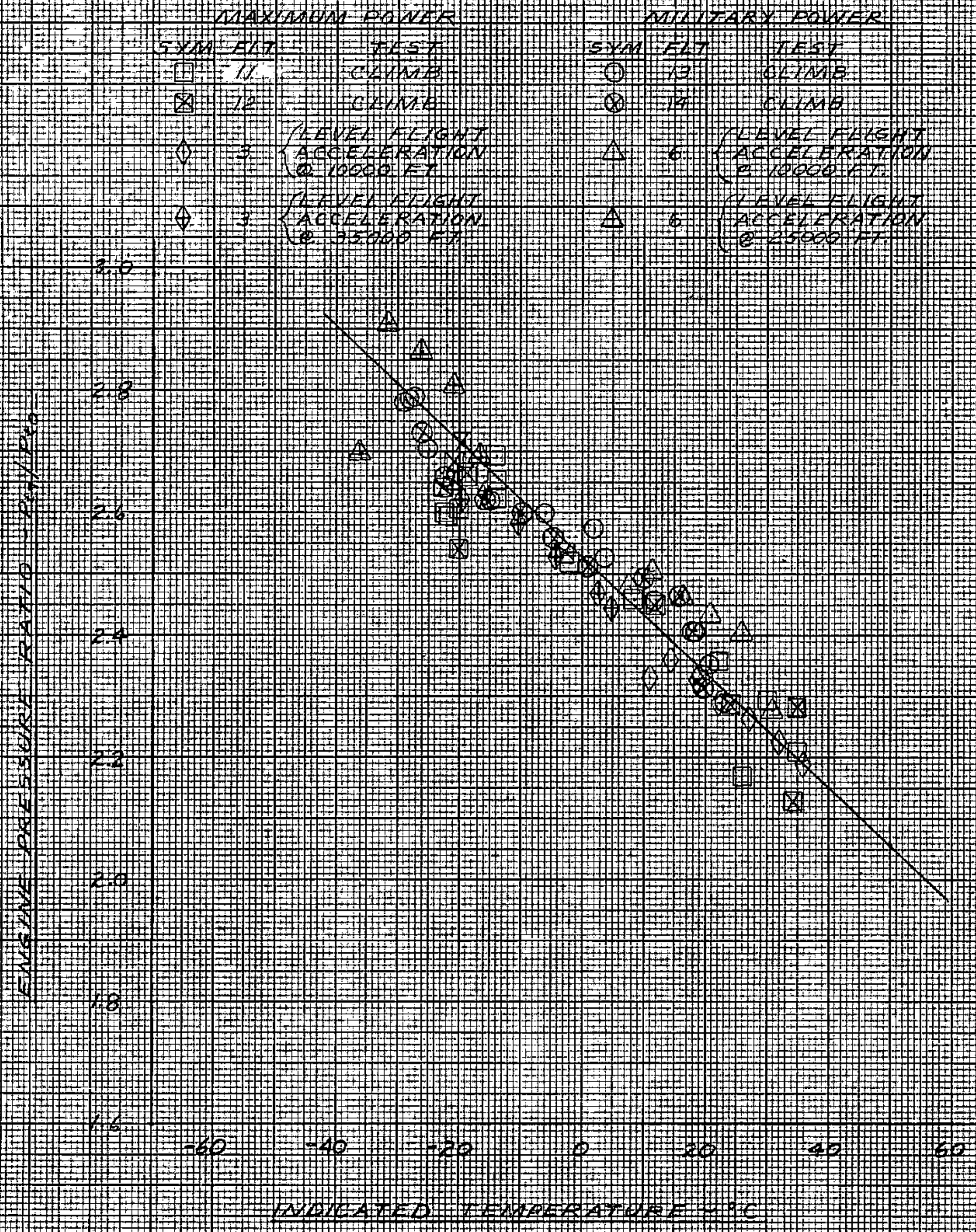


FIGURE NO 12
LEVEL FLIGHT PERFORMANCE
SPECIFIC RANGE
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
NO EXTERNAL STORES

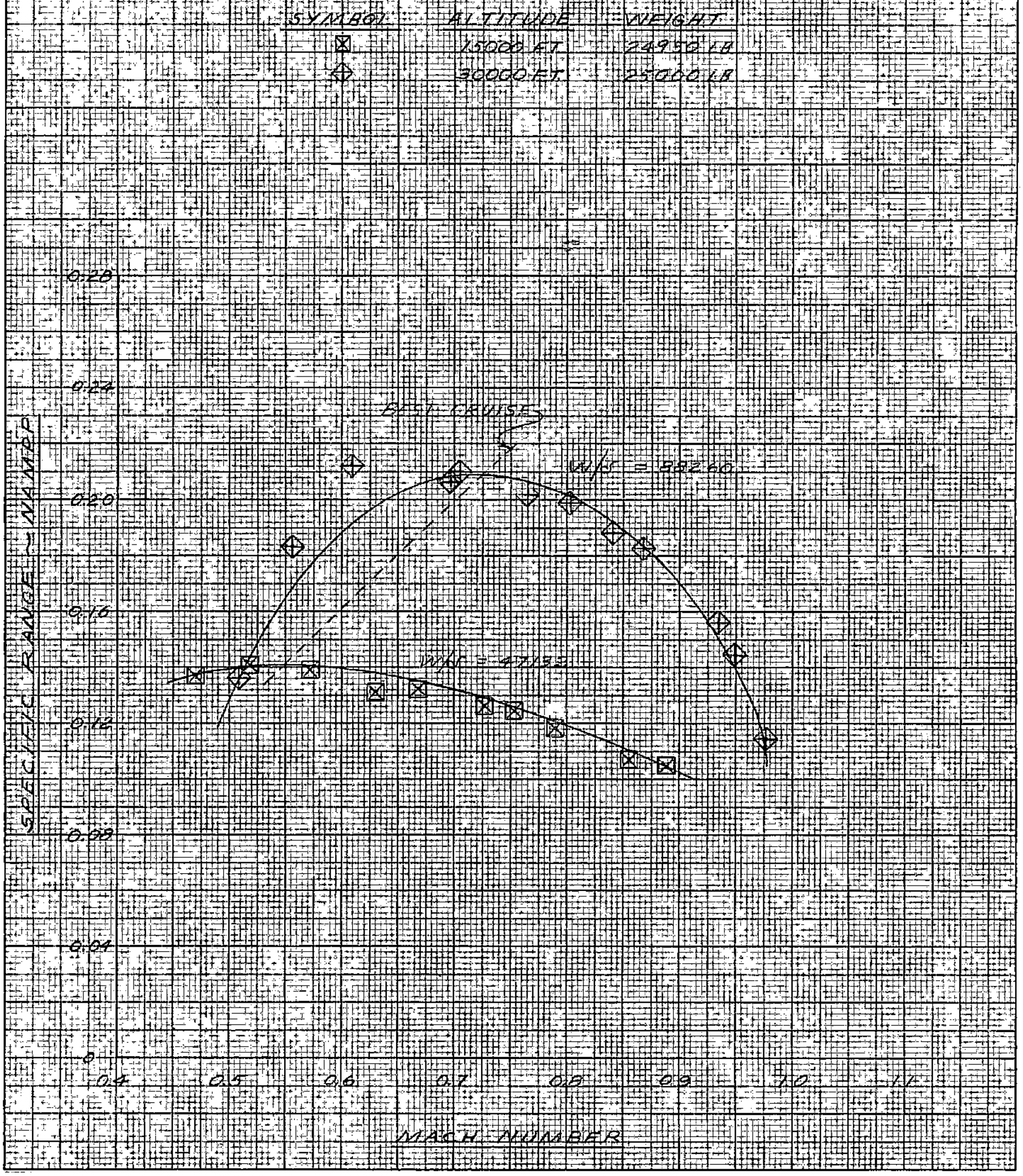


FIGURE NO. 13
LEVEL FLIGHT PERFORMANCE
SPECIFIC RANGE
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

SYMBOL	ALTITUDE	WEIGHT
○	12000 FT	27800 LB
◇	28000 FT	21600 LB
◿	33000 FT	26860 LB
△	38000 FT	26200 LB

NACA STANDARD DAY TEMPERATURE
-55°C ABOVE 35,200 FT.

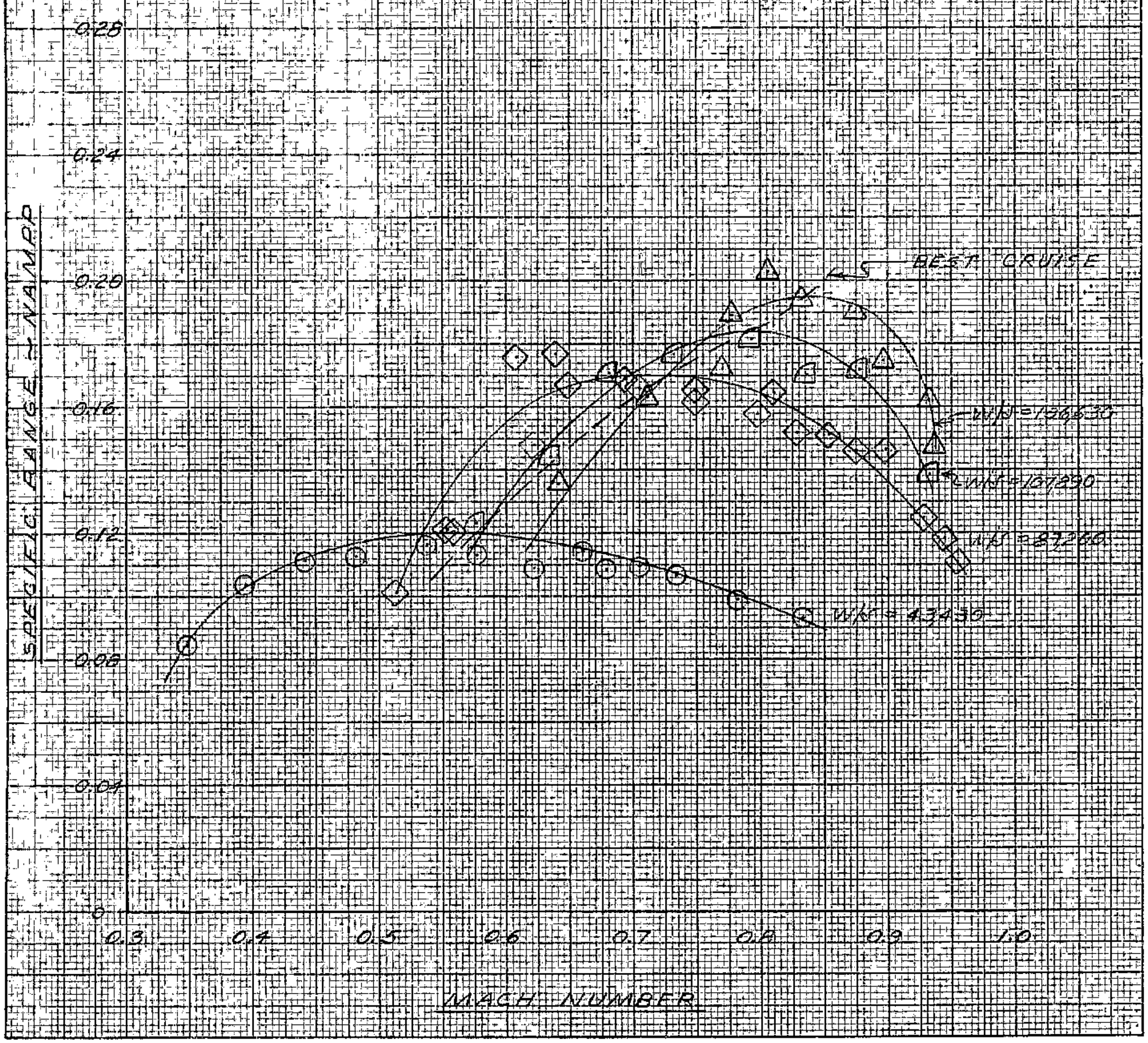


FIGURE NO. 14
LEVEL FLIGHT PERFORMANCE
REQUIRED FUEL CONSUMPTION
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 5110023
NO EXTERNAL STORES

SYMBOL	ALTITUDE	WEIGHT
☒	15000 FT.	24950 LB
◈	30000 FT.	25000 LB

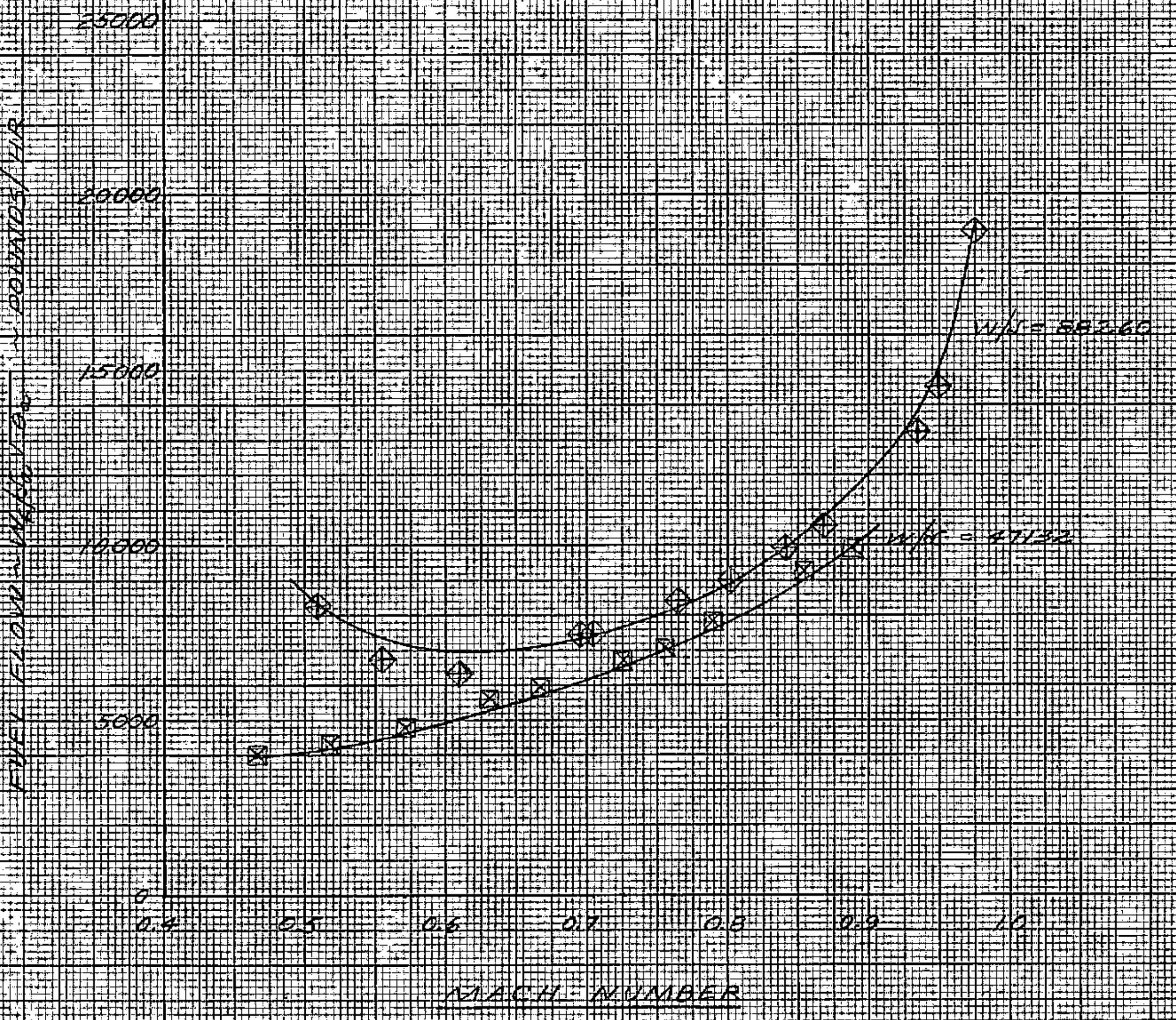
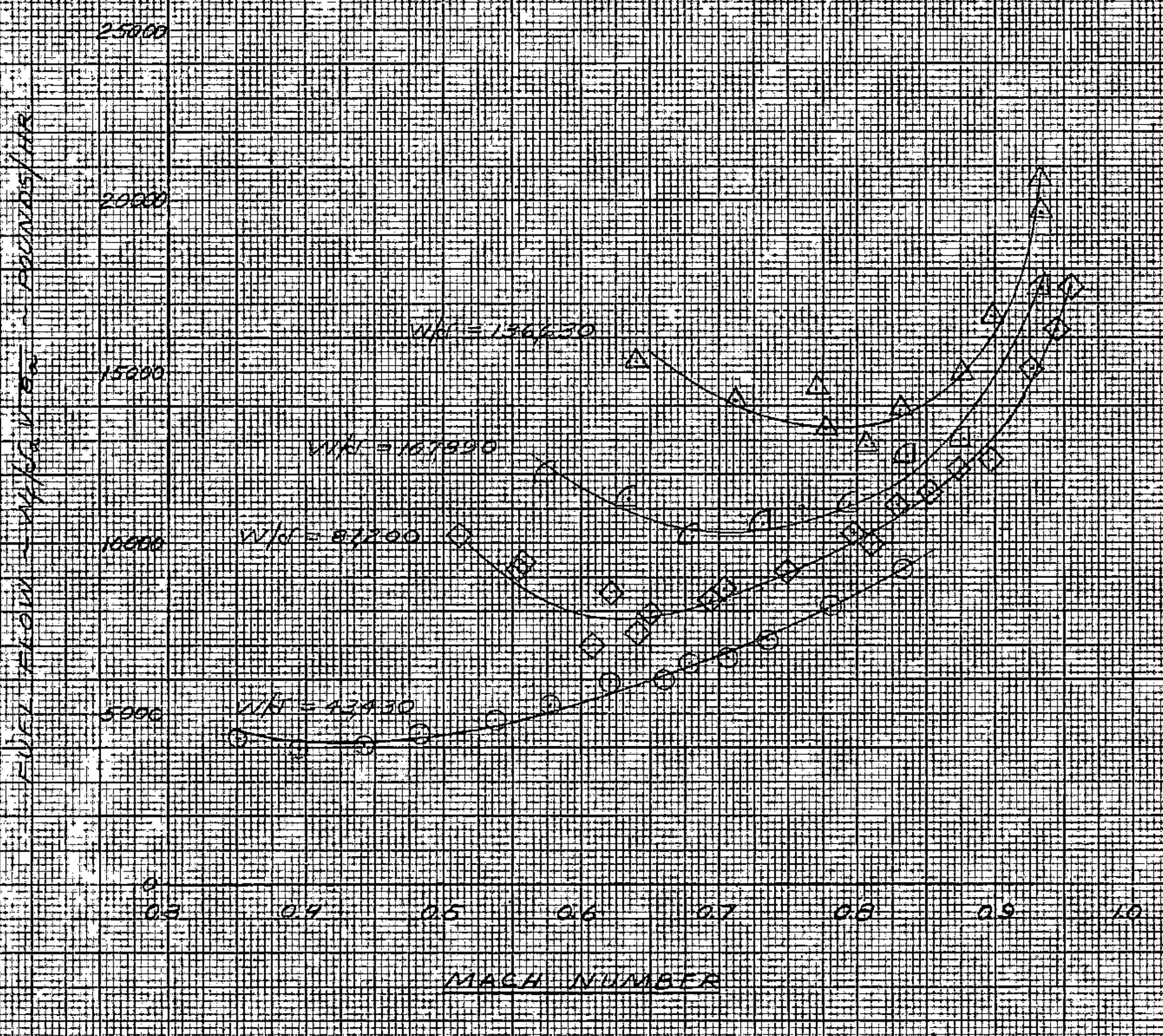


FIGURE NO. 15
LEVEL FLIGHT PERFORMANCE
REQUIRED FUEL CONSUMPTION
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

SYMBOL	ALTITUDE	WEIGHT
○	12,000 FT	27800 LB
◇	28,000 FT	26600 LB
◿	33,000 FT	26800 LB
△	38,000 FT	26200 LB

NACA STANDARD DAY TEMPERATURE
-55°C ABOVE 35300 FEET



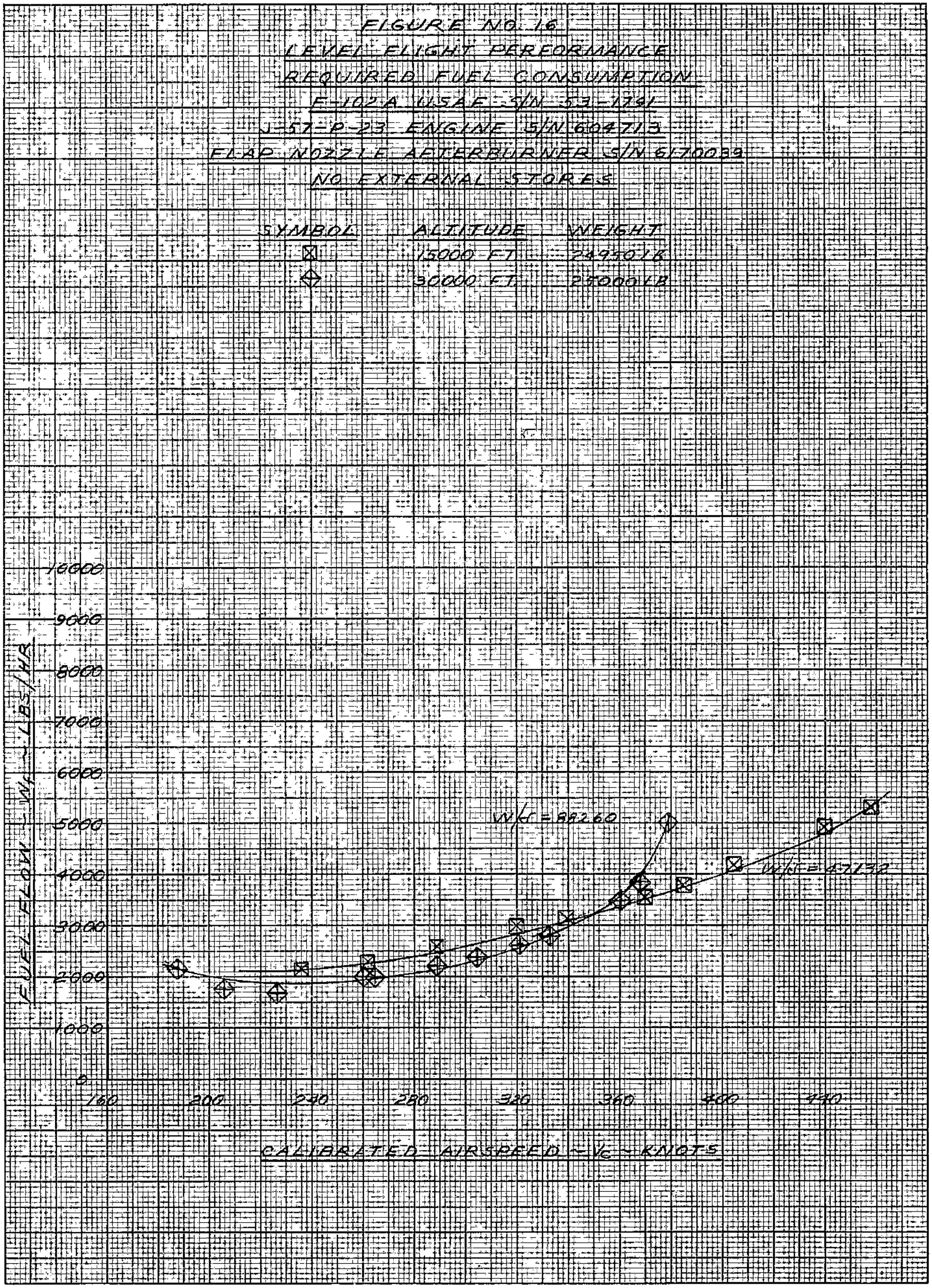


FIGURE NO. 16
LEVEL FLIGHT PERFORMANCE
REQUIRED FUEL CONSUMPTION
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
NO EXTERNAL STORES

FIGURE NO. 17
LEVEL FLIGHT PERFORMANCE
REQUIRED FUEL CONSUMPTION
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

SYMBOL	ALTITUDE	WEIGHT
⊙	12000 FT	27800 LB
◇	28000 FT	26600 LB
◭	33000 FT	26800 LB
△	38000 FT	26200 LB

NACA STANDARD DAY TEMPERATURE
-55°C ABOVE 35,300 FEET

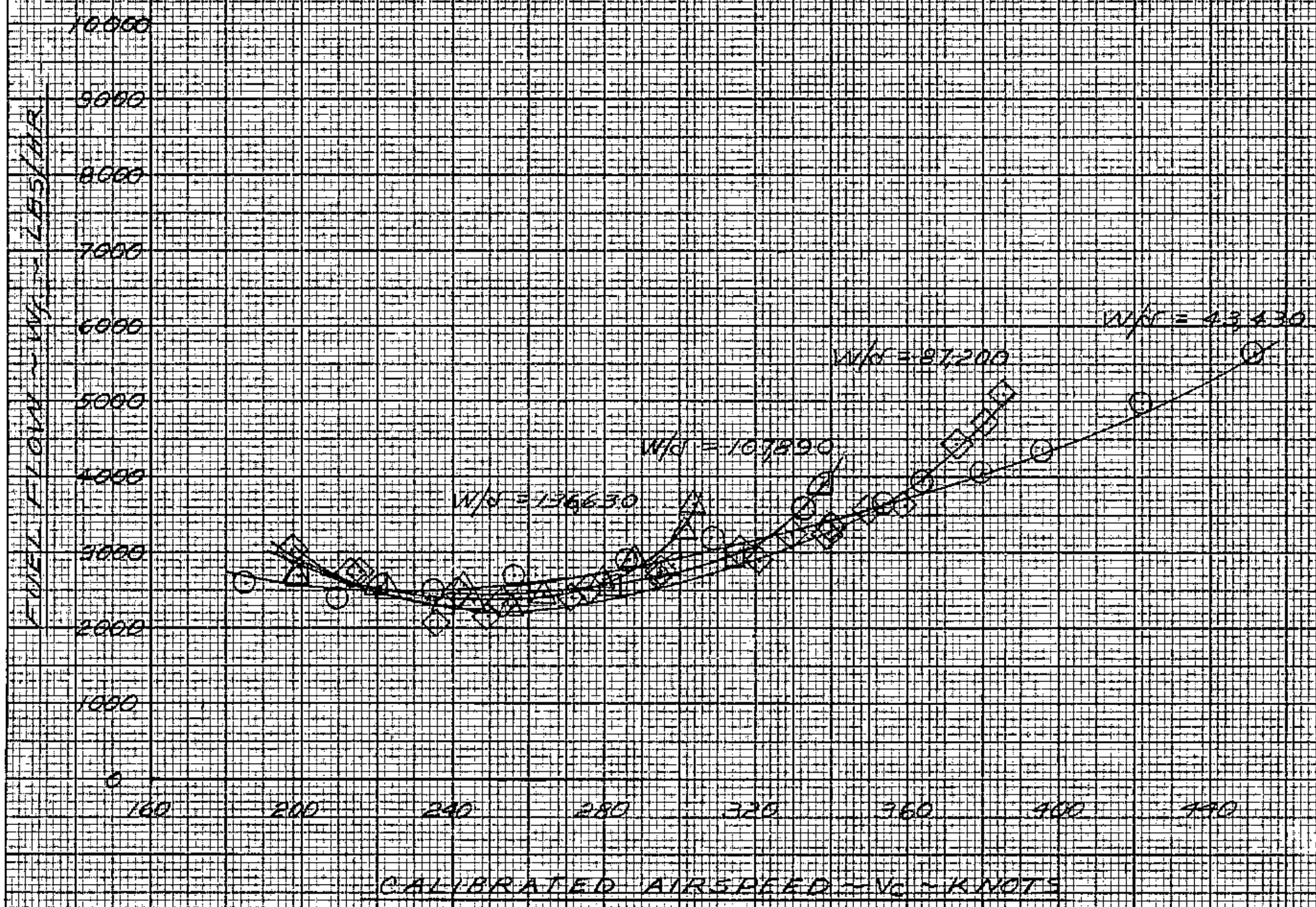
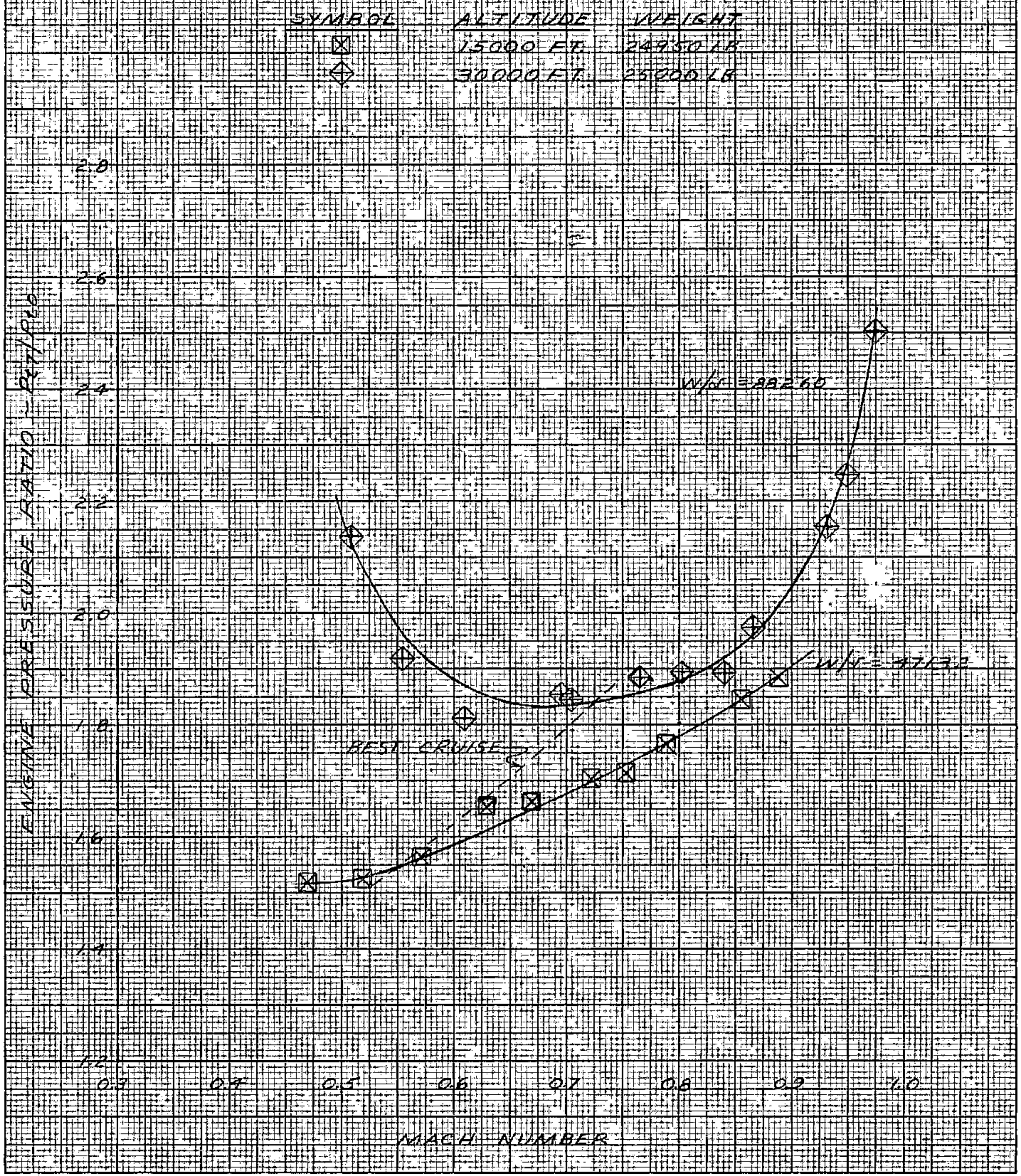


FIGURE NO. 18
LEVEL FLIGHT PERFORMANCE
POWER REQUIRED
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 609713
FLAP NOZZLE AFTERBURNER S/N 6170039
NO EXTERNAL STORES



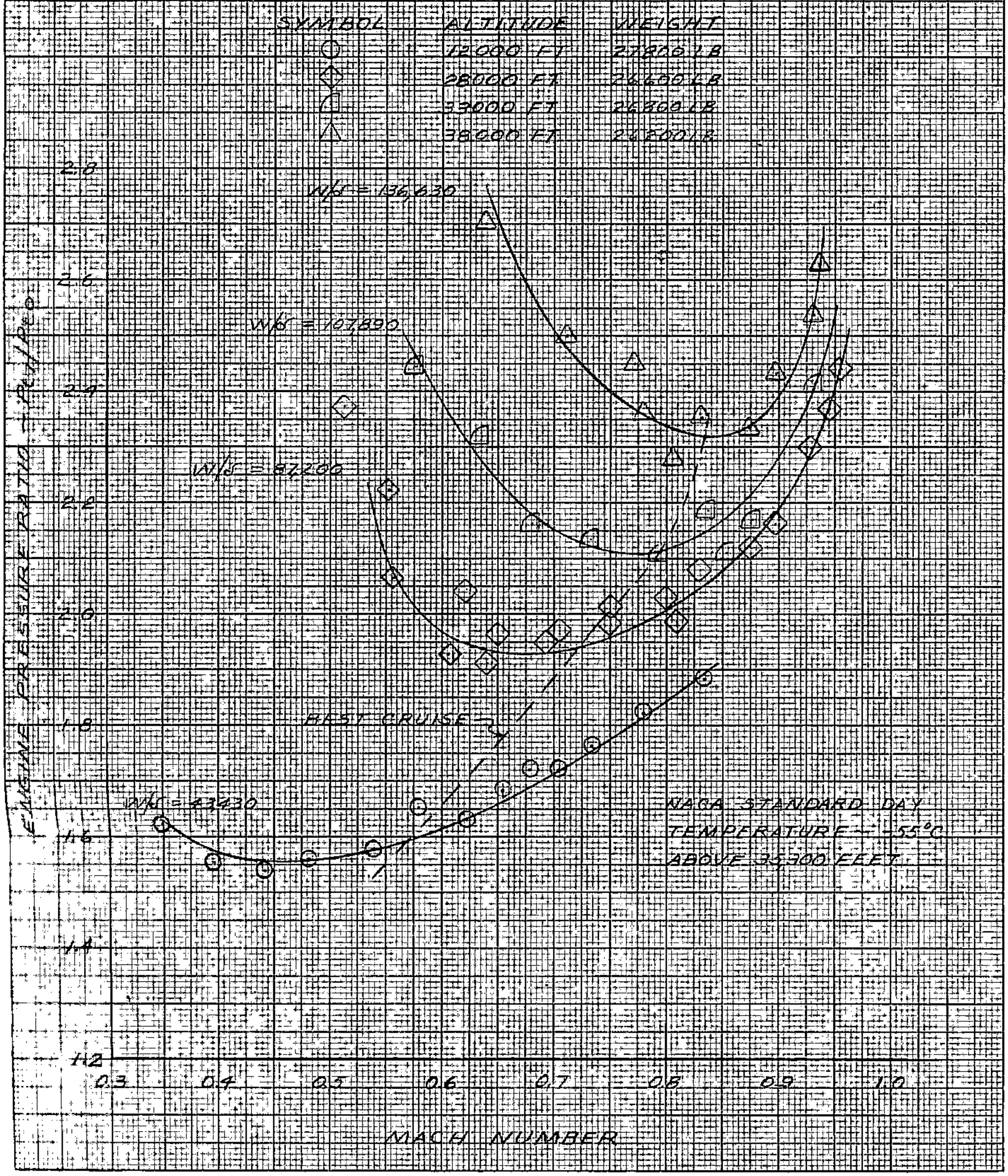


FIGURE NO. 19
LEVEL FLIGHT PERFORMANCE
POWER REQUIRED
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 609713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

SYMBOL	ALTITUDE	WEIGHT
○	12000 FT	27800 LB
◇	28000 FT	26600 LB
□	33000 FT	26800 LB
△	38000 FT	26200 LB

FIGURE NO. 20

LEVEL FLIGHT PERFORMANCE

POWER REQUIRED

F-102A USAF S/N 53-1791

J-57-P-23 ENGINE S/N 609713

FLAP NOZZLE AFTERBURNER S/N 6170039

NO EXTERNAL STORES

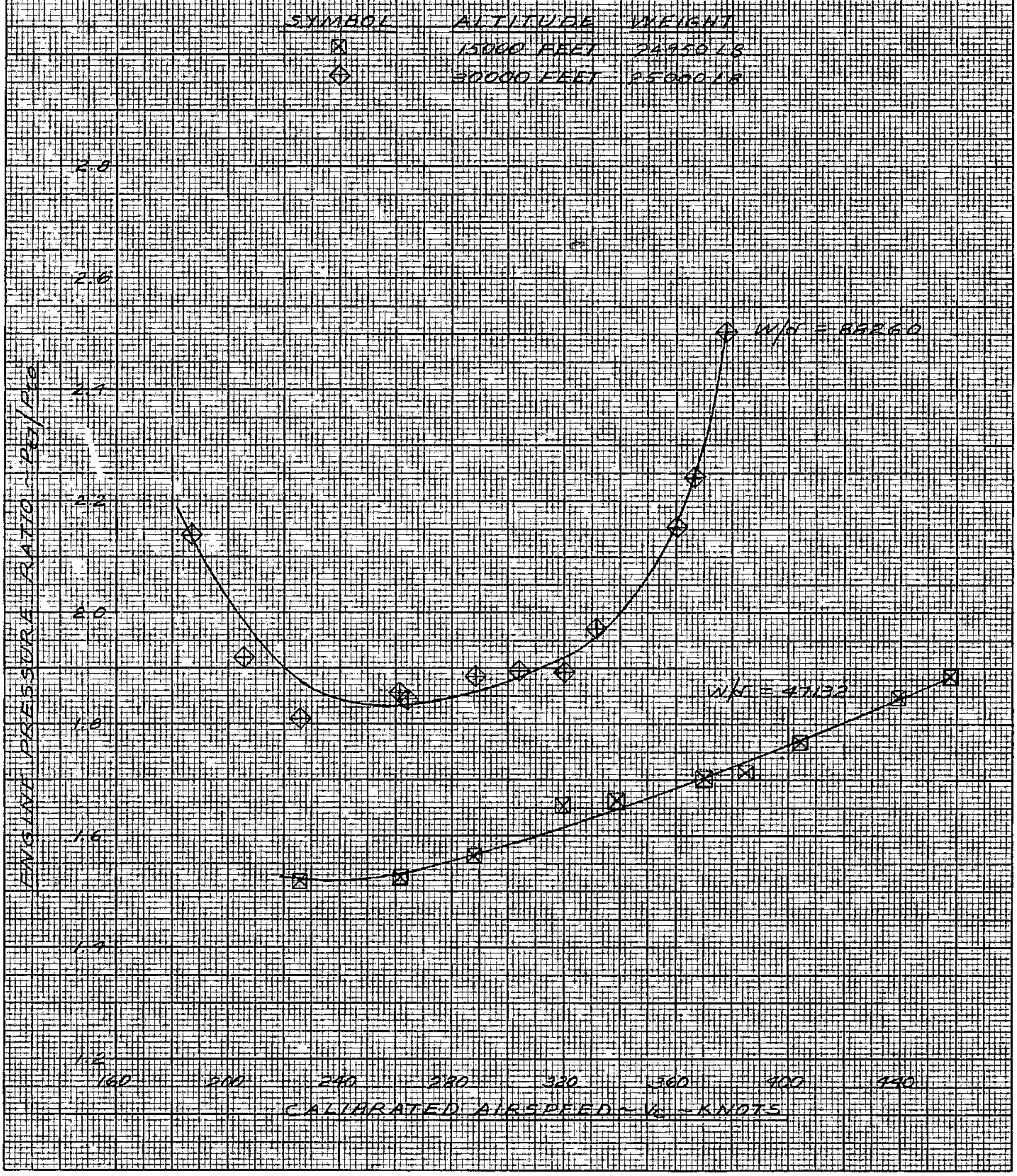


FIGURE NO 21
LEVEL FLIGHT PERFORMANCE
POWER REQUIRED
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

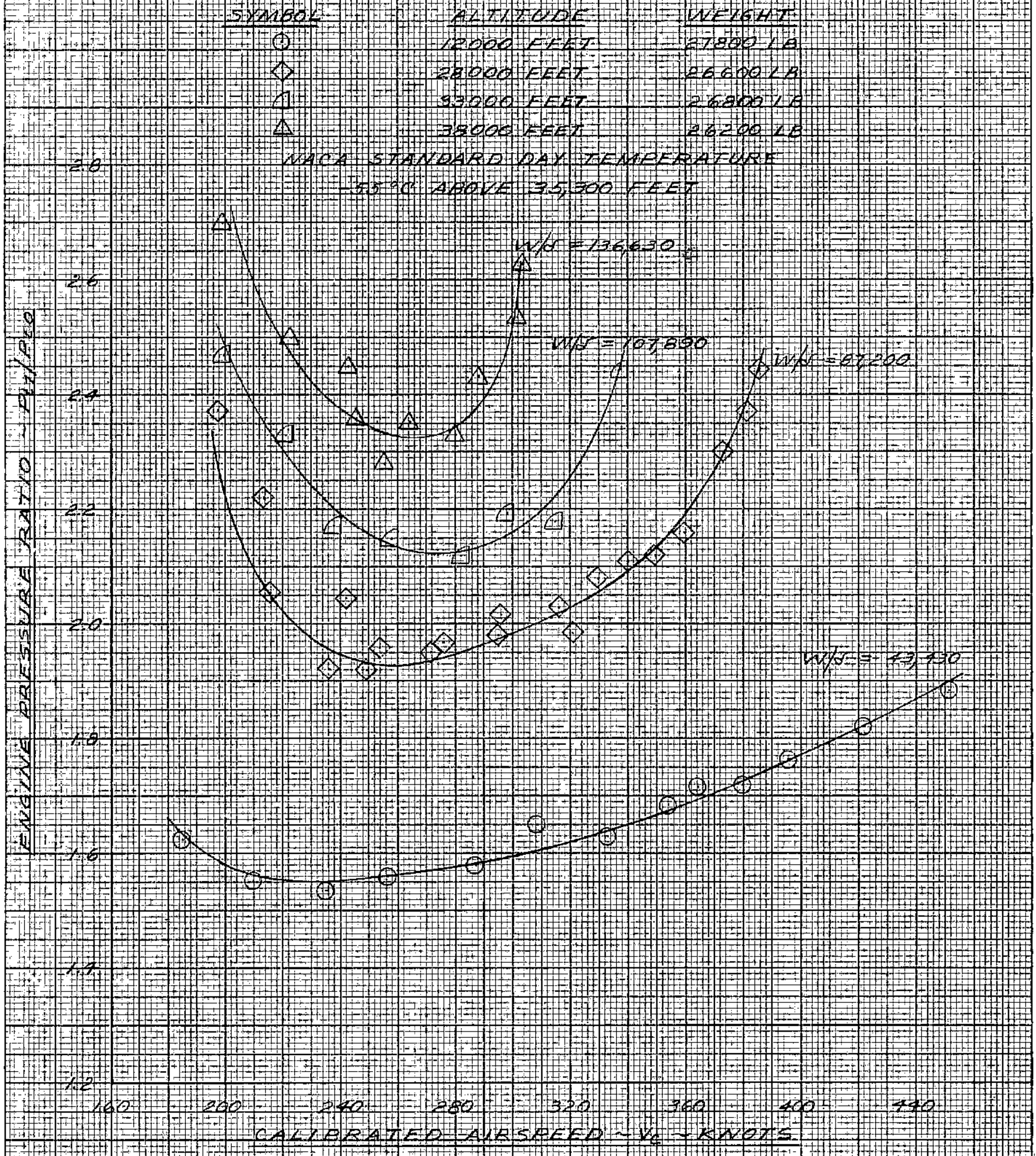


FIGURE NO. 22
LEVEL FLIGHT PERFORMANCE
THRUST REQUIRED
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
NO EXTERNAL STORES

SYMBOL	ALTITUDE	WEIGHT
☒	15000 FT	24950 LB
◇	30000 FT	25000 LB

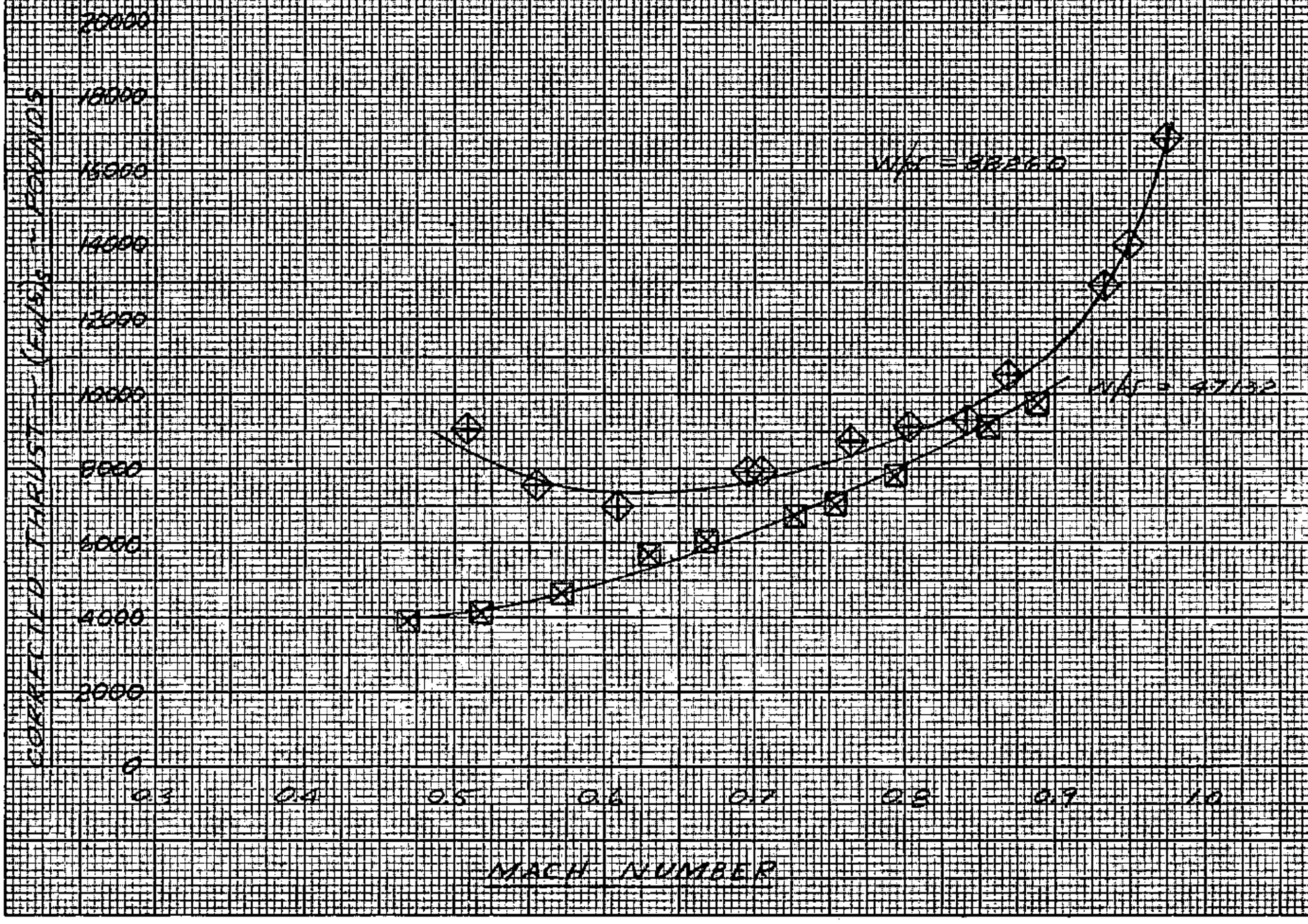


FIGURE NO. 23
LEVEL FLIGHT PERFORMANCE
THRUST REQUIRED
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039
PYLON TANKS INSTALLED

SYMBOL	ALTITUDE	WEIGHT
○	12000 FT	27800 LB
◇	28000 FT	26600 LB
□	33000 FT	26800 LB
△	38000 FT	26200 LB

NACA STANDARD DAY TEMPERATURE
-55°C ABOVE 35,300 FEET

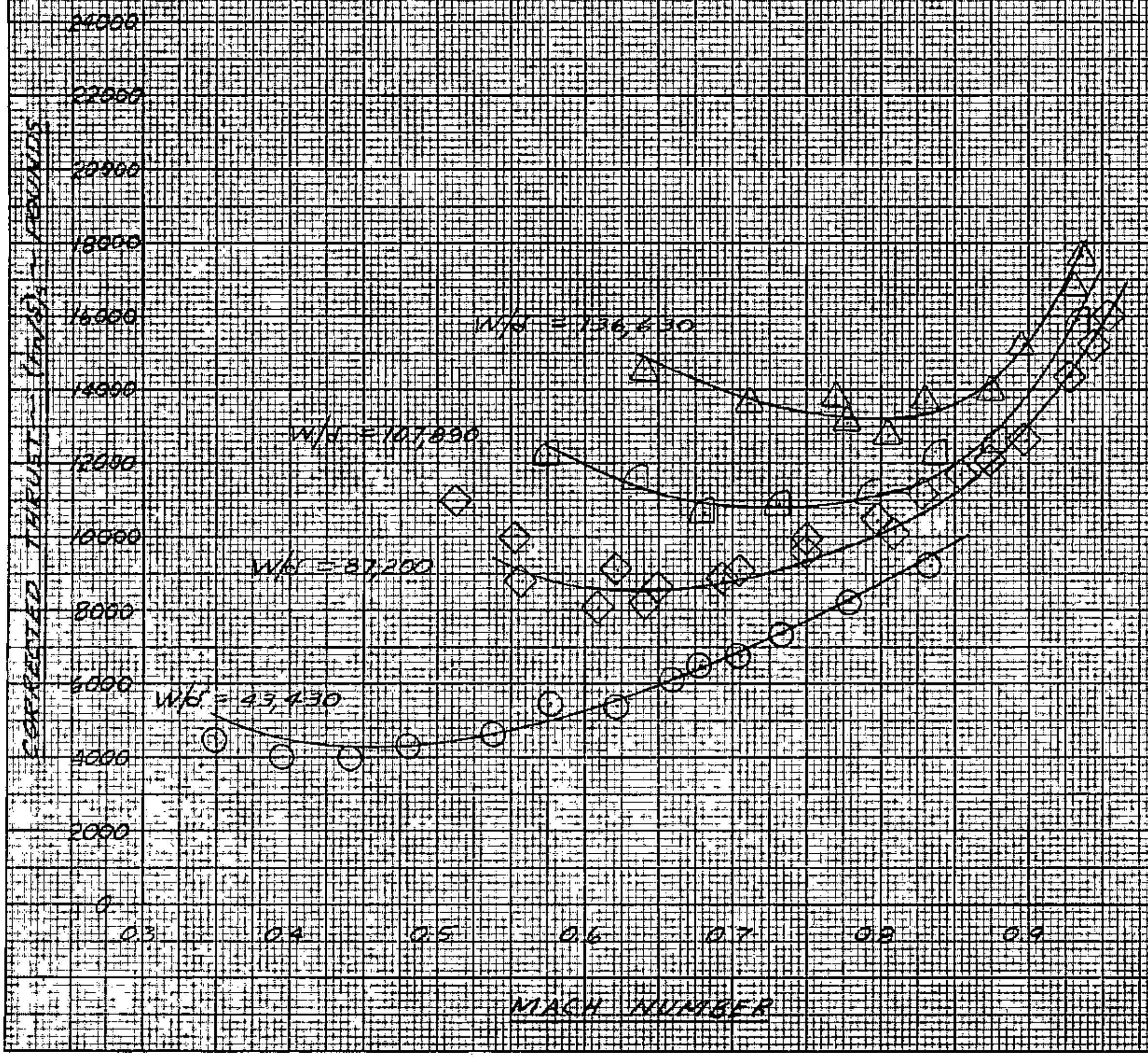
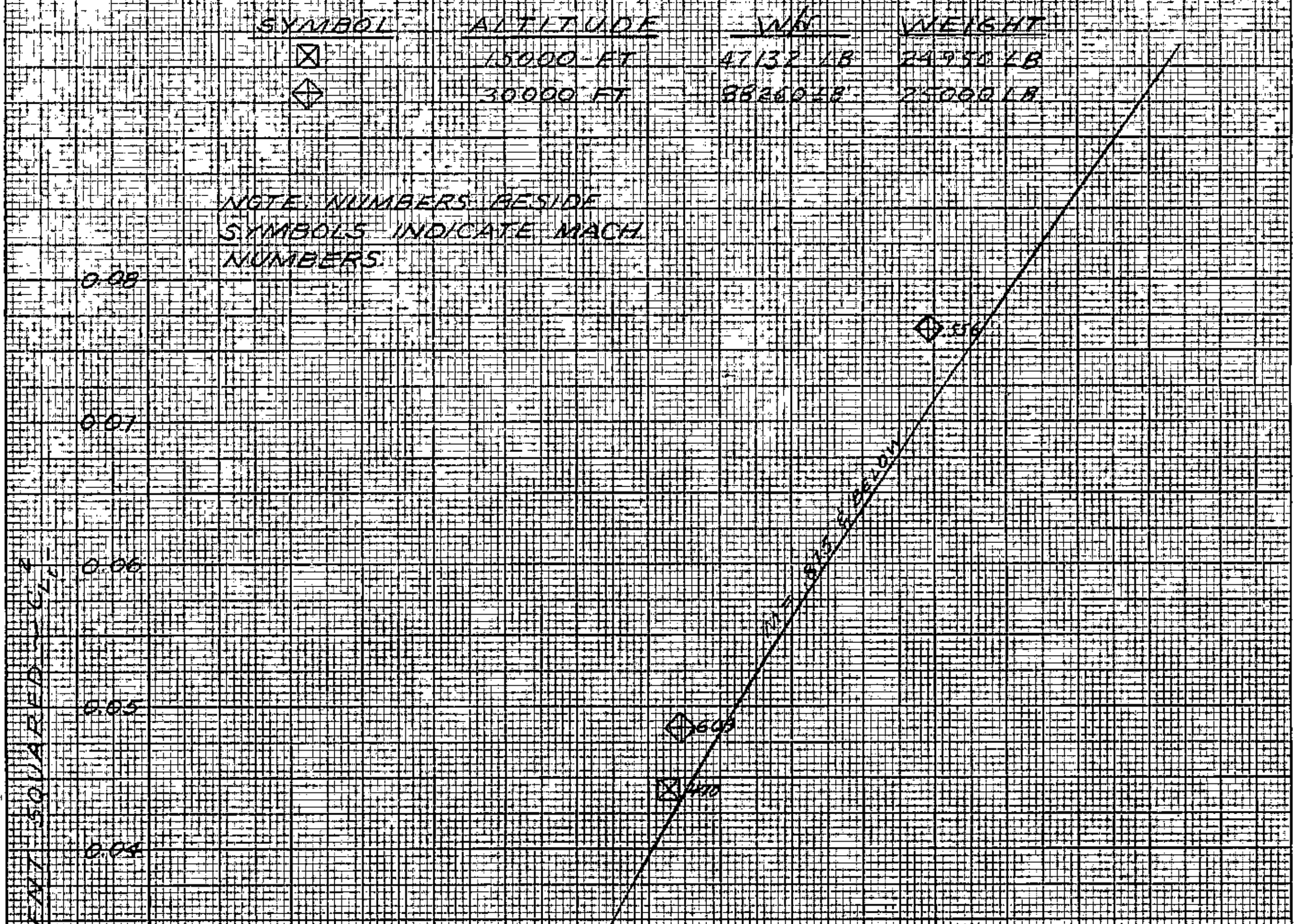


FIGURE NO 24
LEVEL FLIGHT PERFORMANCE
DRAG POLAR
F-102A USAF S/N 53-1791
NO EXTERNAL STORES



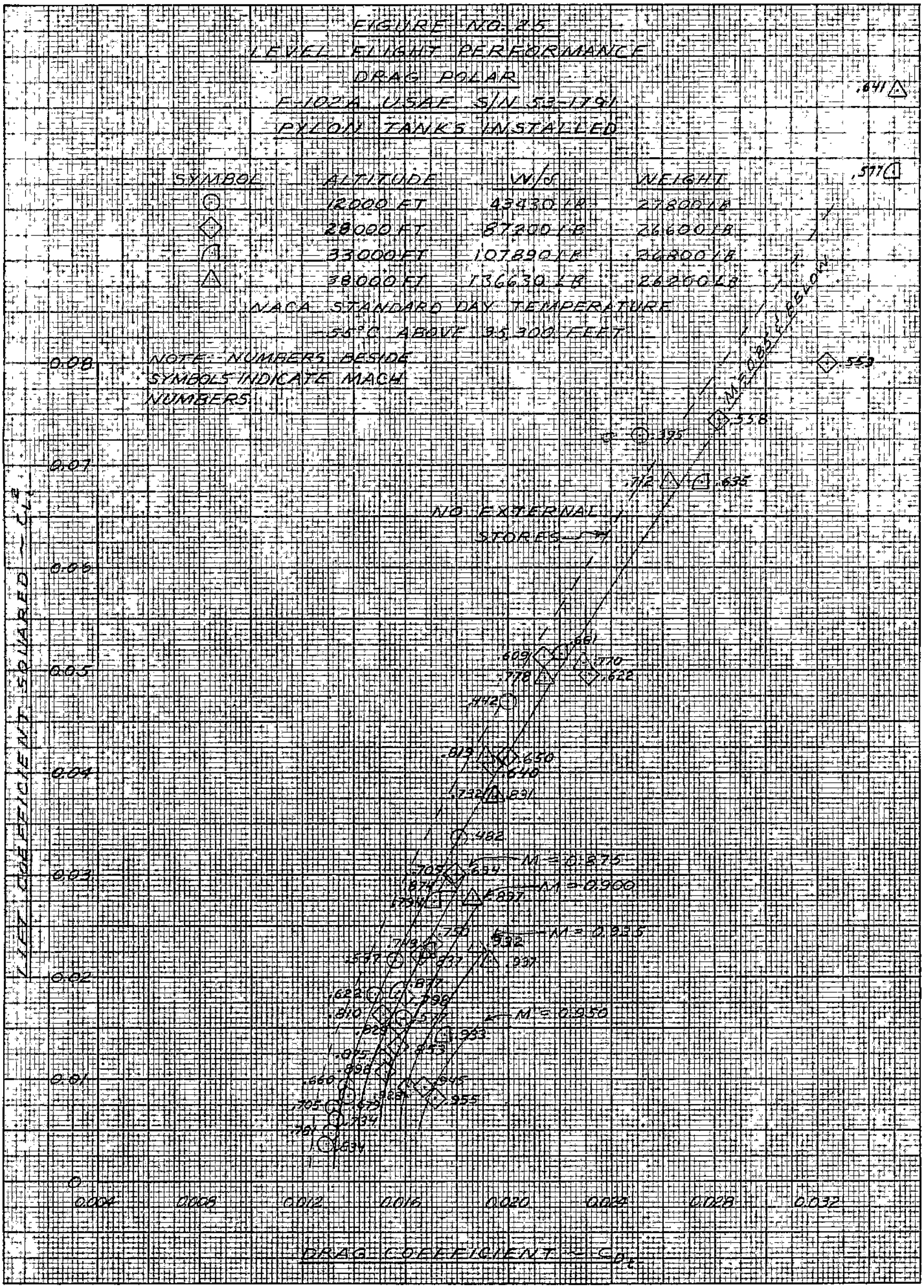
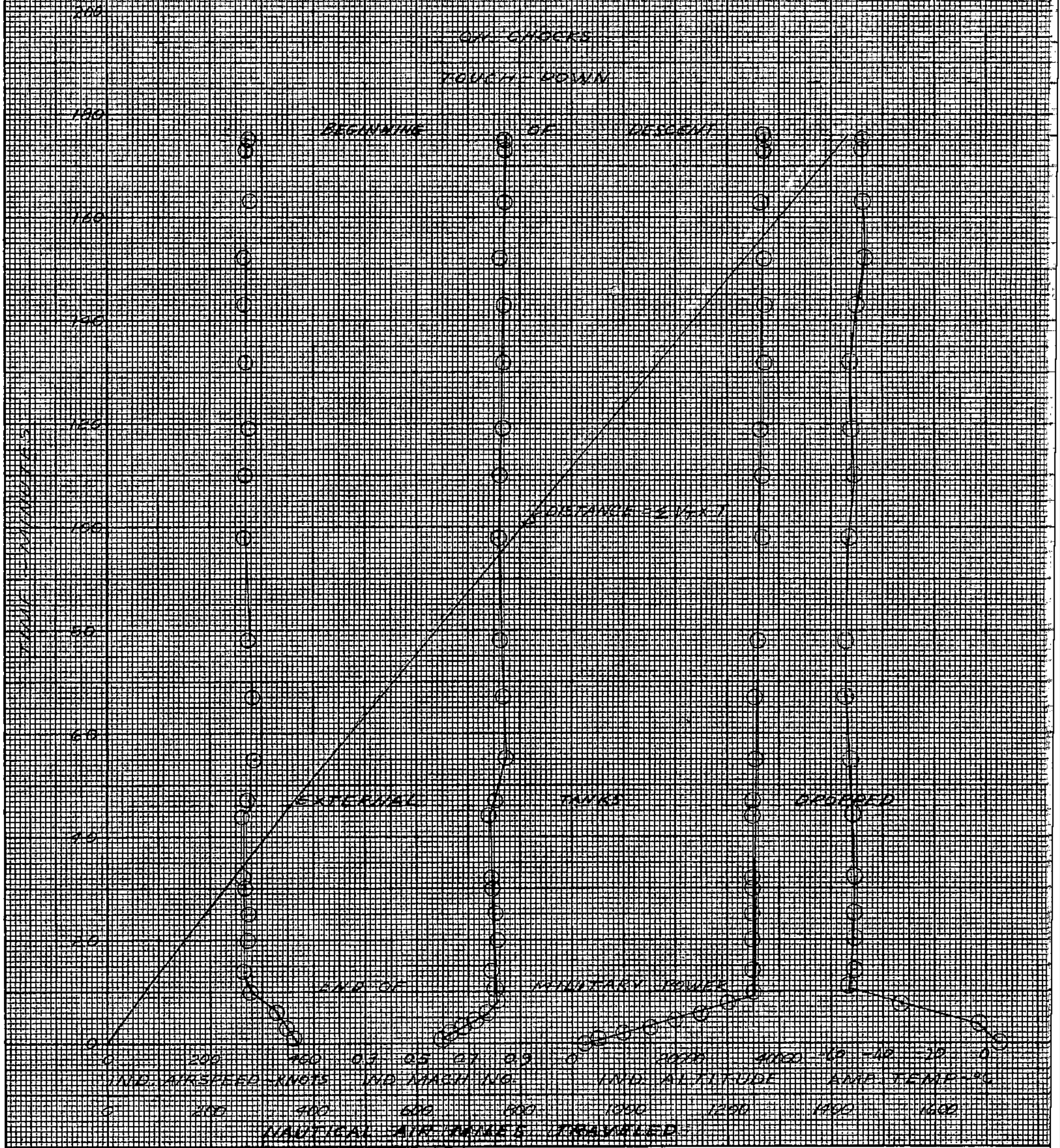


FIGURE NO. 25
LEVEL FLIGHT PERFORMANCE
DRAG POLAR
F-102A USAF S/N 53-1791
PYLON TANKS INSTALLED

FIGURE NO. 26
RANGE MISSION
F-102A USAF S/N 53-1791
30000 POUNDS AT ENGINE START



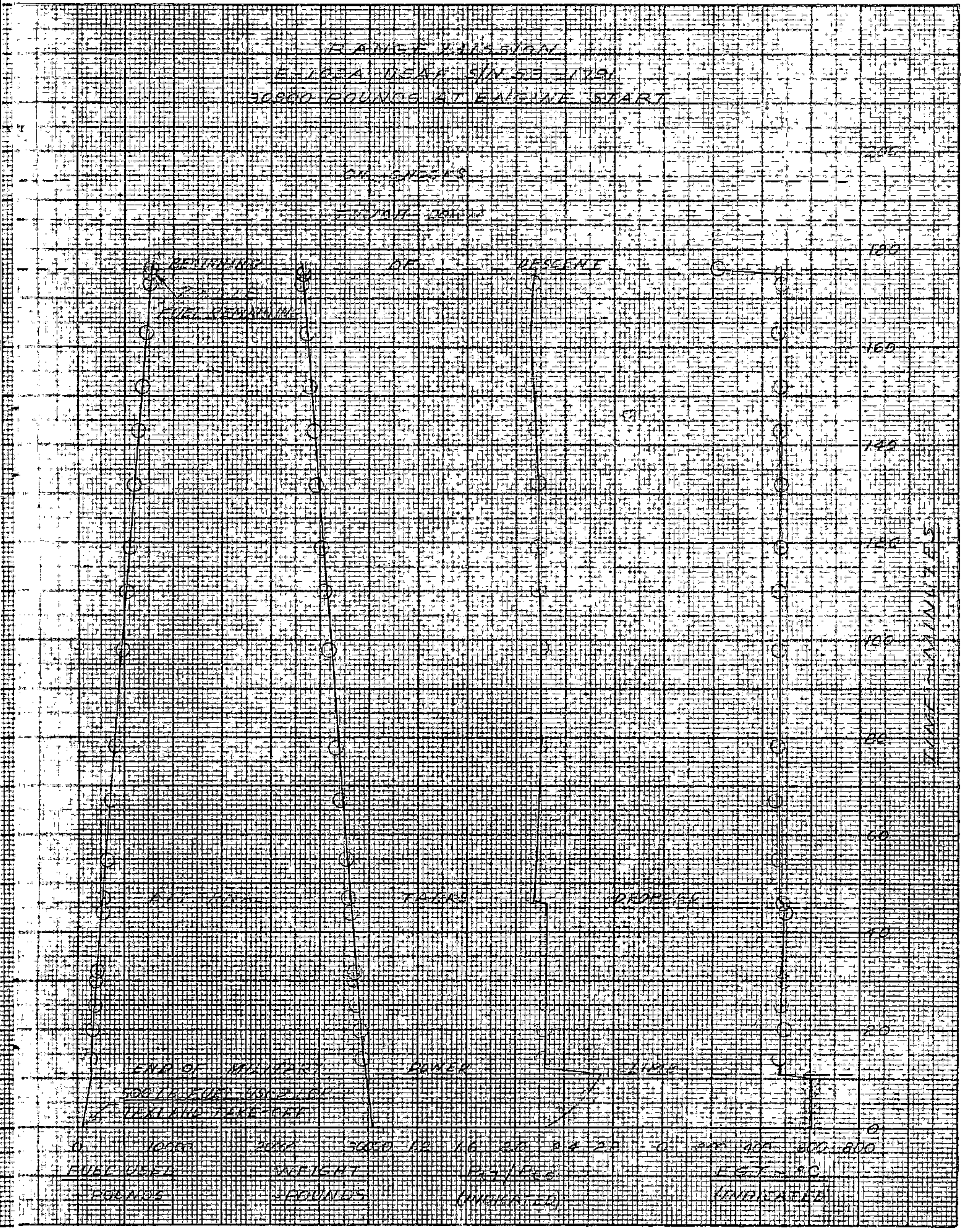
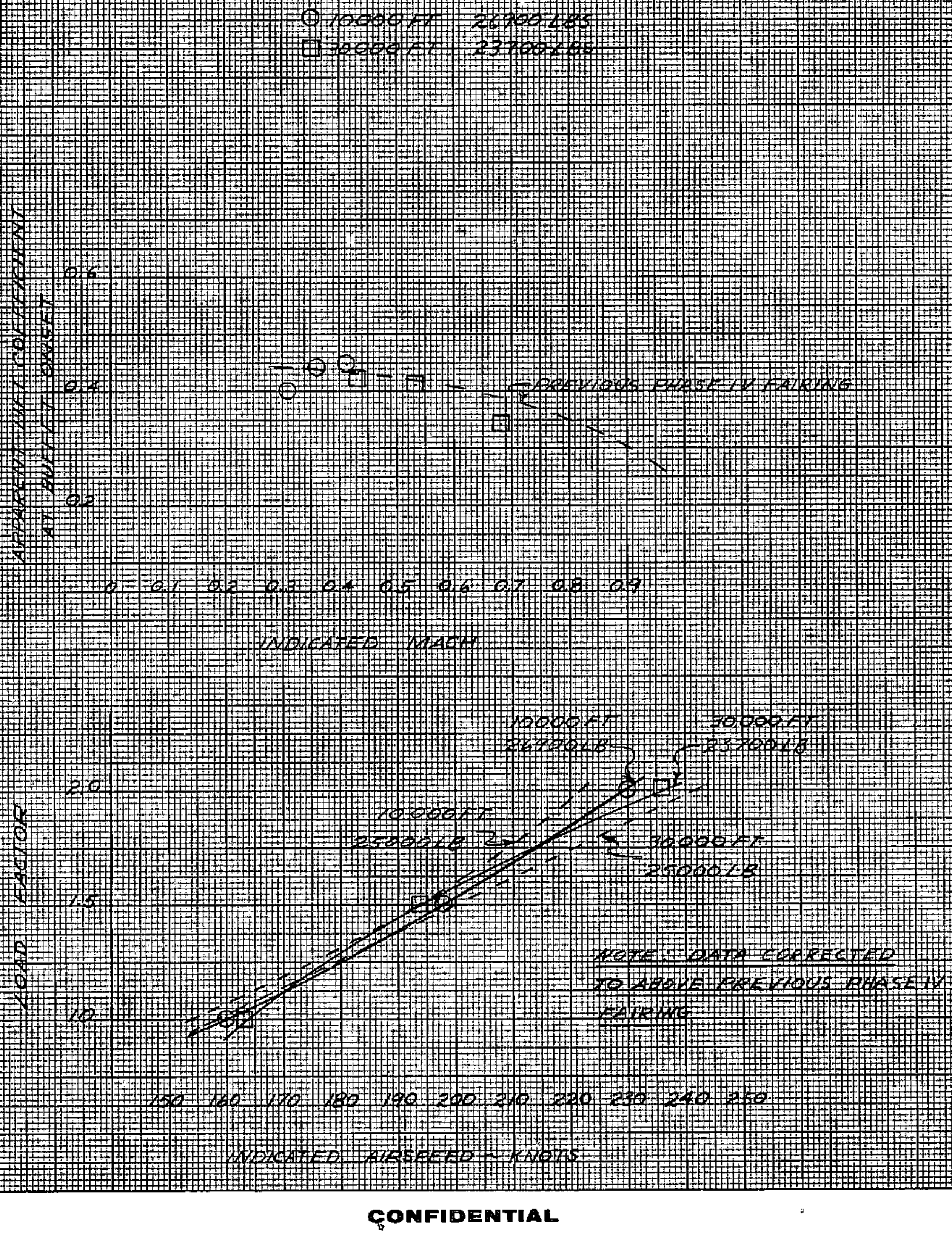
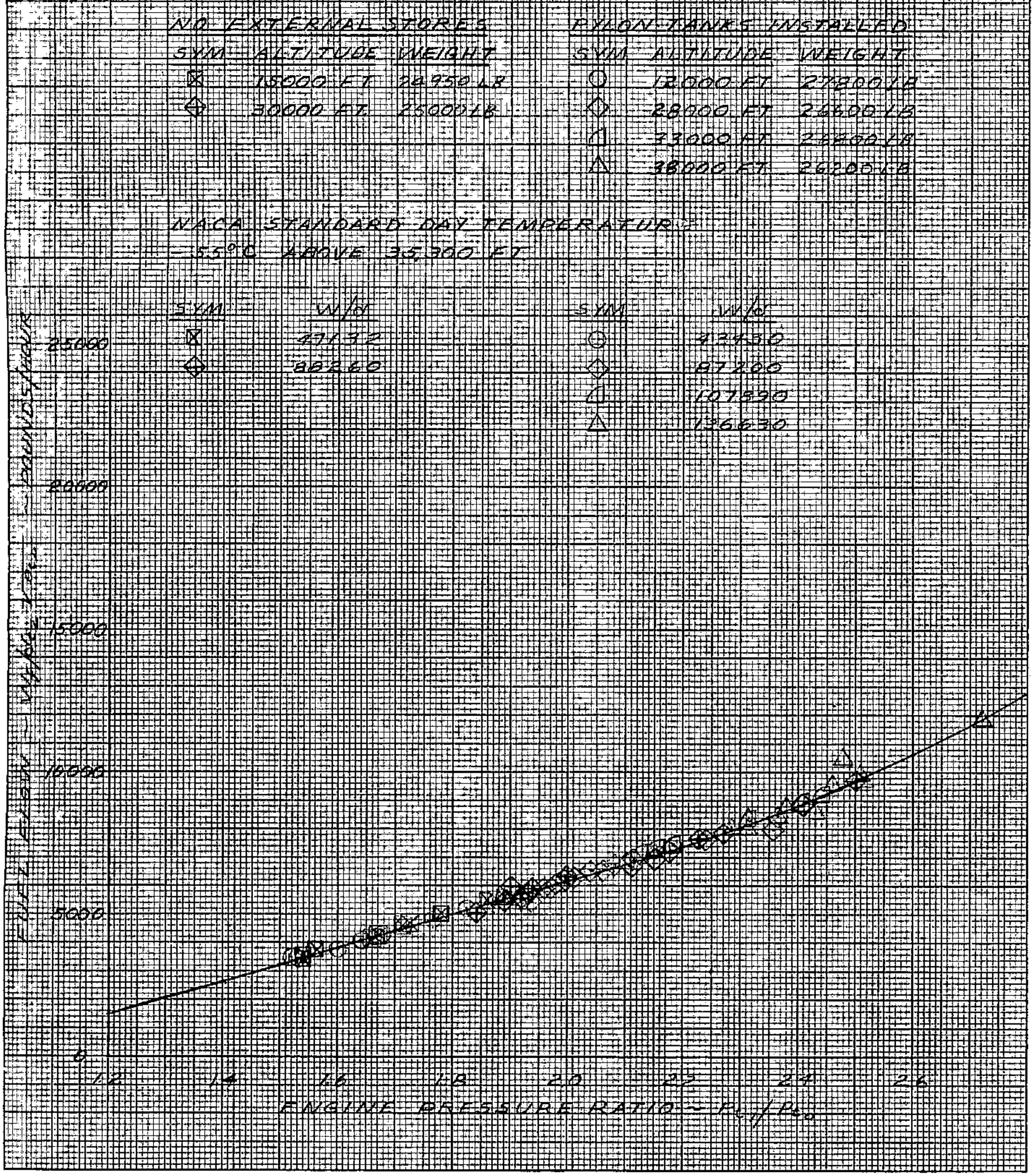

RANGE MISSION
F-102A USAF S/N 53-1791
30900 POUNDS AT ENGINE START
BEGINNING OF DESCENT
FUEL REMAINING
END OF MILITARY POWER CLIMB
5031 LB FUEL USED FOR TAXI AND TAKE-OFF
200
180
160
140
120
100
80
60
40
20
0
TIME ~ MINUTES
0
1000
2000
3000
1.2
1.6
2.0
2.4
2.8
0
200
400
600
800
FUEL USED ~ POUNDS
WEIGHT ~ POUNDS
Pt7/Pt2 (INDICATED)
EGT ~ °C (INDICATED)

FIGURE NO 27
BUFFET BOUNDARIES
F-102A USAF S/N 53-1791
PYLON TANKS INSTALLED



NOTE: DATA CORRECTED TO ABOVE PREVIOUS PHASE IV FAIRING

FIGURE NO. 28
LEVEL FLIGHT PERFORMANCE
FUEL CONSUMPTION
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039



NO EXTERNAL STORES

SYM	ALTITUDE	WEIGHT
☒	15000 FT	24950 LB
◈	30000 FT	25000 LB

PYLON TANKS INSTALLED

SYM	ALTITUDE	WEIGHT
○	12000 FT	27800 LB
◇	28000 FT	26600 LB
⌂	33000 FT	26400 LB
△	38000 FT	26200 LB

NACA STANDARD DAY TEMPERATURE
-55°C ABOVE 35,300 FT

SYM	W/δ
☒	47132
◈	88260

SYM	W/δ
○	42430
◇	87200
⌂	107590
△	136630

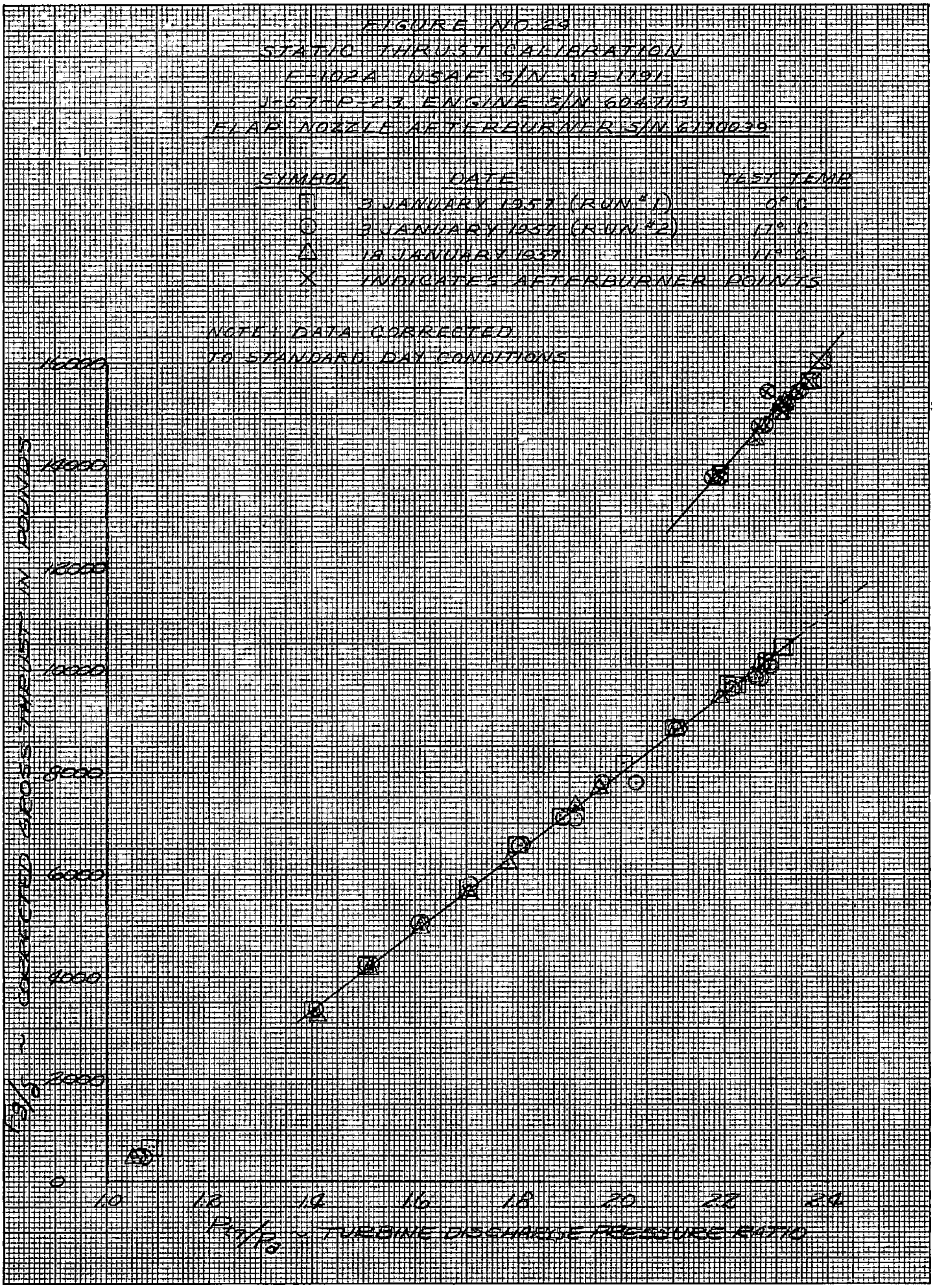


FIGURE NO. 29
STATIC THRUST CALIBRATION
F-102A USAF S/N 53-1791
J-57-P-23 ENGINE S/N 604713
FLAP NOZZLE AFTERBURNER S/N 6170039

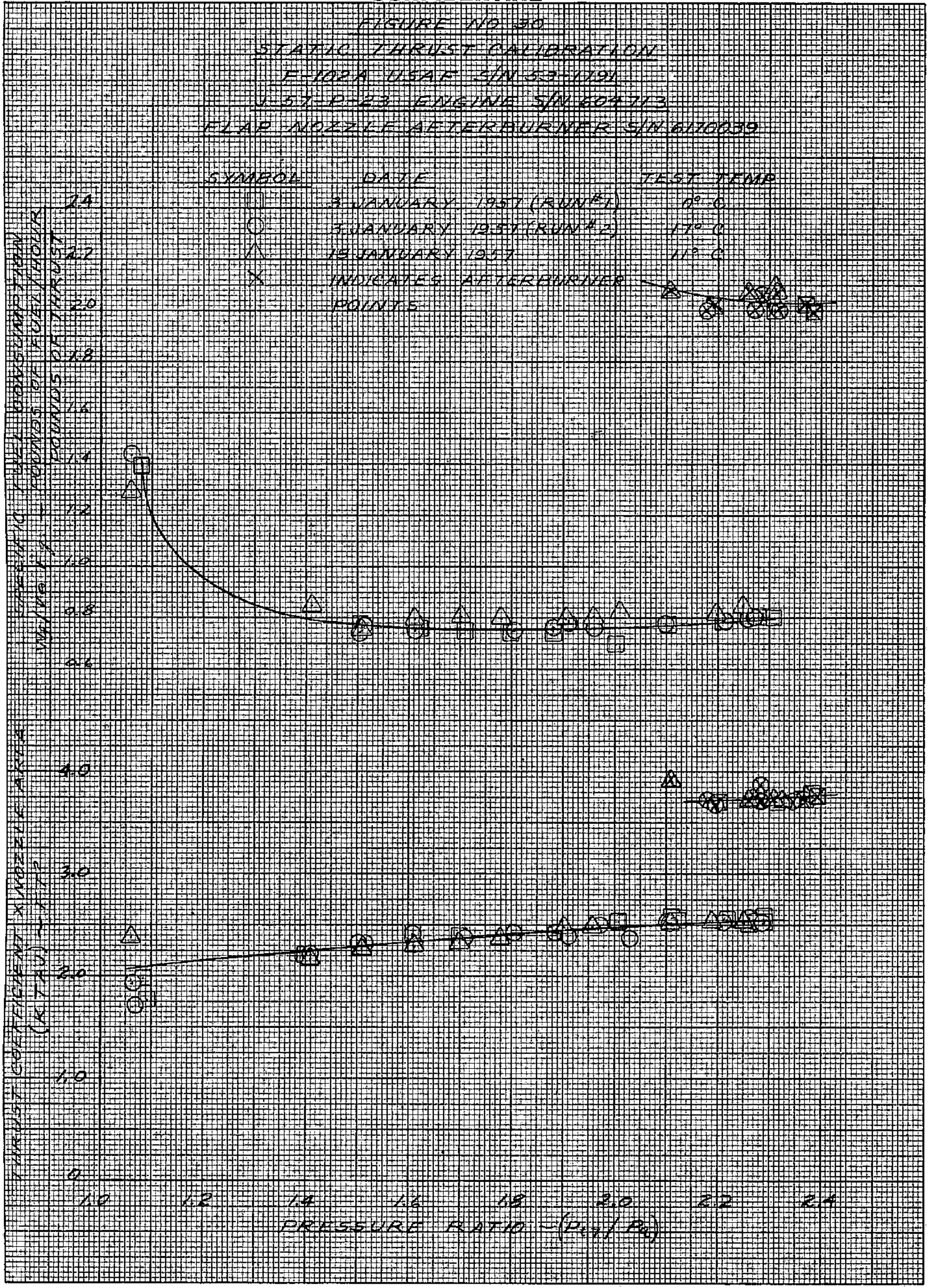


FIGURE NO. 30

STATIC THRUST CALIBRATION

F-102A USAF S/N 53-1791

J-57-P-23 ENGINE S/N 604713

FLAP NOZZLE AFTERBURNER S/N 6170039

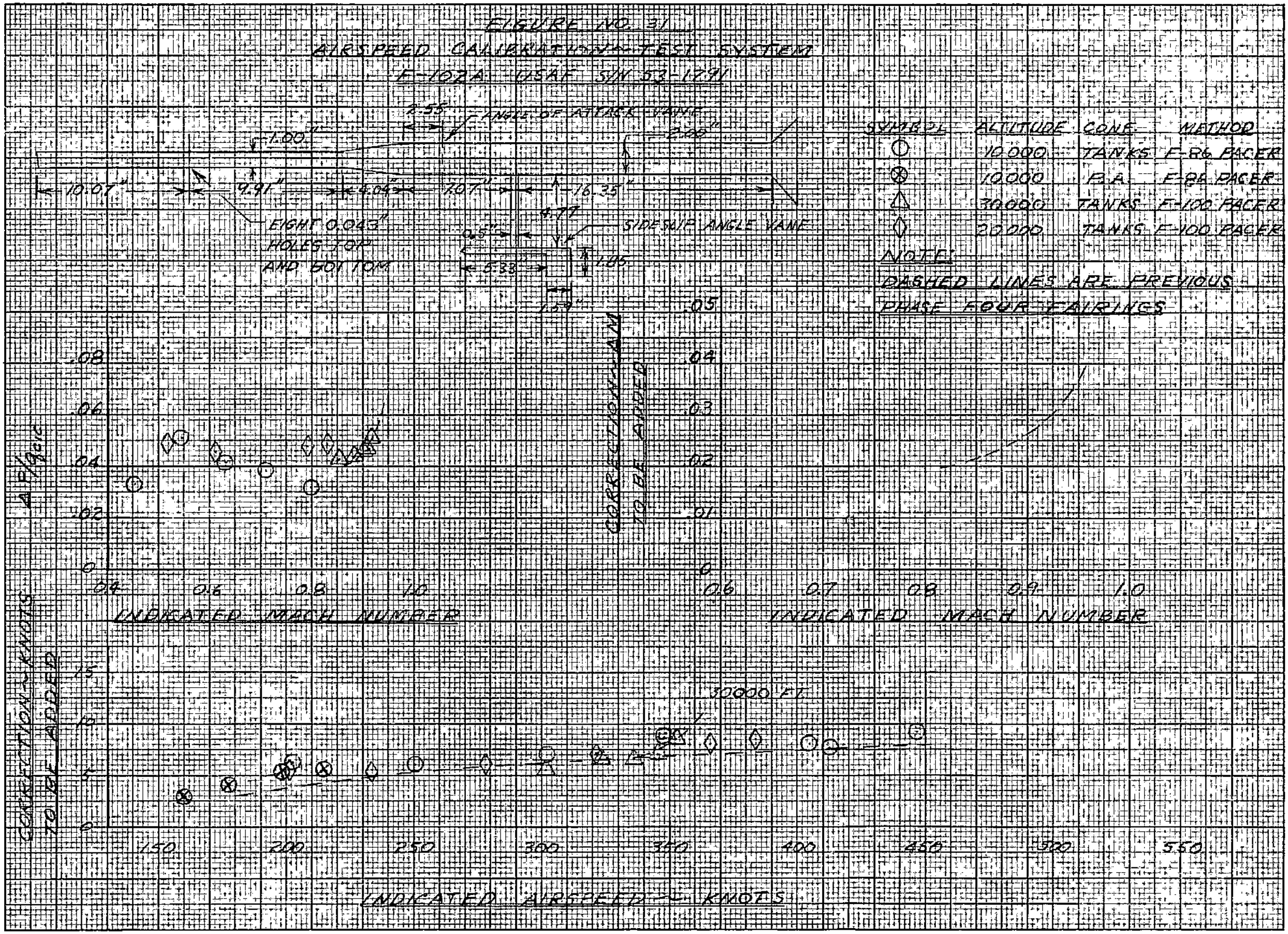


FIGURE NO. 31
AIRSPEED CALIBRATION~TEST SYSTEM
F-102A USAF S/N 53-1791

APPENDIX II

general aircraft information

■ aircraft dimensions

Wing

Area	695.05 sq ft.
Span	38 ft 1.6 in.
MAC	23 ft 9.1 in.
Aspect ratio	2.189
Airfoil section	NACA 0004-65 (Modified)
Incidence	0°
Dihedral	0°
Leading edge sweepback	60°
Trailing edge sweep forward	5°
Camber	7.5%C

Vertical Tail

Area	68.33 sq ft.
Span	8.67 ft.
MAC	10.513 ft.
Aspect ratio	1.10
Airfoil	NACA 0004-65 (Modified)
Leading edge sweepback	60°
Trailing edge sweepback	5°

Fuselage

Length	68 ft 3.3 in.

Elevons

Area	68 sq ft.
Span	12 ft 10.75 in.
Deflection	+34°, −19°

Rudder

Area	7.43 sq ft.
Deflection	±25°

FIN

Area	60.90 sq ft.

flight limitations

(with pylon tanks installed)

Engine

Maximum power	1 min. on ground 15 min. in flight
Military power	5 min. on ground 30 min. in flight
Exhaust gas temperature limit	630°C below 35,000 feet 660°C above 35,000 feet

Airspeed (with tanks installed)

Clean	450 kts. IAS .95 true Mach number
Gear extended	220 kts. IAS
Drag chute	160 kts. IAS
Canopy open	30 kts.
Ram air turbine	345 kts. IAS

Load Factors

Maximum symmetrical	−0.8 to 2.4g tanks full −2.15 to 4.25g tanks empty
Maximum asymmetrical	0 to 1.86g full tank 0 to 3.1g empty tank

Aircraft is restricted from inverted flight and to 3 seconds of zero or negative g flight.

Armament bay doors cannot be opened while pylon tanks are installed on the airplane.

roll limitations

Full aileron is permissible except at speeds above 270 knots IAS where maximum deflection shall be one-half throw.

rudder induced maneuvers

Excessive rudder induced maneuvers should be avoided. Not more than 144 pounds of rapidly applied rudder force should be applied.

fuel system

Fuel is transferred from the pylon tanks to the internal tanks by pressurizing the pylon tanks. The pylon tanks increase the fuel capacity about 448 gallons giving a total capacity of 1558 gallons.

power plant

A Pratt and Whitney J57-P-23 turbo jet engine with afterburner is installed in this aircraft. The engine is rated at 16,000 pounds sea level static thrust with afterburning and 9,800 pounds nonafterburning.

weight and balance

(with pylon tanks installed)

The aircraft was ballasted to simulate the combat loaded production airplane for this program. The gross weight at engine start was 30,889 pounds including the pilot, fuel and oil. The center of gravity varies with fuel consumed from 31.0 percent MAC at engine start to 25.1 percent MAC when fuel load is expended.

test instrumentation

PILOT'S PANEL

- Angle of pitch
- Angle of yaw
- Normal acceleration
- Lateral acceleration
- Afterburner nozzle position
- Rudder position
- Elevon position
- Bleed valve position
- Correlation counter

Exhaust gas temperature
RPM (N_2)
Engine pressure ratio (P_{t_7}/P_{t_0})
Engine fuel flow
Fuel quantity
Directional heading
Indicated Mach number
Indicated airspeed
Turn and bank indicator
Pressure altitude
Magnetic compass (J-2)
Rate of climb
Cabin altitude

PHOTO PANEL

Low pressure tachometer (N_1)
High pressure tachometer (N_2)
Engine fuel flow
Fuel quantity consumed
Outside air temperature
Indicated airspeed
Pressure altitude
Afterburner fuel flow
Afterburner fuel quantity consumed
Power lever position
Correlation counter
Time of day
Afterburner nozzle position
Turbine discharge pressure (P_{t_7})
Left hand bleed valve position
Time stop watch
Main gear touchdown light
Nose gear touchdown light
Turbine discharge temperature (T_{t_7})
An oscillograph was used to record fuel flow.

■ notations

C_L	lift coefficient
C_D	drag coefficient
F_g	gross thrust
F_n	net thrust
g	normal acceleration or load factor
M	Mach number
NAMPP	nautical air miles per pound of fuel
NAMT	nautical air miles traveled
N_2	high speed engine rotor speed
P_{t_7}	turbine exhaust pressure
P_{t_0}	free stream total pressure
W	weight
W_f	fuel flow
θ	ratio of ambient temperature to sea level temperature
S	ratio of ambient pressure to sea level pressure
EGT or T_{t_7}	exhaust gas temperature
V_C	calibrated airspeed
V_T	true airspeed
$K_T A_J$	thrust coefficient x nozzle area

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F-102A PHASE IV PERFORMANCE TEST (WITH TWO STANDARD CLASS II 230-GALLON PYLON TANKS). By L. F. McNamar, L. W. Davis, Captain, USAF. July 1957. 56 Pages. (Addendum to AFFTC-TR-56-23.)

Installation of two Class II 230 gallon tanks decreases the maximum power combat ceiling from 56,000 feet to 50,000 feet and the military power ceiling from 45,000 to 42,000 feet. The tank limit speed can be obtained at all altitudes below 35,000 feet. The additional fuel provided by the tanks increases the ferry range to 1435 nautical air miles, an addition of 250 miles. If the tanks are dropped when empty, the range is increased to 1685 miles.

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Jettison of the tanks is satisfactory when the landing gear is retracted, but may endanger the landing gear when it is extended.

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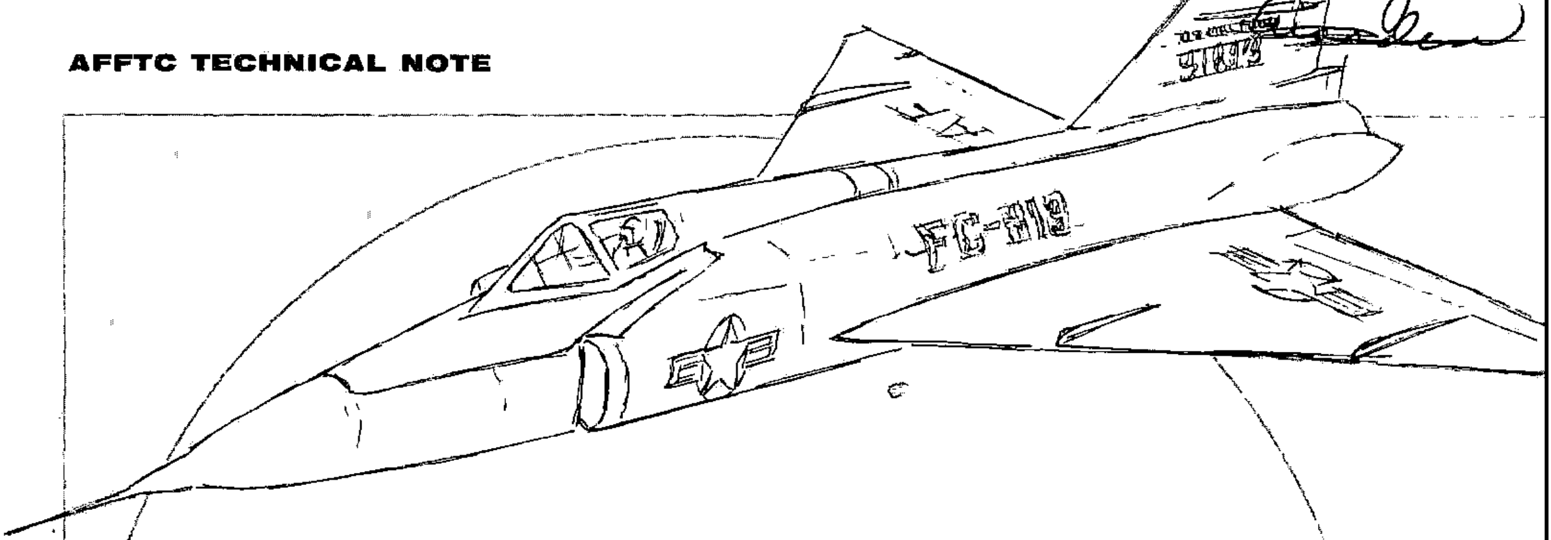

ROLL EVALUATION OF THE F-102A WITH AN ENLARGED VERTICAL STABILIZER

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MAJOR ROBERT L. STEPHENS
Project Pilot

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AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE

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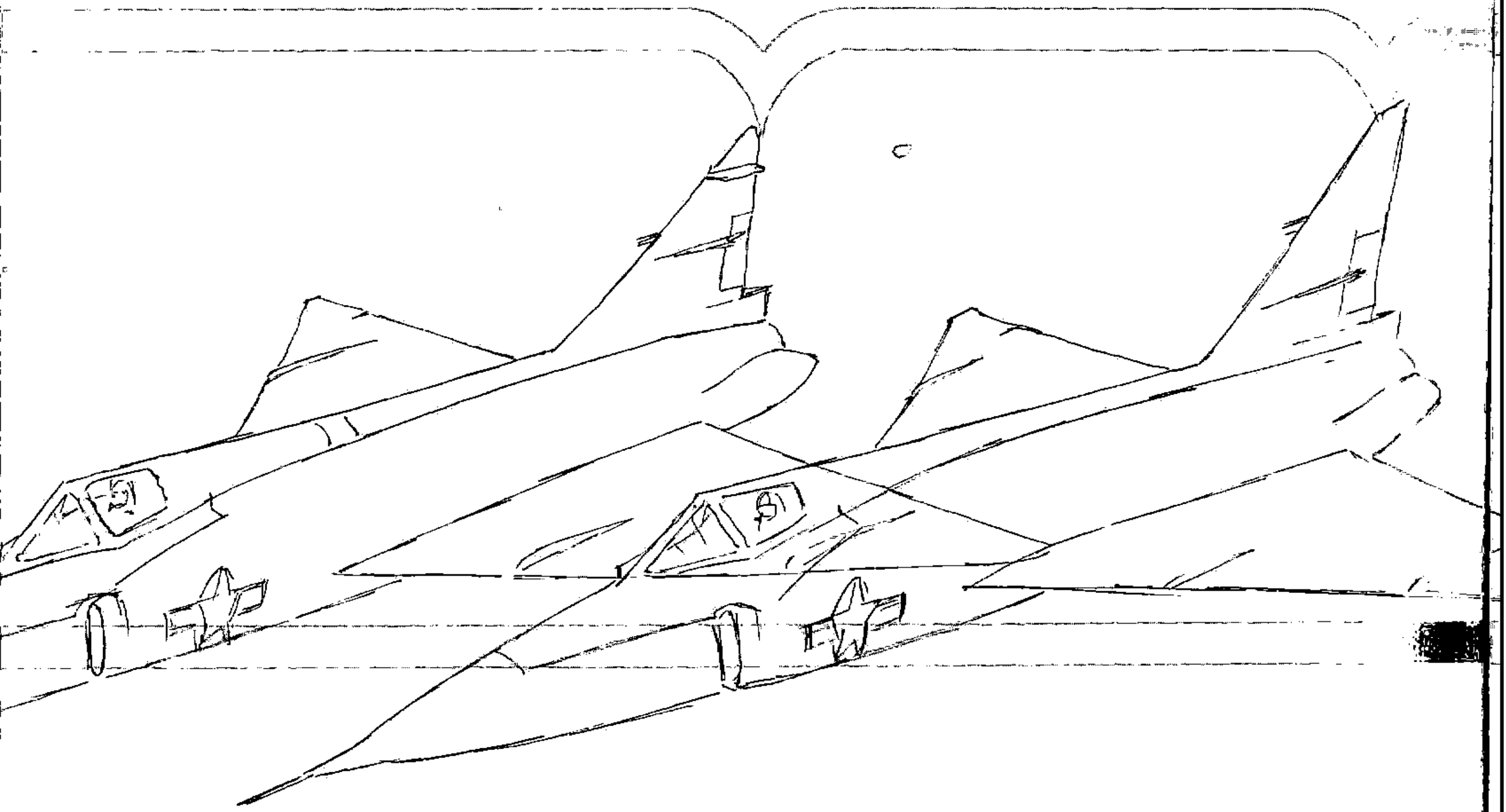

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AFFTC-TN-56-15 APRIL 1956

ROLL EVALUATION OF THE F-102A WITH AN ENLARGED VERTICAL STABILIZER

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CONFIDENTIAL

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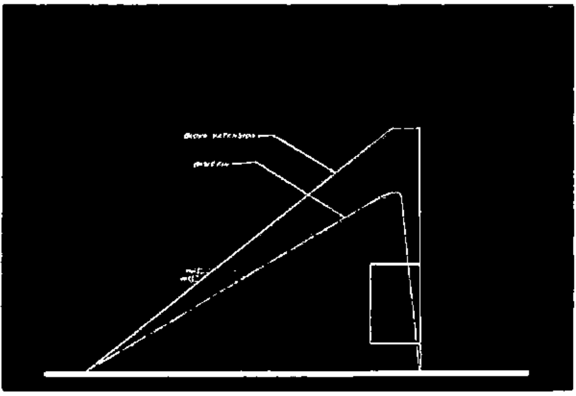

abstract

The inherent stability provided by the 40% increase in vertical tail area is sufficient to eliminate any roll restrictions because of inertial coupling dangers. These roll tests show that the test airplane has less inertial coupling effects and better handling and general flying qualities than the standard F-102A with the sideslip feed-back operating. The 100% reliability of the large tail is much more desireable than the small tail with a sideslip feed-back which does not incorporate a satisfactory safety device. The performance of the airplane has not been perceptibly changed. An aileron restriction of 10 degrees (total deflection) brought about structural considerations is adequate for operational requirements of the clean airplane. Aileron requirements for spin recovery and asymmetric load conditions, however, were not determined.

Does this mean ?
5° up x 5° down ?
?

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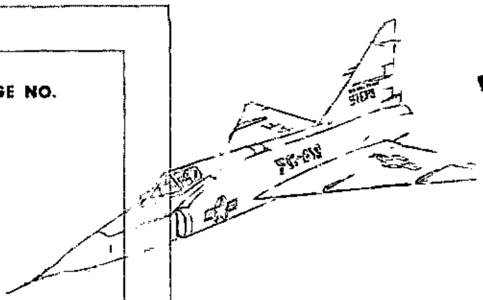

introduction

This Technical Note presents an evaluation of the rolling characteristics of the F-102A with a 40% enlarged vertical stabilizer. These flight tests were conducted under the authority of the Commander, Air Force Flight Test Center, at the request of the Air Research and Development Command. The tests were accomplished at Edwards Air Force Base from 11 January to 24 February 1956. Fifteen flights totaling 12 hours and 5 minutes were required in the F-102A airplane USAF No. 53-1813 to complete the evaluation. The evaluation pilots were: J. S. Holtoner, Brig. General, USAF; H. A. Hanes, Colonel, USAF; R. M. Baker, Colonel, USAF; F. K. Everest, Lt. Colonel, USAF; R. L. Stephens, Major, USAF; S. R. Childs, Major, USAF; R. M. White, Captain, USAF.

The F-102A airplane USAF No. 53-1813 is a standard production F-102A except that it has a "glove fit" vertical stabilizer which increased the area from 68.33 to 95.14 square feet. The test airplane does not have a production "yoke" stick but the total aileron available is 14 degrees, the same as a production airplane. The test airplane was powered by a J57-P-41 trimmed-up to simulate a J57-P-23 engine. All other F-102A aircraft have J57-P-23 engines installed. The engine start gross weight of the airplane was 27,640 pounds with a c g of 29.6% MAC. The enlarged vertical stabilizer is quoted as increasing the airplane gross weight by 166 pounds. The control system is a standard F-102A control system which includes a yaw damper, and a turn coordinator. The pitch damper was installed but inoperative during these tests. Telemetering facilities were used to obtain airplane response and control position data and for ground monitoring during the test flights. Performance data was obtained by photopanel.

The limitations for this evaluation described by the contractor are as follows:

ROLL LIMITATIONS:

- ■ Abrupt roll maneuvers are limited to altitudes in excess of 12,000 feet.
- ■ Roll maneuvers may be performed with maximum available aileron and to 2 to 3 degrees of critical elevator.

AIRSPEED LIMITATIONS:

- ■ Maximum airspeed, clean configuration: 450 knots IAS or 1.25 Mach number, whichever occurs first.
- ■ Maximum airspeed, gear down: 240 knots IAS.

LOAD LIMITATIONS:

- ■ Normal load factor — Symmetrical +3.5g −1.8g — Asymmetrical +3.0g −1.0g.
- ■ Lateral load factor −0.5g (This restriction may be lifted as data is obtained and analyzed).

test results

roll evaluation

Inertial coupling effects in rolls have been reduced in the F-102A airplane with the enlarged vertical tail area so that no divergence in sideslip is encountered. Several two and three turn continuous rolls were made and the data shows the sideslip angle did not diverge during these rolls. The effect of elevator deflections were checked during the rolls and the increase in sideslip angles was negligible. The maximum sideslip angle encountered during these tests was 6.5 degrees at 15,000 feet and 0.62 Mach number.

The directional stability provided by the enlarged vertical tail is better than that of the standard F-102A with sideslip feed-back operating. At 35,000 feet and 0.77 Mach the test airplane rolled 185 degrees per second with 13 degrees of aileron and maximum sideslip of 6.0 degrees. The standard F-102A under similar flight conditions, with sideslip feed-back operating, rolled 170 degrees per second with 15 degrees of aileron and maximum sideslip angles of 5 to 9 degrees.

The contractor maintains that the structural loads incurred with a "glove fit" tail approach the limit loads in rolls with full aileron deflection at low altitudes and high airspeeds. An aileron restriction of 10 degrees instead of present 14 degree (total aileron deflection) has been recommended by the contractor to prevent conditions which might result in structural failure. A 10 degree aileron limitation has little effect on the roll capabilities at low airspeeds. In the power approach configuration at 180 knots, there is little decrease in roll rate experienced with decrease in aileron deflection from 14 degrees to 10 degrees. The time required to roll through 30 degrees of bank angle is .70 seconds with 14 degrees aileron and .83 seconds with 9 degrees aileron. The roll capabilities at higher speeds are reduced with a 10 degree aileron limitation so that at 15,000 feet and 0.53 Mach number the maximum roll rate is reduced from 150 degrees per second to 110 degrees per second. At 0.81 Mach number, the maximum roll rate is reduced from 205 degrees per second to 180 degrees per second. The time to roll through 100 degrees of bank angle using 13 degrees of aileron was 0.95 seconds at 15,000 feet and 0.62 Mach number. Under the same conditions of speed and altitude, 1.05 seconds were required for 100 degrees of bank angle using 10.2 degrees of aileron. The amount of aileron obtainable decreases with increase of IAS at low altitudes so that 14 degrees of aileron is available at 180 knots IAS and only 10.5 degrees is available at 480 knots. With a limitation of 10 degrees, the maximum aileron available at high speed would undoubtedly be reduced further. The aileron requirements for spin recovery or asymmetrical load conditions are unknown.

Presently being considered by the contractor is non-linear aileron force-deflection gradient such that the gradient is essentially constant to 10° aileron deflection and jumps to a higher constant gradient for the range of 10 to 14 degree total deflection.

Rolls were accomplished at 180 knots IAS with gear down at 12,000 feet. These rolls were made through approximately 120 degrees change in bank angle (60 to 60 degrees). The yaw damper and turn coordinator were operating during these rolls. In these rolls the roll rate tends to a constant value at large aileron angles possibly because of increasing sideslip angles (approaching five degrees) and loss of aileron effectiveness. The data is presented in Figure 1, Appendix I, and the roll data is listed in Table 1 of Appendix I.

Rolls were accomplished in the cruise configuration at 15,000 feet in the speed range of 0.53 to 0.95 Mach number. Rolls were made through 120, 360 and 720 degrees change in bank angle. The 720 degree rolls were made to determine if any adverse effects occurred in continuous rolls. The roll time histories in Figures 6 through 8, Appendix I show that the sideslip angle reached a peak in approximately 1 second and did not reach a steady state response at time of recovery. In most 360 degree rolls the sideslip angle reached a peak in 1 second and a second slightly higher peak occurred during recovery. The maximum sideslip angle obtained during all of the roll tests was 6.5 degrees in a two turn roll with 175 degrees per second rate of roll at 0.62 Mach number and 15,000 feet altitude. Elevator deflections of 2.5 degrees up to 1.25 degrees down were made during the rolls. The effects on sideslip angle were not noticeable. The roll characteristics are presented in Figure 2, Appendix I. The sideslip angles are negligible in rolls using up to 10 degrees of aileron deflection and vary from 1 to 6.5 degrees with 10 to 14 degrees of aileron deflection. These rolls are presented in tabular form in Table 2, Appendix I.

Rolls were accomplished at 0.77, 0.92 and 1.12 Mach number at 35,000 feet from wings level through 360 degrees or more. Time histories of some typical rolls are presented in Figures 9 through 13, Appendix I. These rolls show the characteristic of the sideslip angle is not divergent during one to four continuous rolls. This data shows that the inherent stability of the aircraft is sufficient to be free of inertial coupling dangers. Elevator deflections of 2.5 degrees up (A/C nose up) to 1.4 degrees down were made during these tests and the data did not show any appreciable increase in sideslip angle. The roll characteristics are presented in Figures 3 through 5, and Table 3, Appendix I.

■ dynamic lateral-directional stability

The dynamic lateral-directional stability characteristics of this airplane are excellent with the yaw damper and turn coordinator operating. It requires 1.0 to 1.5 cycles for the oscillations to completely damp out. With the yaw damper off the oscillations damp out very slowly and do not meet Air Force Specifications, although the conditions encountered are not dangerous. The tests were accomplished at 35,000 feet at 0.77, 0.92 and 1.12 Mach numbers. The data is presented as time histories in Figures 14 through 16, Appendix I.

■ static directional stability

The airplane has satisfactory static directional stability. It requires right rudder and left aileron to produce a left sideslip. The static directional gradient of this airplane with the enlarged tail is about 2 degrees of rudder per 1 degree of sideslip and the rudder area is 1130 square inches. The YF-102A had a rudder area of 1383 square inches and 1 degree of rudder produced 1 degree of sideslip. The rudder area of the standard F-102A is 1070 square inches.

The static directional tests were conducted at altitudes of 12,000 and 35,000 feet with the yaw damper and turn coordinator operating. The pilot attempted to obtain full rudder deflection during the tests but was not able to because the "q" box is scheduled to limit rudder deflection with "q". The data is presented in Figures 17 and 18, Appendix I. These plots show the rudder effectiveness and the maximum rudder deflections for the speeds presented.

■ performance evaluation

The climb performance of the test aircraft is essentially the same as obtained on the Phase II YF-102A airplane. The effects of the large tail on climbs are within the limit of test accuracy. Two check climbs at maximum power were made to altitudes above 53,000 feet. The data obtained below 45,000 feet is not valid for comparison purposes because the speed schedule as flown at lower altitudes was much greater than the recommended climb speed schedule. The climb data is corrected to the same standard thrust used for the Phase II tests (AFFTC-TR-55-31). The data is presented in Figure 19, Appendix I, with the Phase II data included as dashed lines for comparison.

Maximum level flight speeds obtained at 37,000, 44,000 and 51,000 feet indicated a very slight change from those of the aircraft with a small tail. The differences, however, are within the limits of test accuracy. This data is presented in Figures 20 and 21 as drag versus Mach number and drag coefficient versus Mach number.

The engine used for these tests was a J57-P-41 trimmed-up to simulate a J57-P-23 engine, as was done in the Phase II program. The static thrust run indicates slightly less thrust at maximum and military power for static conditions than was obtained during the Phase II program. The thrust coefficient is also less than the earlier test and is presented in Figure 22, Appendix I.

conclusions

It is concluded that:

- No roll limitations are required because of inertial coupling effects.
- The roll capabilities with the increased area tail meet the roll requirements to perform operational and non-operational type maneuvers.
- The directional and roll-coupling stability of the aircraft has been improved; however, the requirement for the yaw damper and turn coördinator still exists.
- The performance of the airplane has not been appreciably affected as there is negligible change in ceiling and maximum speed.
- A restriction of 10 degrees (total) aileron deflection has been requested by the contractor to prevent structural failure of the enlarged tail. This restriction is acceptable for normal operation providing the pilot is able to obtain 14 degrees (total) aileron deflection for emergency conditions. At present the aileron requirements for spin recovery and asymmetric load conditions are unknown.

recommendations

It is recommended that:

- The increased area tail be included on all F-102A aircraft.
- This change be incorporated prior to delivery of any F-102A aircraft to using agencies.
- Spin requirements and asymmetric conditions be investigated as soon as possible to determine the aileron requirements for all flight conditions.
- An investigation be made to determine the aileron restriction with "q" and Mach number to satisfy structural limits.

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discussion

The elevons of the F-102A airplane perform a dual function by operating differentially as ailerons and in unison as elevators. Total aileron was computed by taking the algebraic summation of the elevon positions with down elevon considered negative. The direction of aileron (left or right) was chosen according to the direction it tends to make the aircraft roll. The time history plots in this Appendix have been presented as one half of the total aileron displacement. It is not the trace of a single elevon.

Elevator displacement is the average of the two elevons with up deflection tending to cause the aircraft to pitch up. The elevator deflections presented in the time history plots in this Appendix show the elevator deflections from trim and are not absolute values of elevator position.

The sideslip angle, as presented, indicates the opposite of yaw angle as measured with a yaw vane installed on the nose boom.

The climb and level flight performance data were corrected to standard conditions in the same manner as outlined in AFFTC-TR-55-31, the Phase II report on the YF-102A. The engine used for these tests (also for the YF-102A) was a J57-P-41, trimmed-up to simulate a J57-P-23 engine.

FIGURE I

AILERON ROLL CHARACTERISTICS

POWER APPROACH CONFIG.
ALTITUDE 12,000 FT.

⊙ 180 KNOTS, IAS

60 TO 60 DEGREE ROLLS

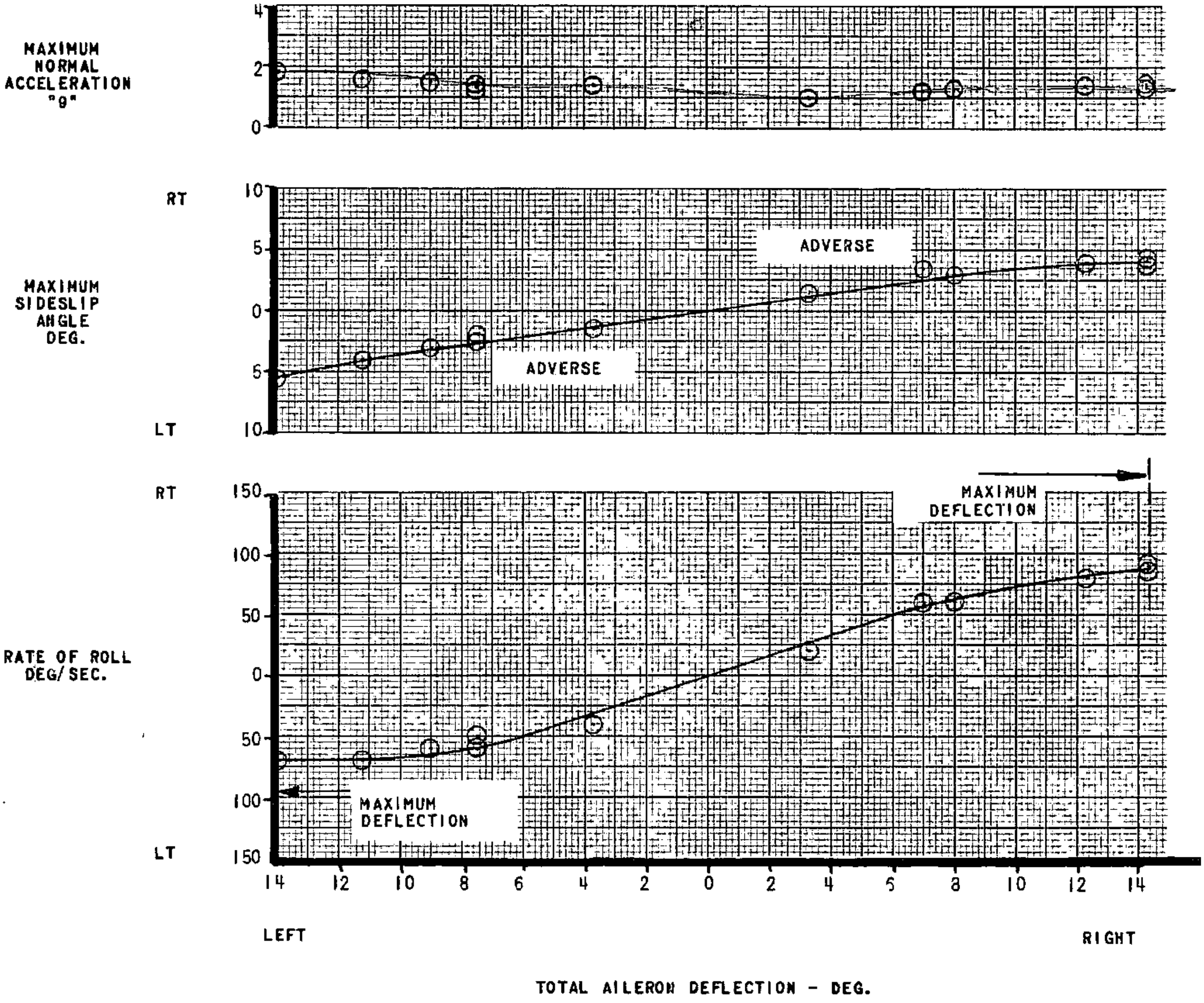


FIGURE 2

AILERON ROLL CHARACTERISTICS

CRUISE CONFIGURATION
ALTITUDE 15,000 FT.

	V_c - KNOTS	MACH
◇	492	.95
⊙	415	.81
⊡	314	.62
△	266	.53

1 TURN ROLL - DOT IN SYMBOL
2 TURN ROLL - VERTICAL LINE THRU SYMBOL

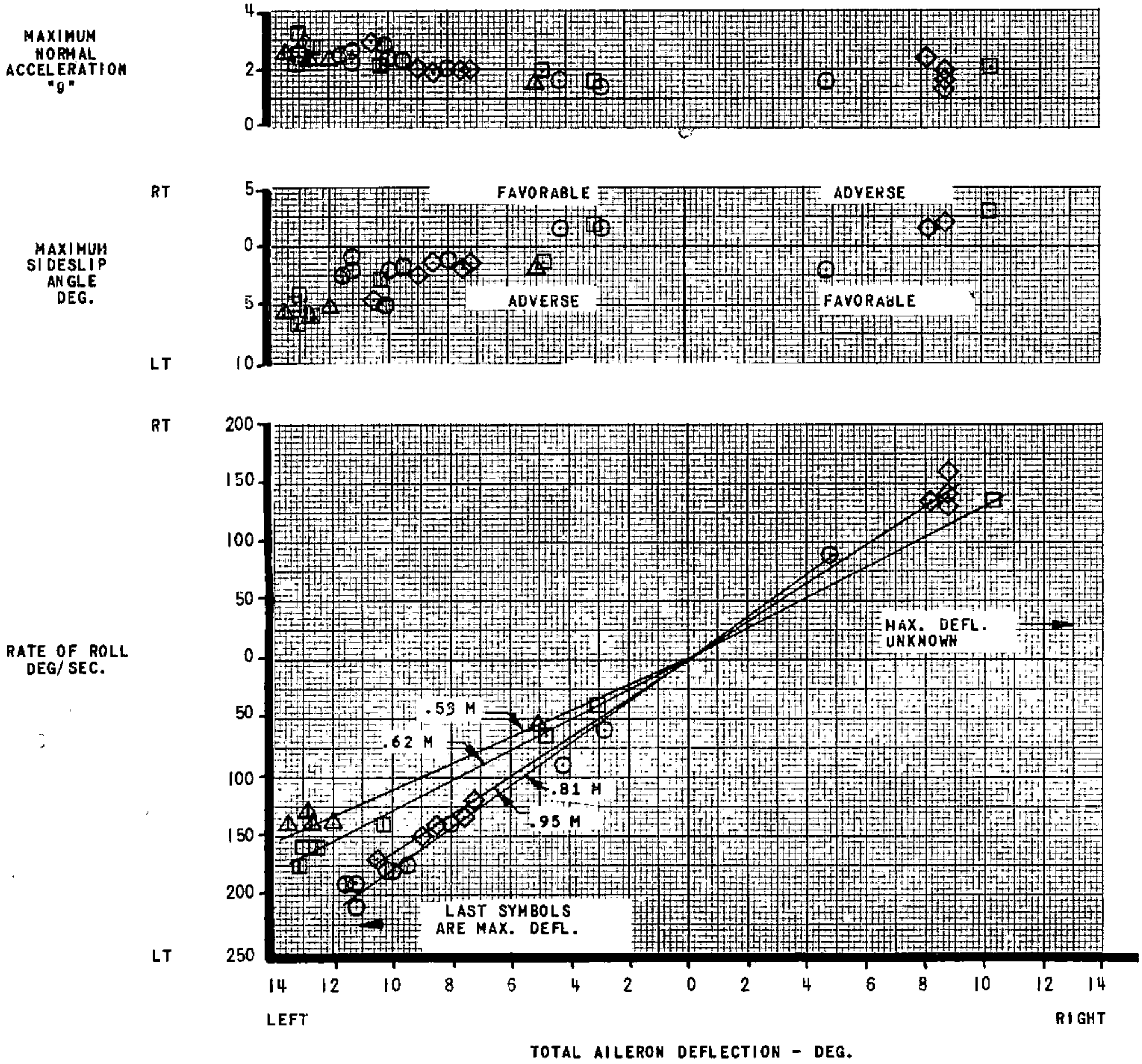


FIGURE 3 **AILERON ROLL CHARACTERISTICS**

CRUISE CONFIGURATION

⊙ 1 TURN ROLL
⦶ 2 TURN ROLL
⊕ 3 TURN ROLL
● 4 TURN ROLL

MACH NO. 77
ALTITUDE 35,000 FT.
V_c 261 KNOTS

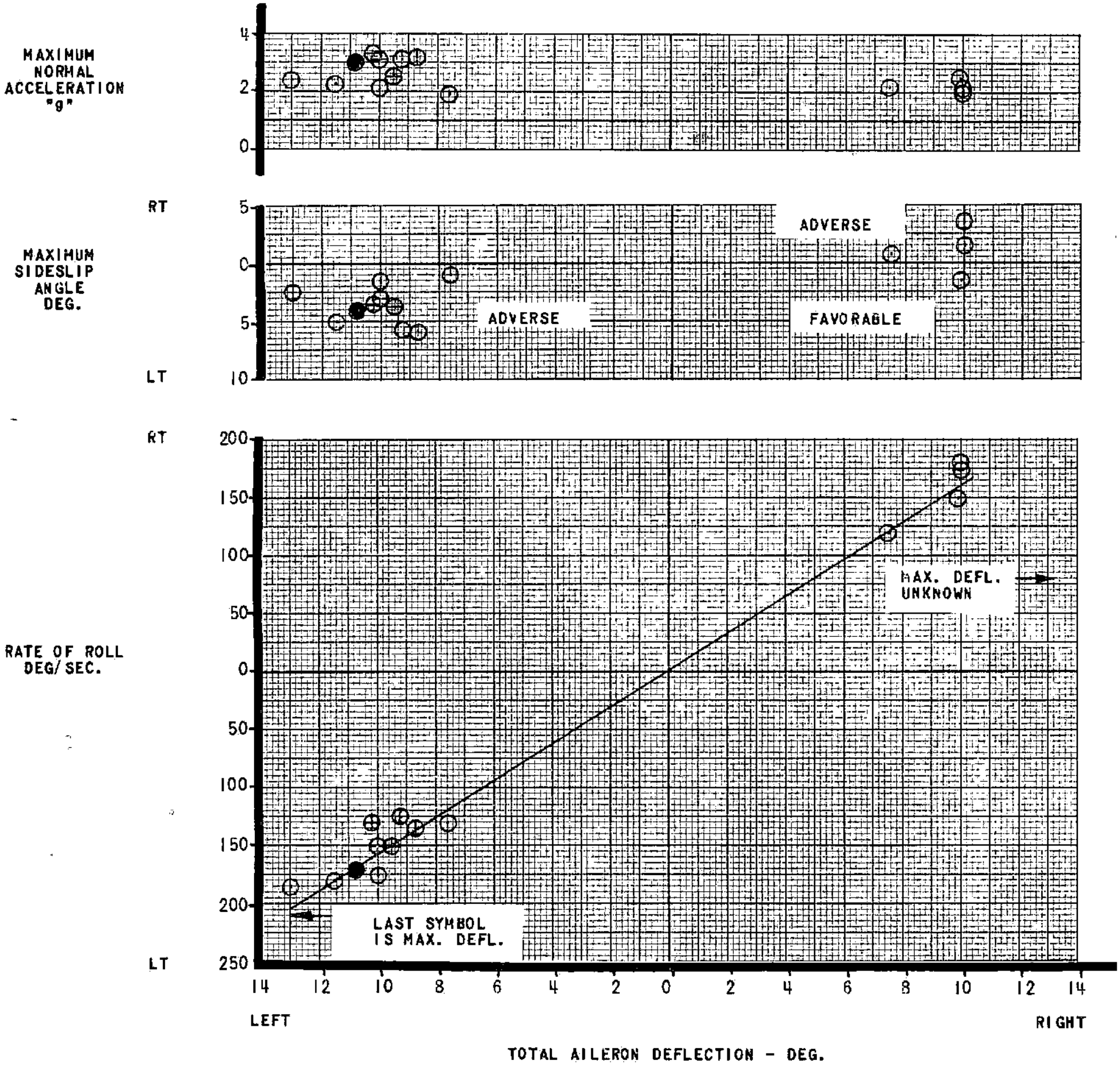


FIGURE 4

AILERON ROLL CHARACTERISTICS

CRUISE CONFIGURATION

⊙ 1 TURN ROLL
⊕ 3 TURN ROLL

MACH. NO. 92
ALTITUDE 35,000 FT.
V_c 320 KNOTS

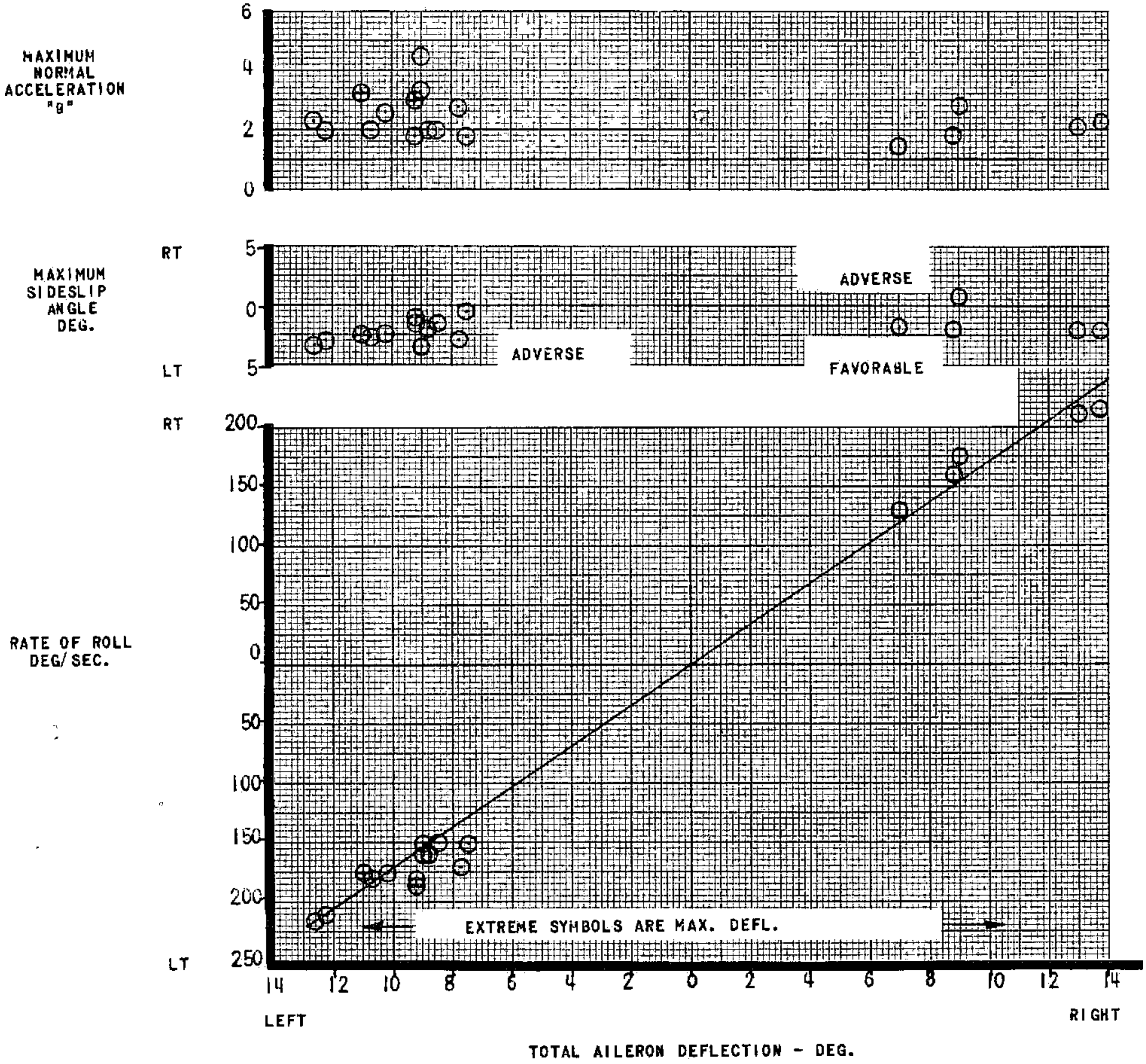


FIGURE 5

AILERON ROLL CHARACTERISTICS

COMBAT CONFIGURATION

⊙ 1 TURN ROLL

⦶ 2 TURN ROLL

MACH. NO. 1.12
ALTITUDE 35,000 FT.
V_c 400 KNOTS

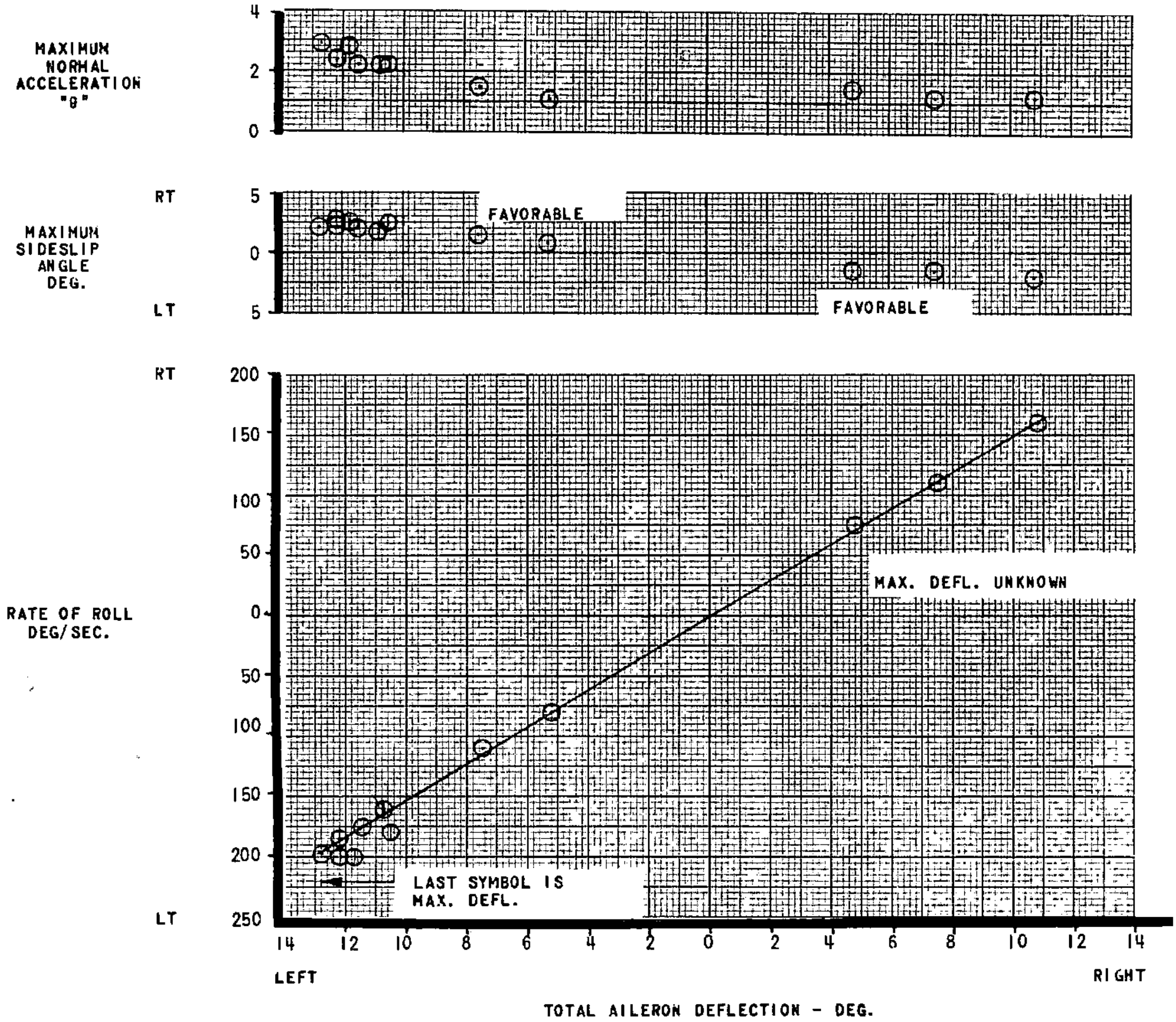


Figure No. 6
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.50
Alt 14,800 Ft.
CAS 253 Knots
Yaw Damper On

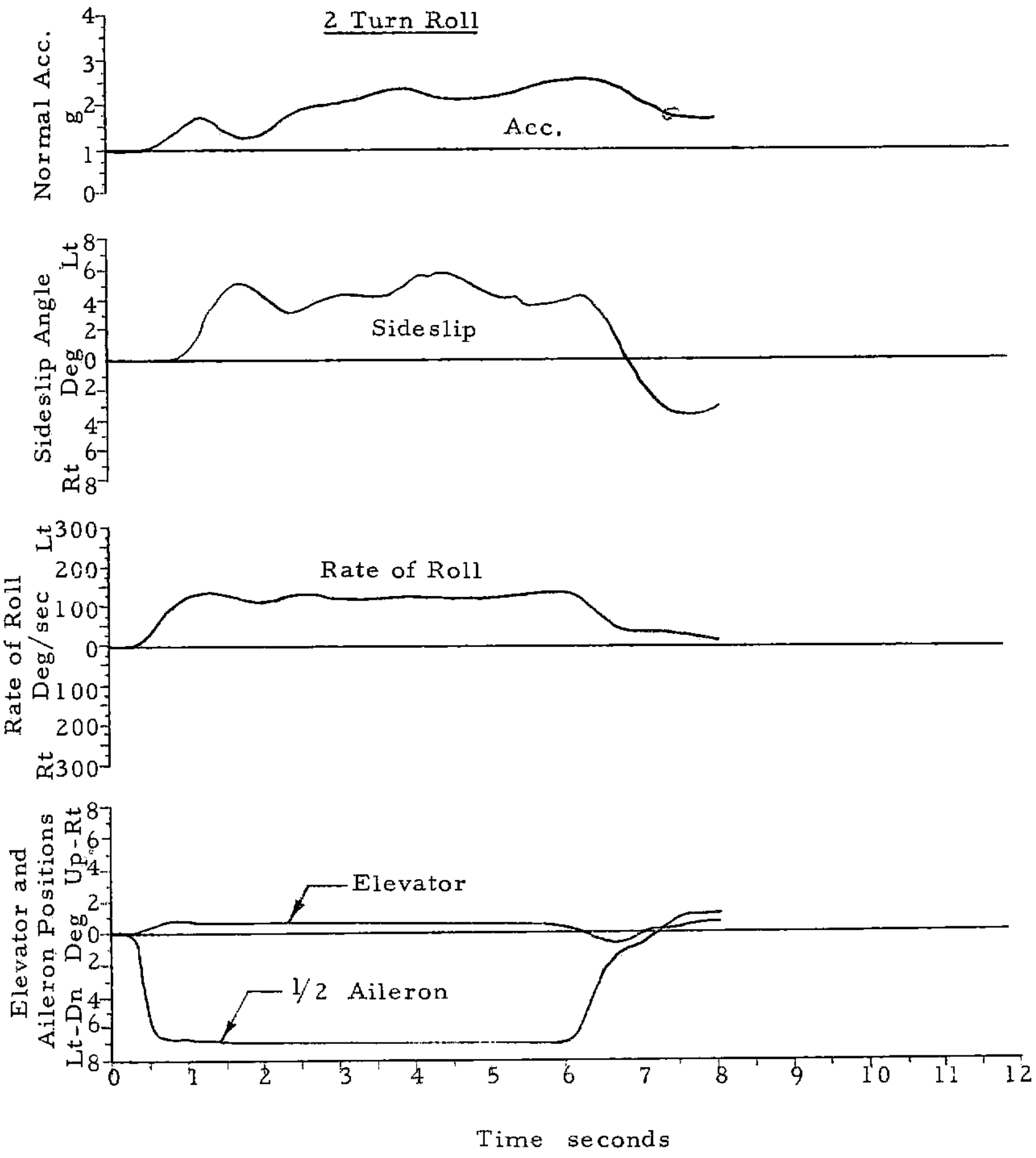


Figure No. 7
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.50
Alt 14,800 Ft.
CAS 253 Knots
Yaw Damper On

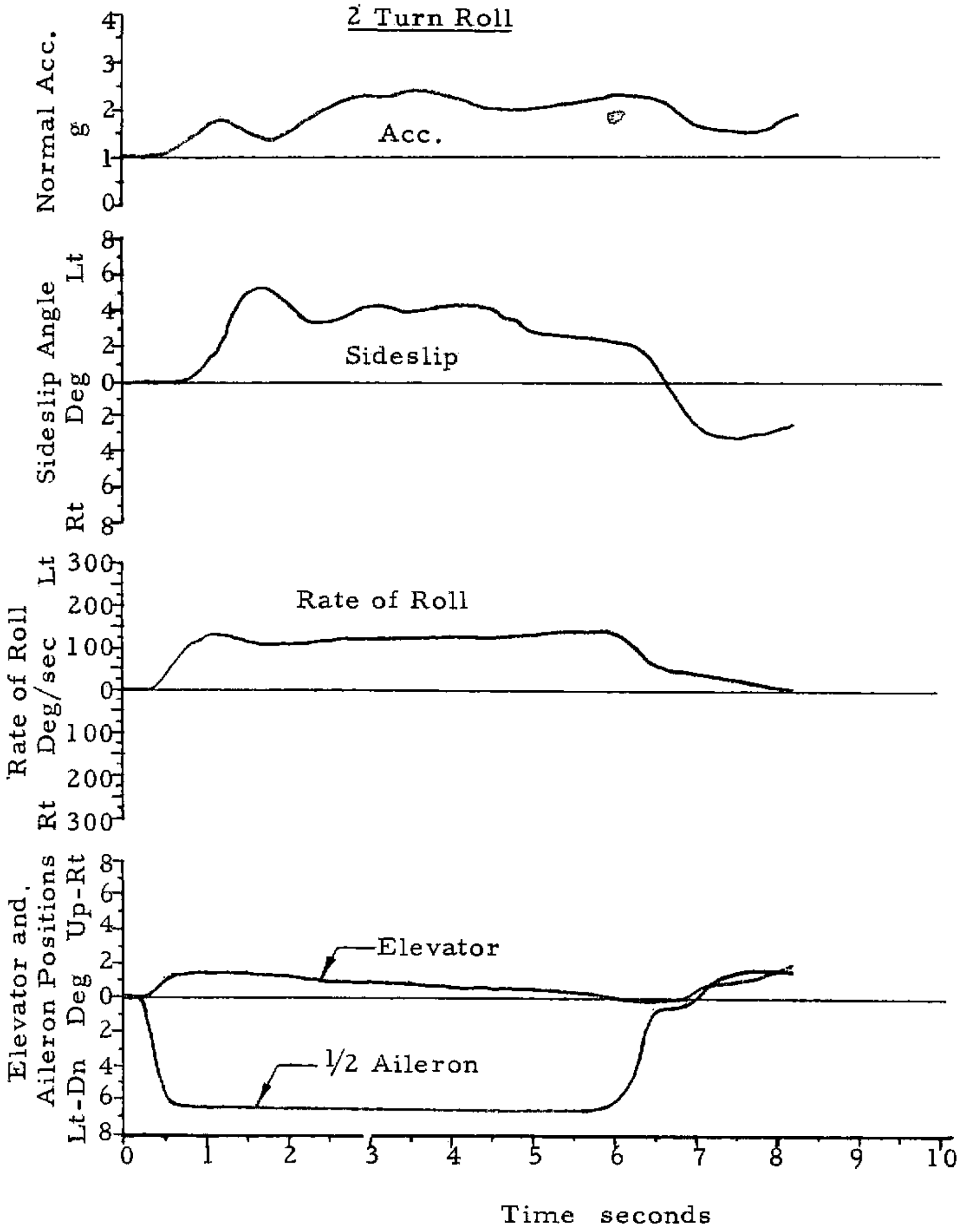


Figure No. 8
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.82
Alt 14,500 Ft.
CAS 410 Knots
Yaw Damper On

Mach No. 0.61
Alt 14,100 Ft.
CAS 309 Knots
Yaw Damper On

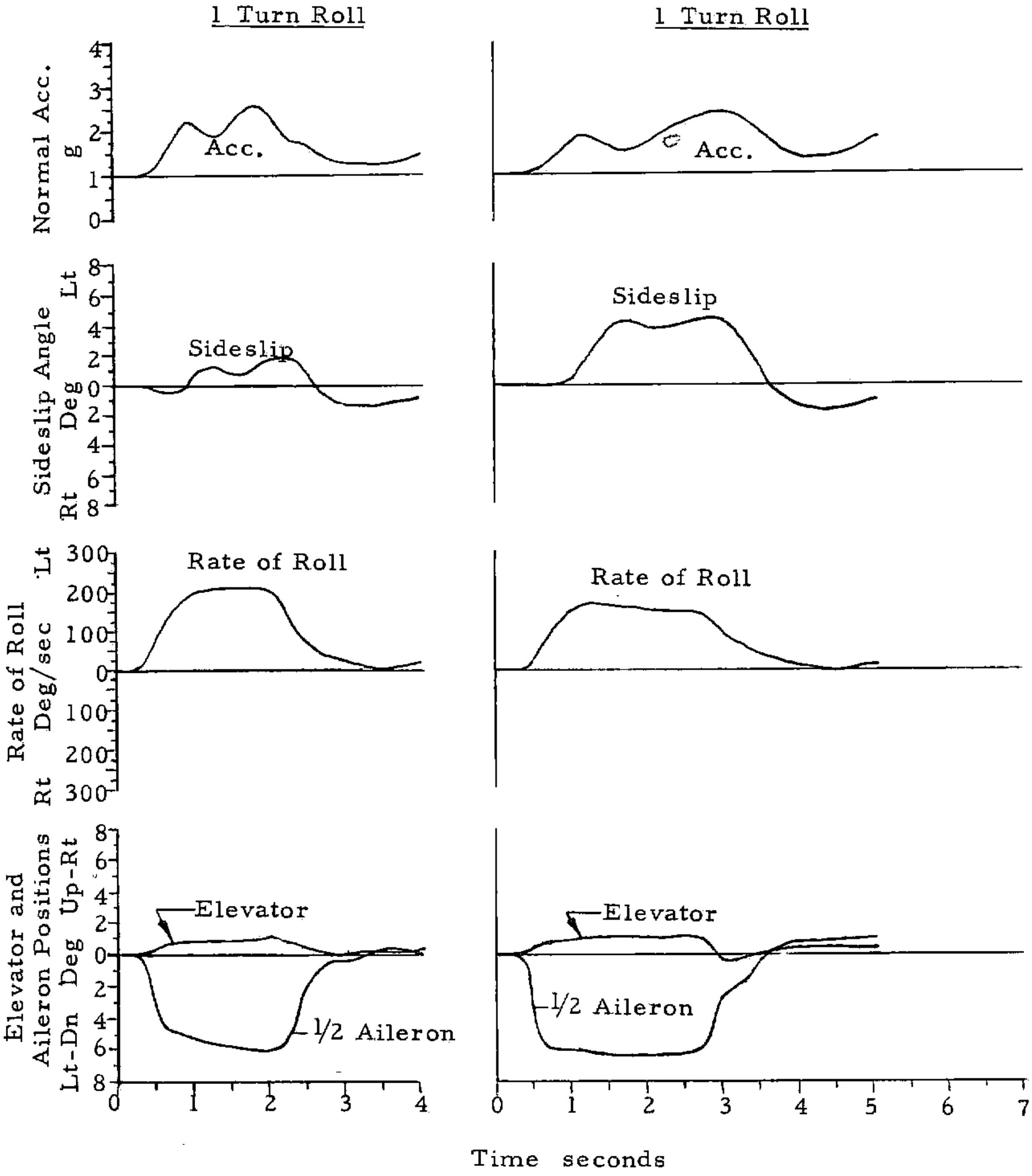


Figure No. 9
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

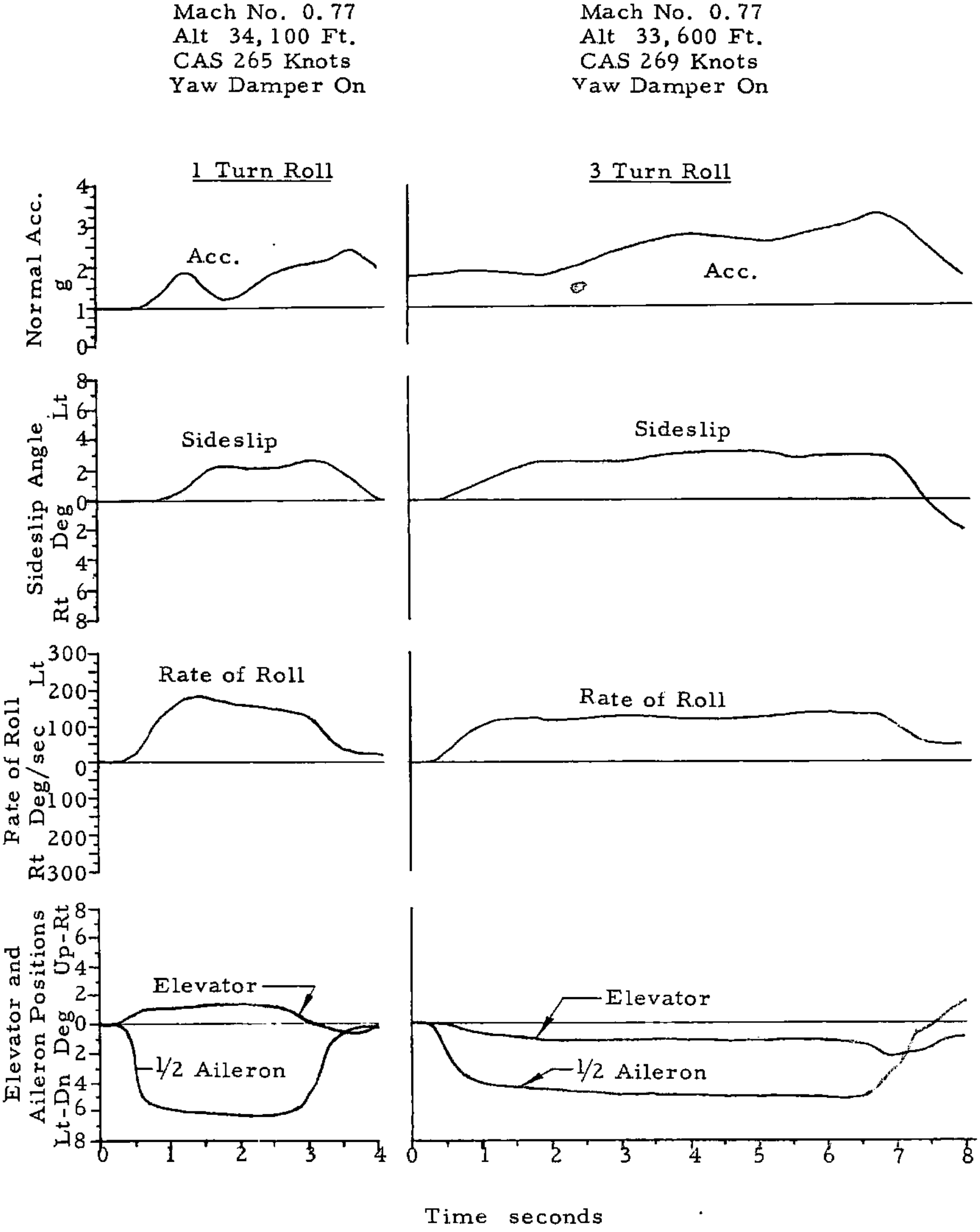


Figure No. 10
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.77
Alt 34,900 Ft.
CAS 260 Knots
Yaw Damper On

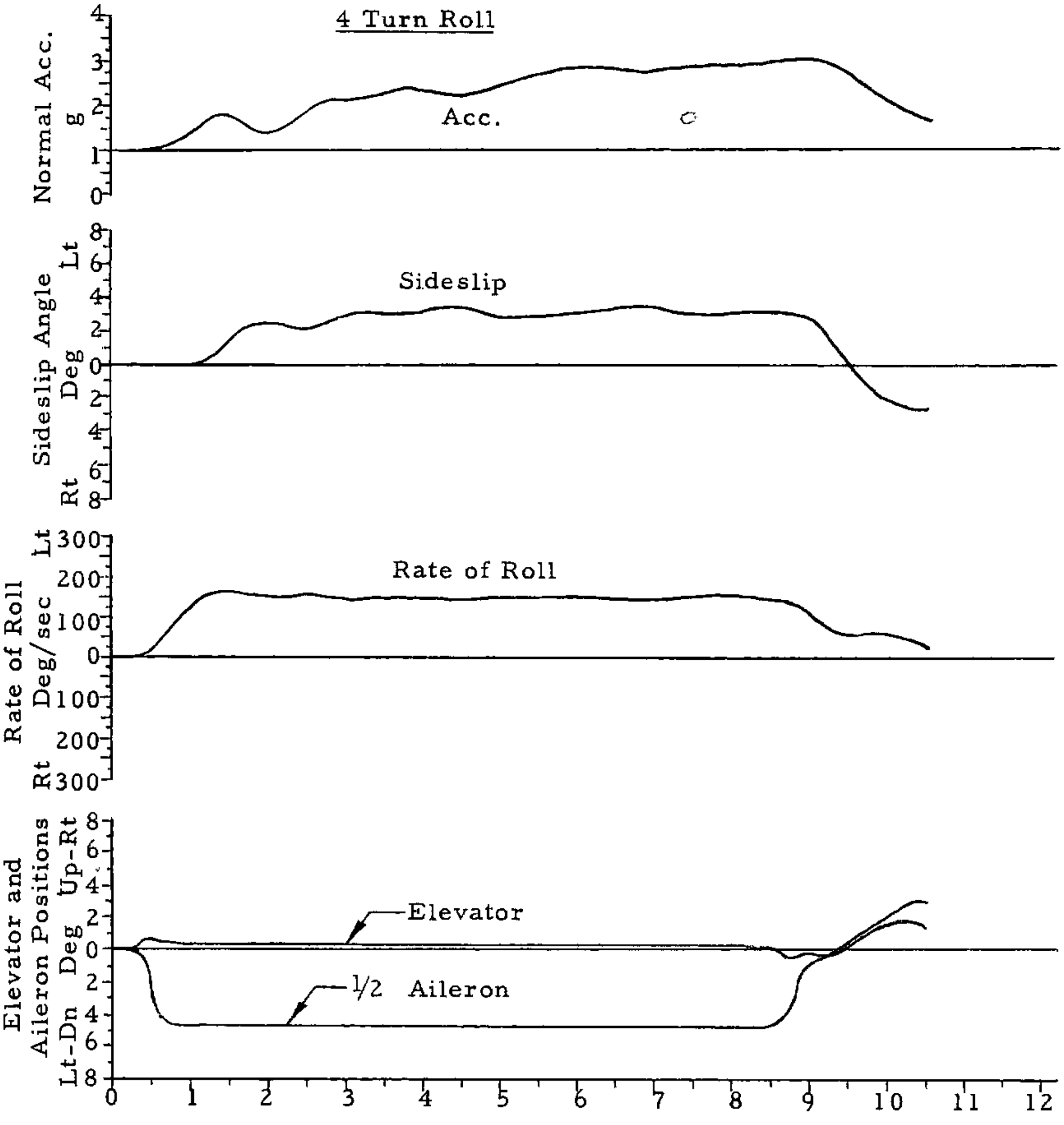


Figure No. 11
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

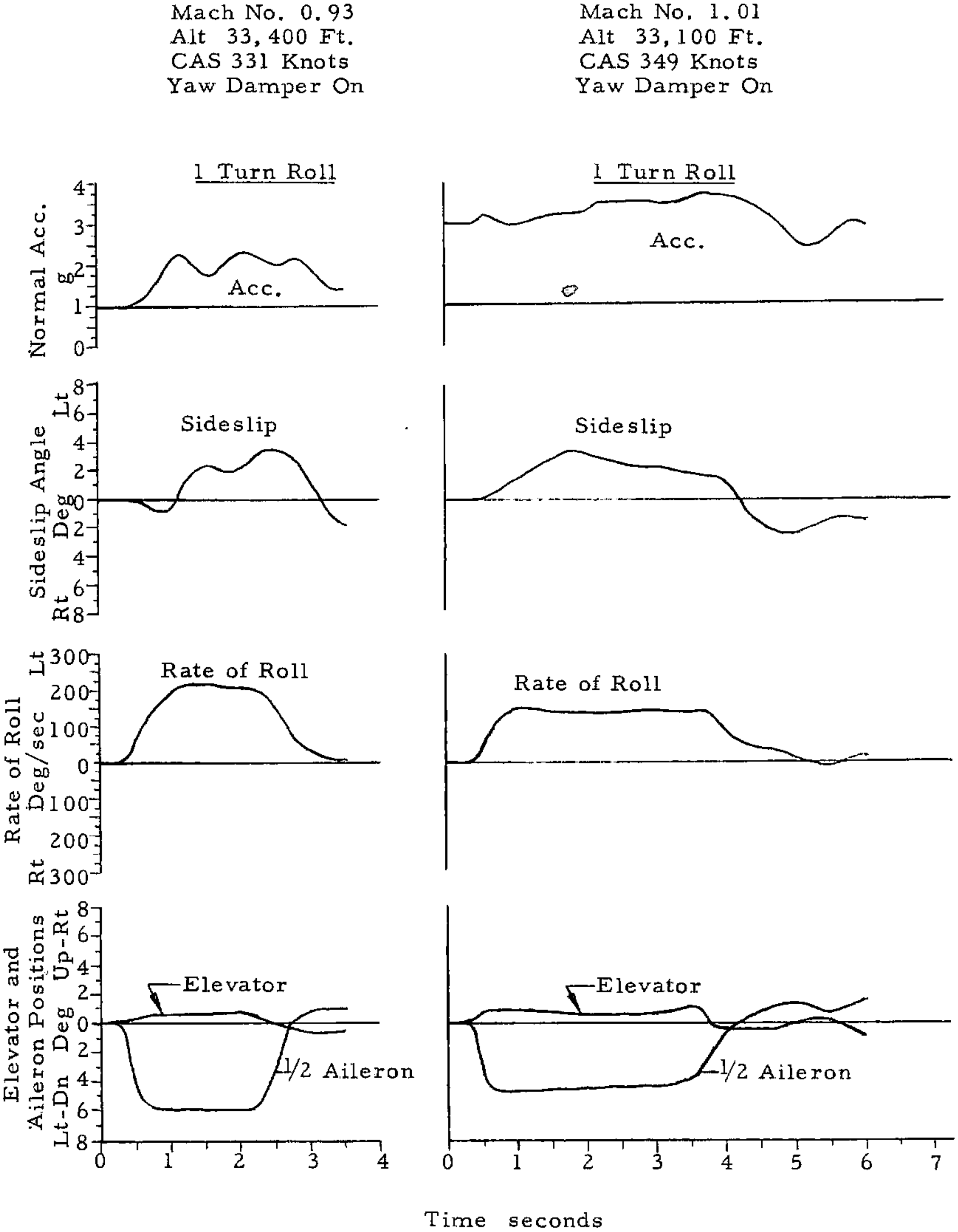
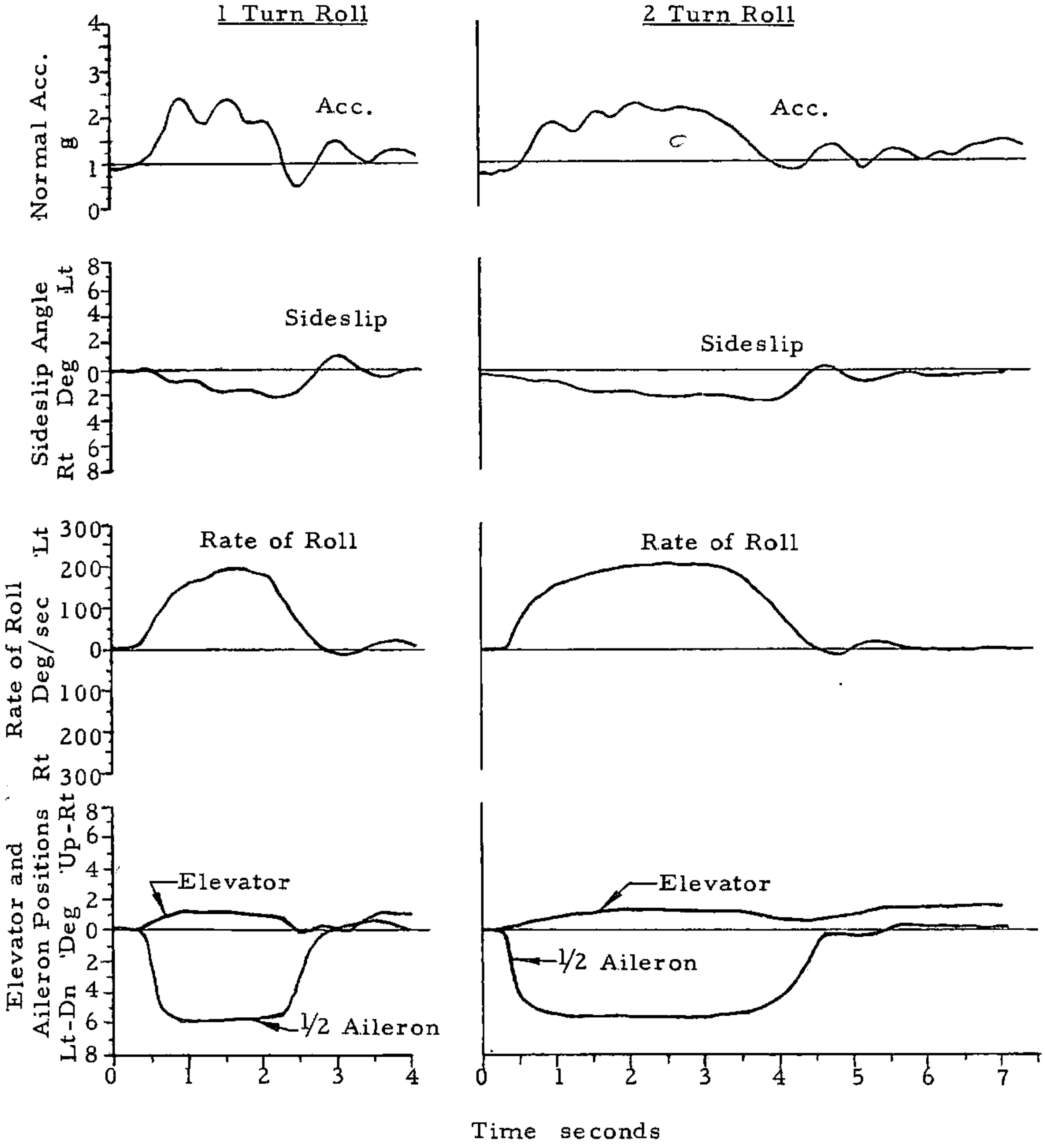


Figure No. 12
AILERON ROLLS
F-102A USAF No. 53-1813
Combat Configuration



Mach No. 1.13
Alt 34,600 Ft.
CAS 409 Knots
Yaw Damper On

Mach No. 1.15
Alt 34,600 Ft.
CAS 417 Knots
Yaw Damper Off

Figure No. 13
AILERON ROLLS
F-102A USAF No. 53-1813
Combat Configuration

Mach No. 1.11
Alt 33,200 Ft.
CAS 412 Knots
Yaw Damper Off

Mach No. 1.10
Alt 33,500 Ft.
CAS 400 Knots
Yaw Damper On

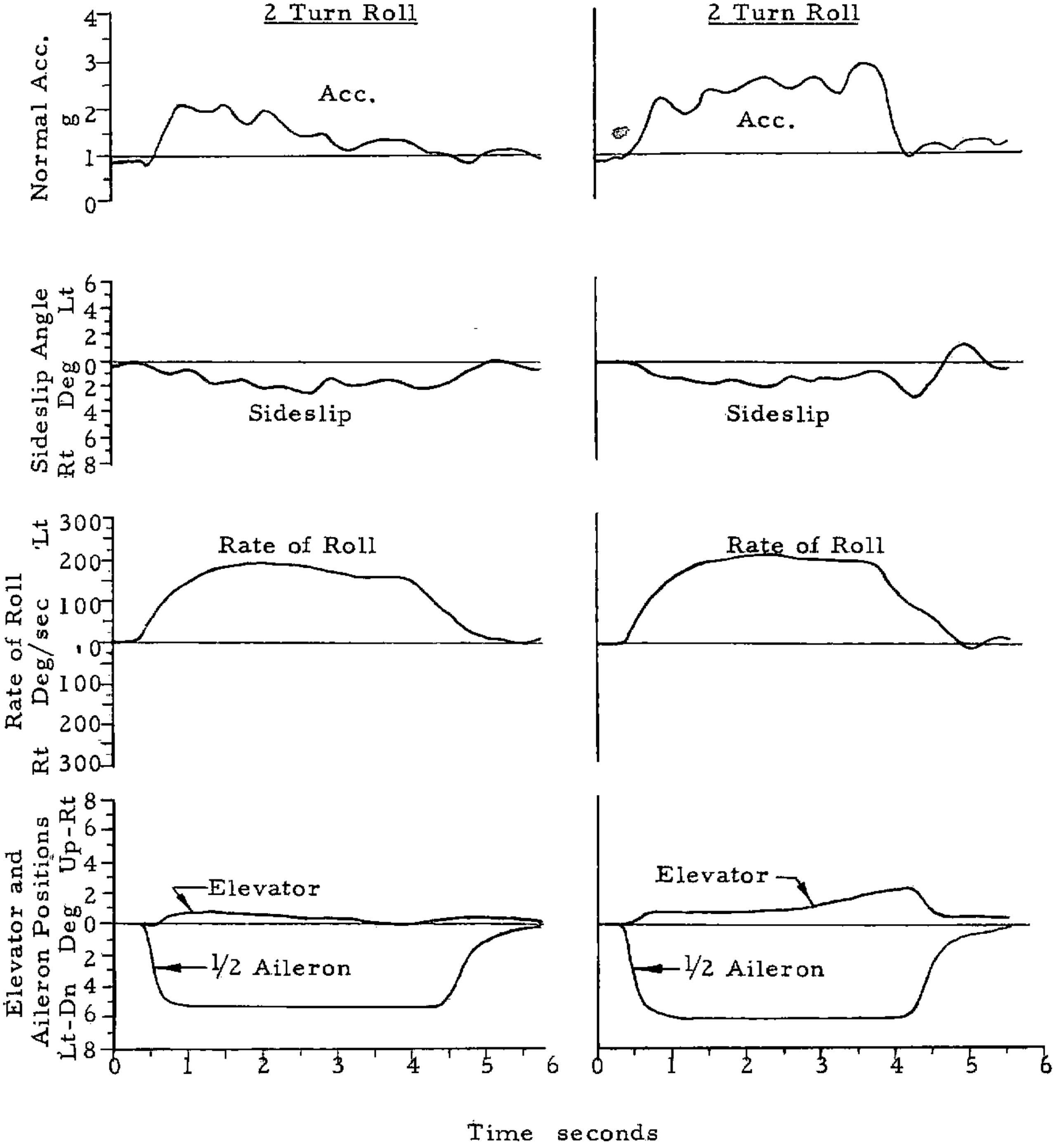


Figure No. 14
DYNAMIC LATERAL-DIRECTIONAL STABILITY
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.77 Altitude 35,000 Ft.

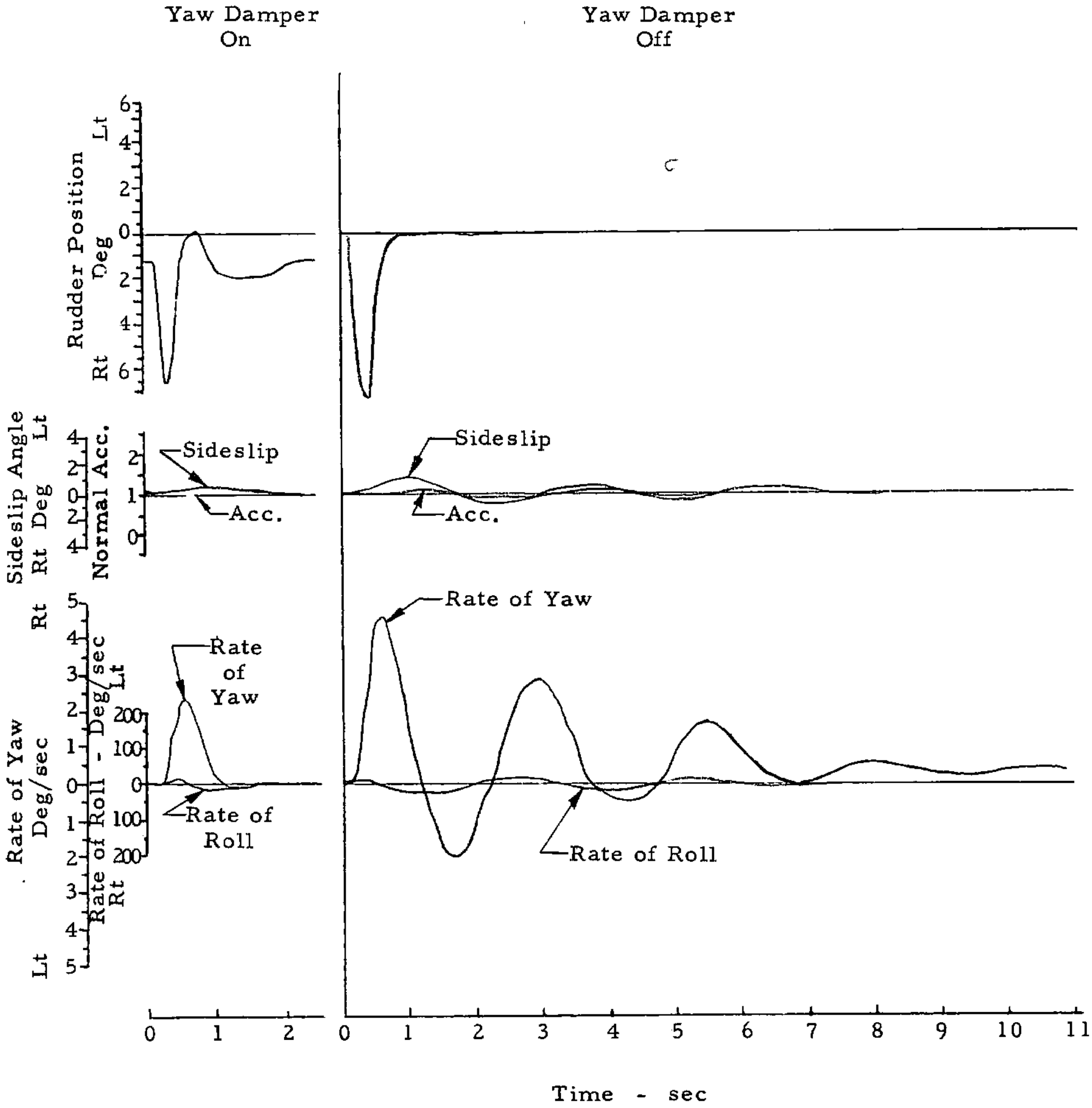


Figure No. 15
DYNAMIC LATERAL-DIRECTIONAL STABILITY
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.93 Altitude 35,000 Ft.

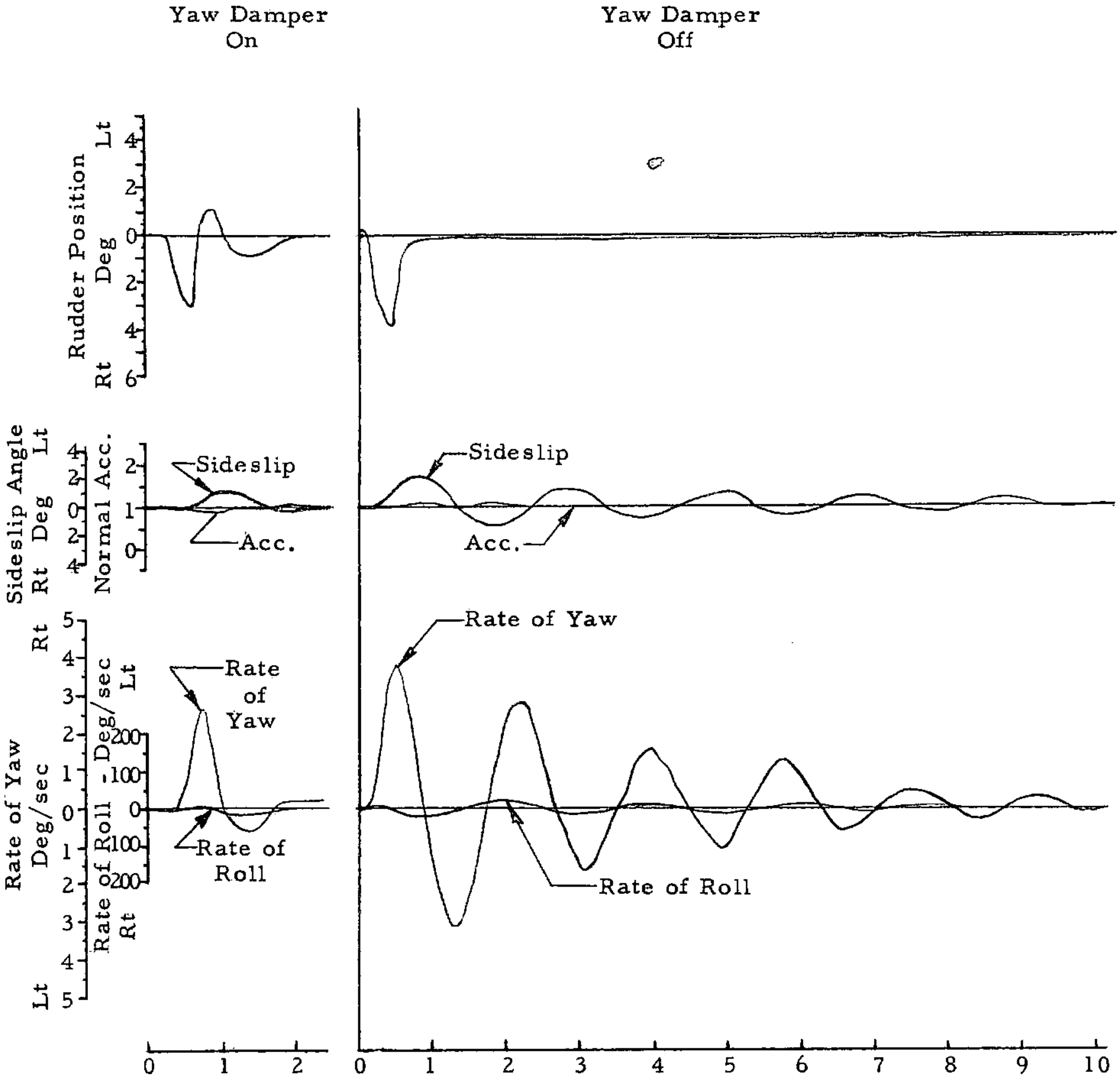


Figure No. 16
DYNAMIC LATERAL-DIRECTIONAL STABILITY
F-102A USAF No. 53-1813
Combat Configuration

Mach No. 1.16 Altitude 35,000 Ft.

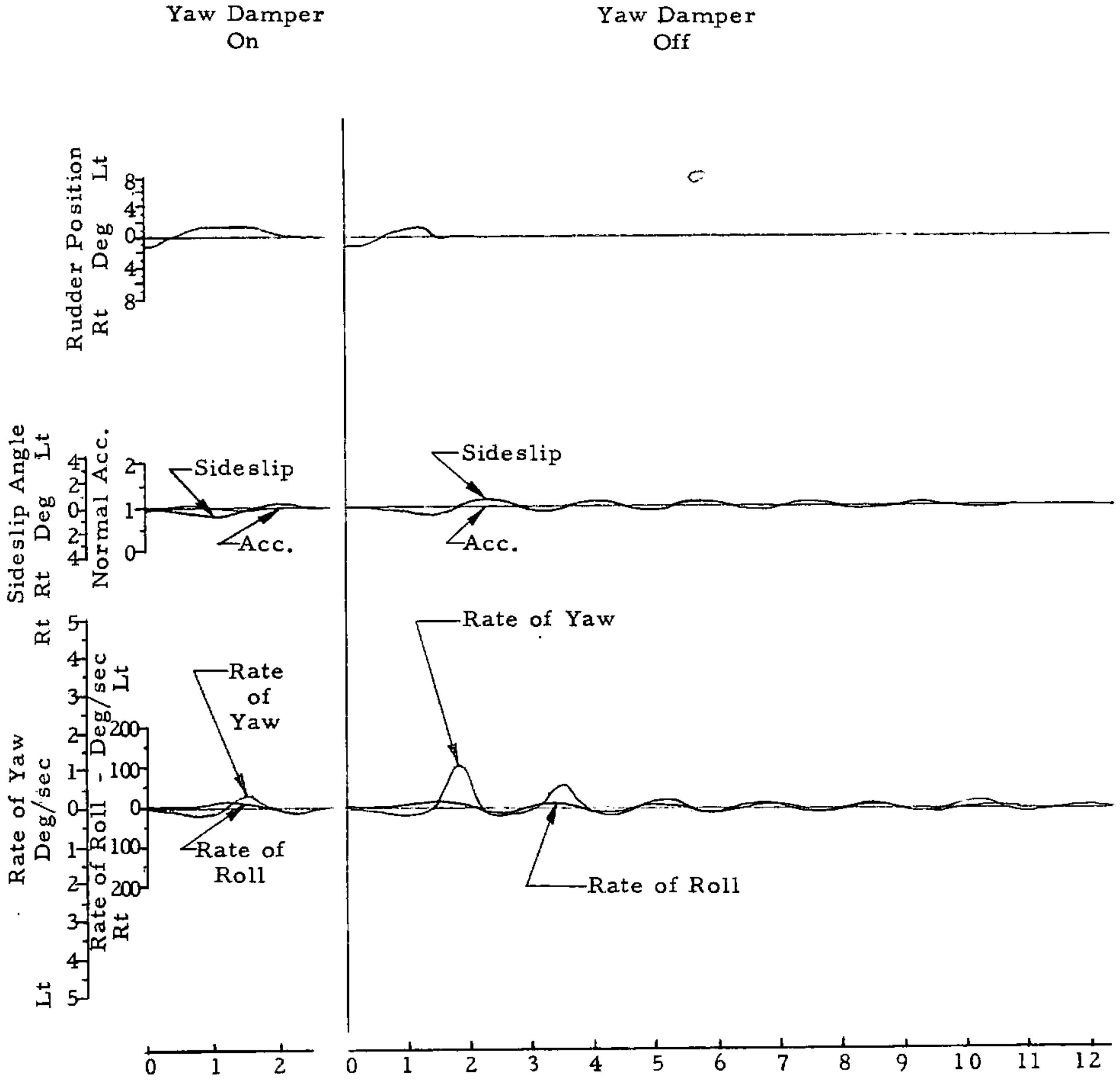


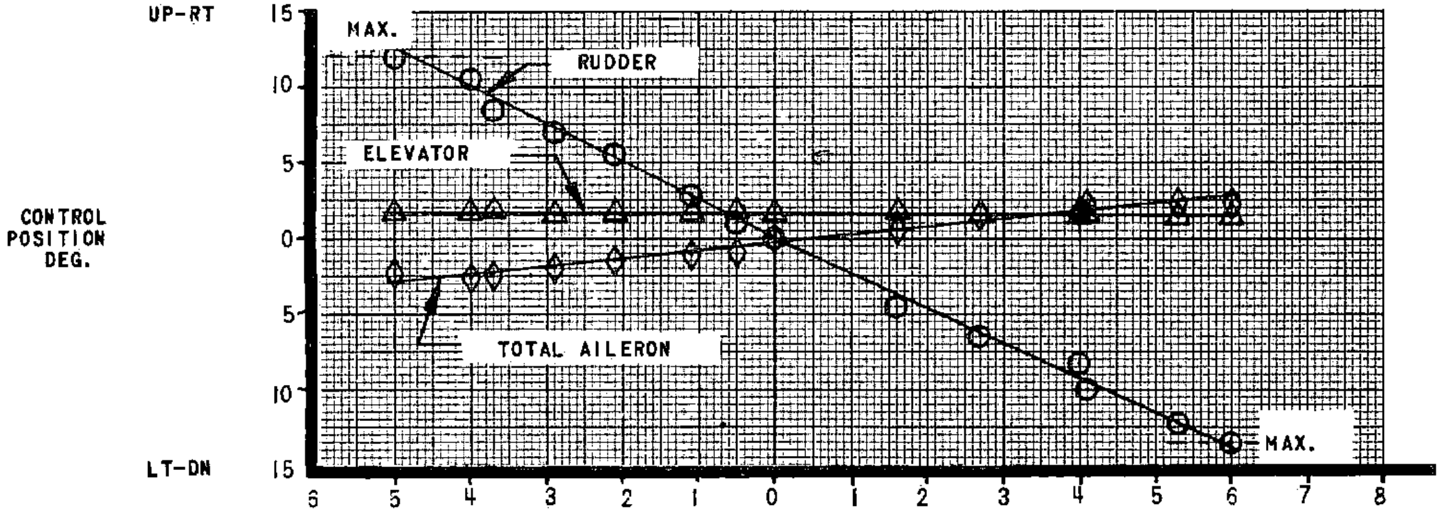
FIGURE 17

STATIC DIRECTIONAL STABILITY

TURN COORDINATOR AND YAW DAMPER ON

CRUISE CONFIG.
IAS 255 KNOTS
ALT. 34,700 FT.
MACH NO. 0.77

NOTE: MAXIMUM RUDDER DEFLECTIONS ATTAINABLE WERE LIMITED BY A "q" BOX



POWER APPROACH CONFIG.
IAS 180 KNOTS
ALT. 12,000 FT.

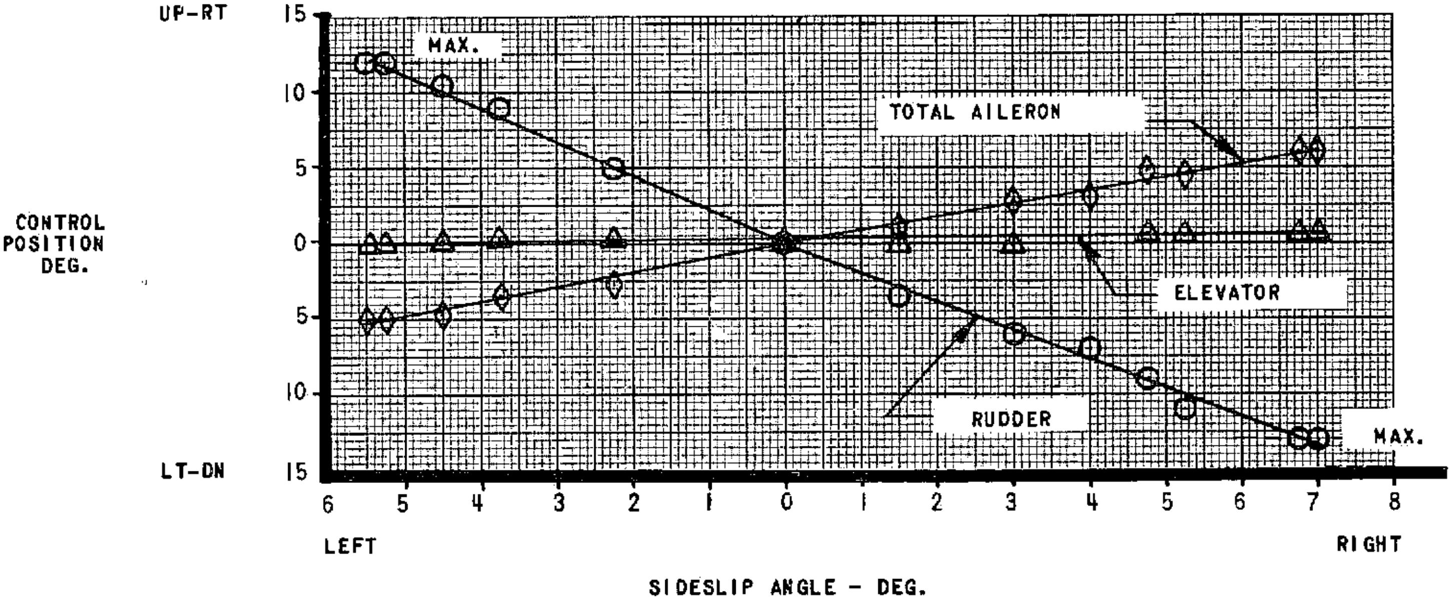
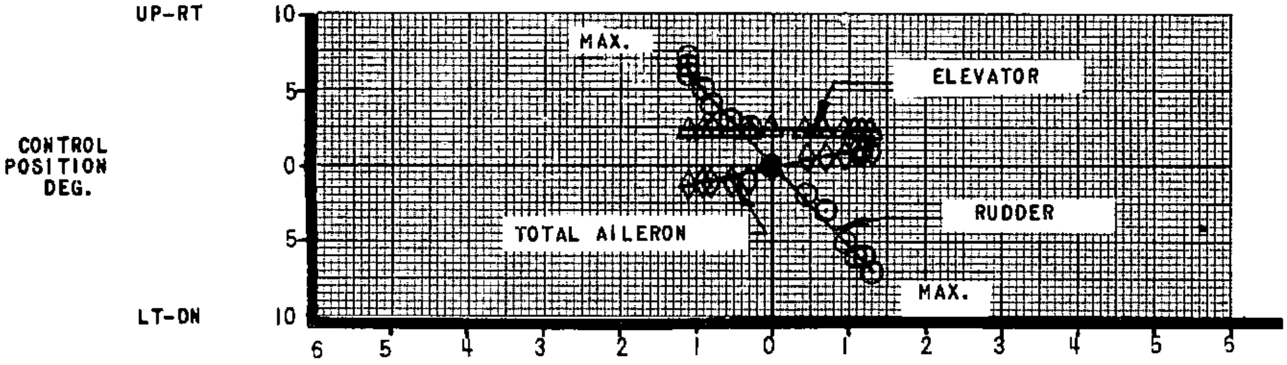


FIGURE 18 STATIC DIRECTIONAL STABILITY

TURN COORDINATOR AND YAW DAMPER ON

COMBAT CONFIG.
IAS 426 KNOTS
ALT. 35,300 FT.
MACH. NO. 1.18

NOTE: MAXIMUM RUDDER DEFLECTIONS ATTAINABLE WERE LIMITED BY A "q" BOX



CRUISE CONFIG.
IAS 312 KNOTS
ALT. 35,000 FT.
MACH. NO. 0.91

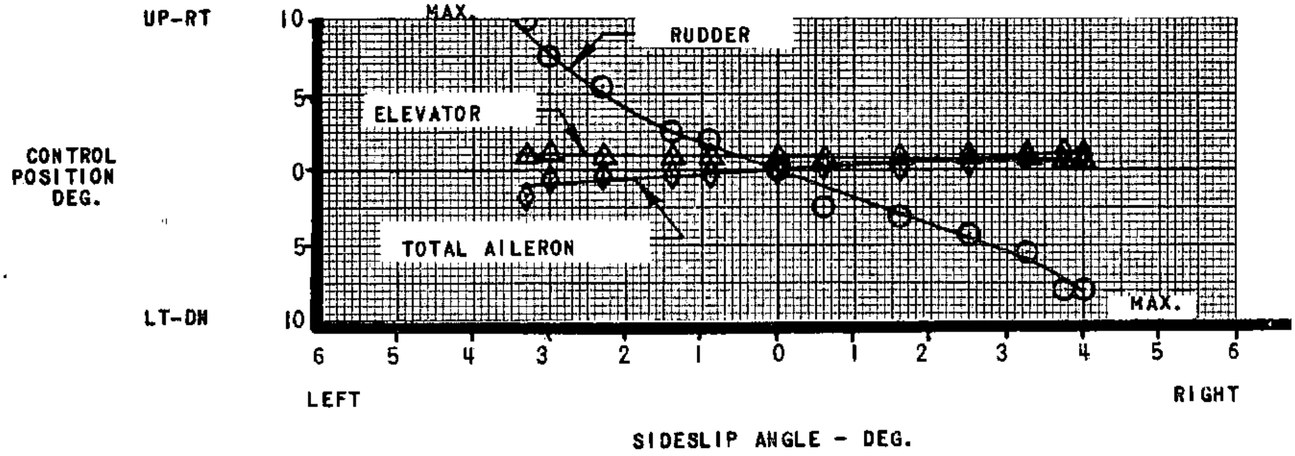


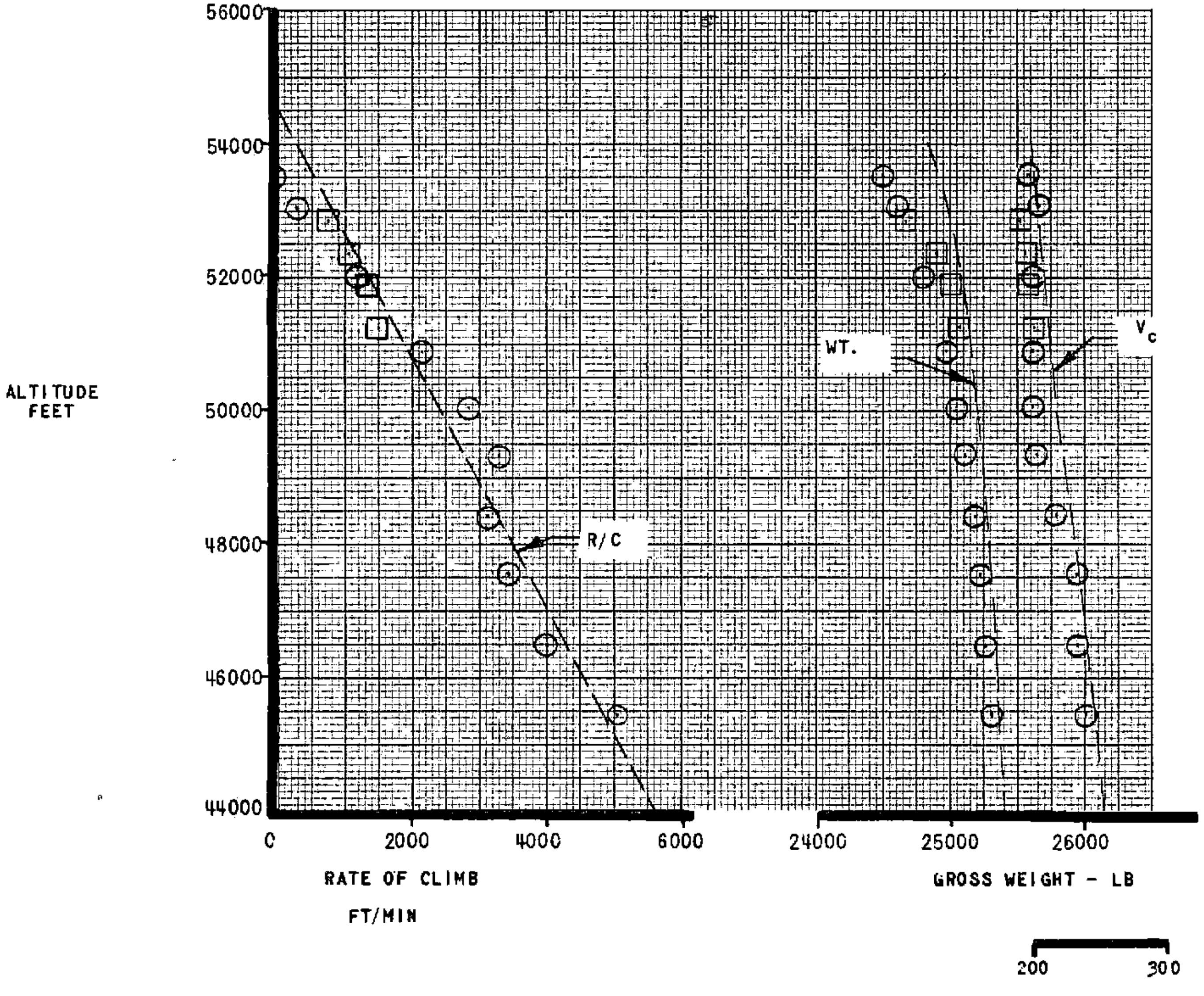
FIGURE 19

CLIMB PERFORMANCE

MAXIMUM POWER

J57-P-41 ENG S/N 604217
A/B S/N 600563
TRIMMED TO SIMULATE A
J57-P-23 ENG.

NOTE: DASHED LINES ARE DATA FROM PHASE II YF-102A, AFFTC-TR-55-31



NOTE: DATA NOT PRESENTED AT LOWER ALT. BECAUSE CLIMBS WERE NOT FLOWN AT CHECK CLIMB SPEED SCHEDULE

FIGURE 20

LEVEL FLIGHT PERFORMANCE

CORRECTED THRUST vs MACH NO.

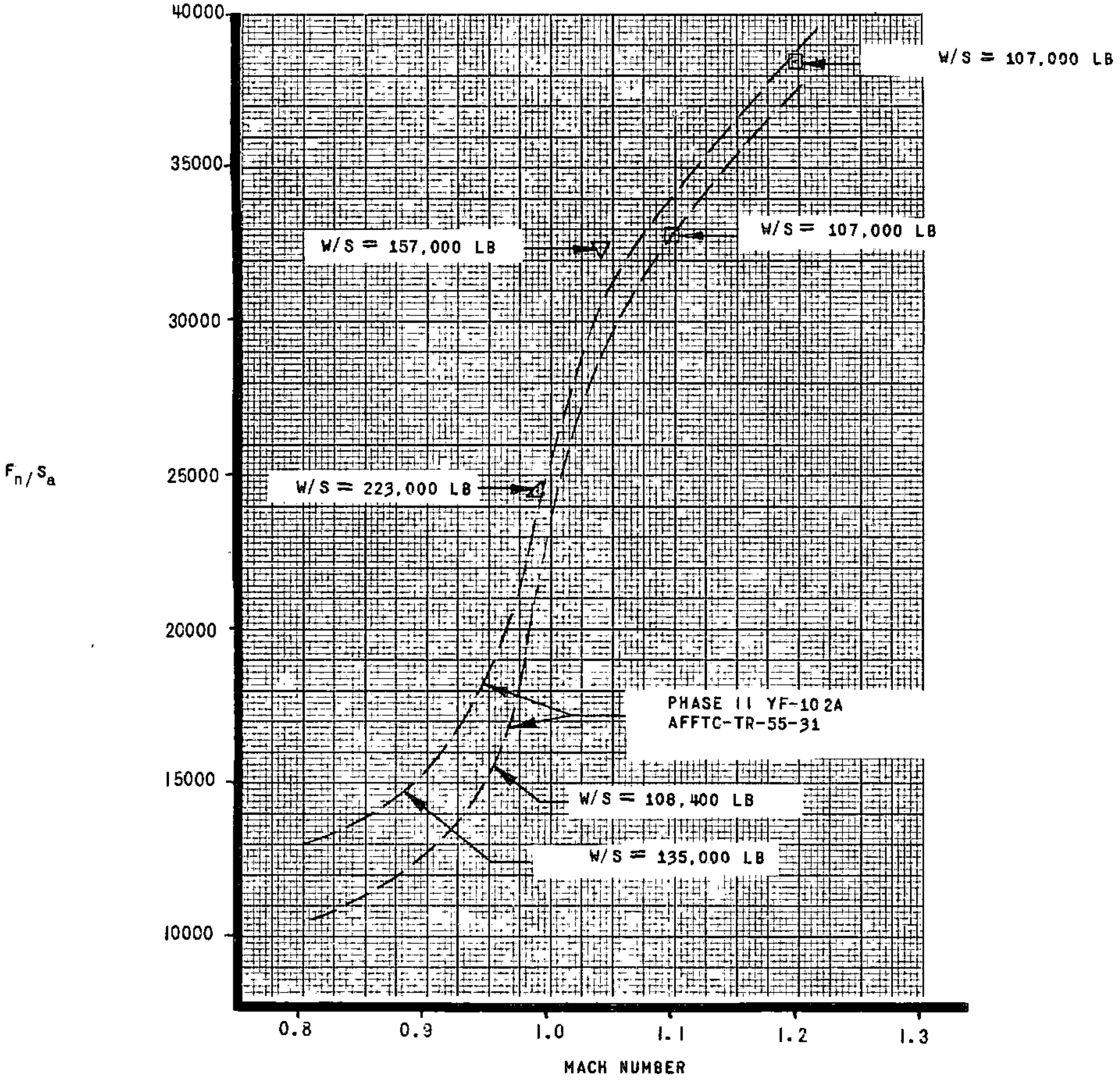


FIGURE 21 DRAG RISE

SYM	ALTITUDE	W/S
⊡	37,000 FT.	107,000 LB
▽	44,000 FT.	157,000 LB
◿	50,500 FT.	223,000 LB

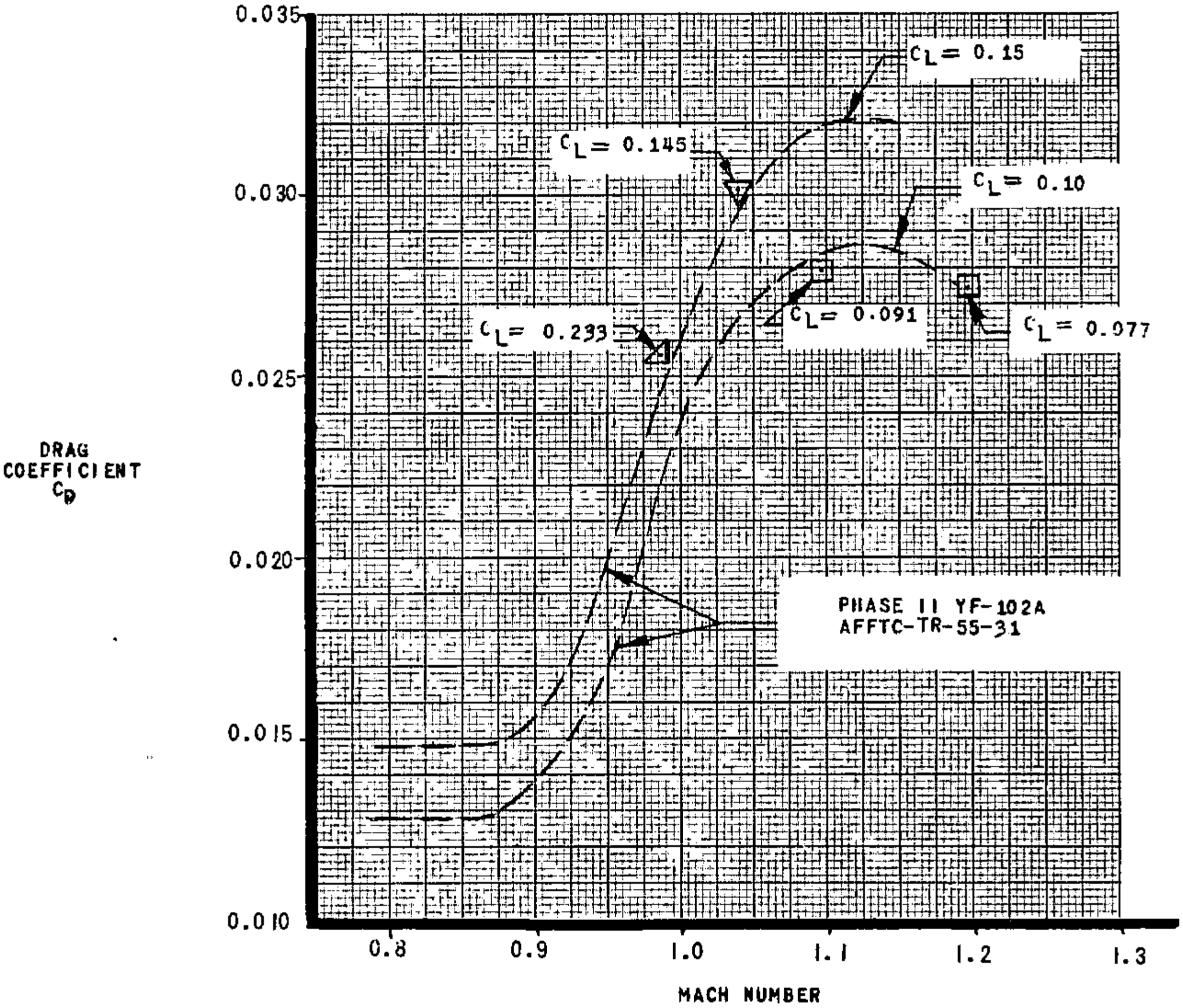


FIGURE 22 STATIC THRUST CALIBRATION

SYM	DATE
⊙	17 JAN.
⊡	17 JAN.

J57-P-41 S/N 604217
AFTERBURNER S/N 600563
IRIS TYPE NOZZLE
DOWNSTREAM PROBES

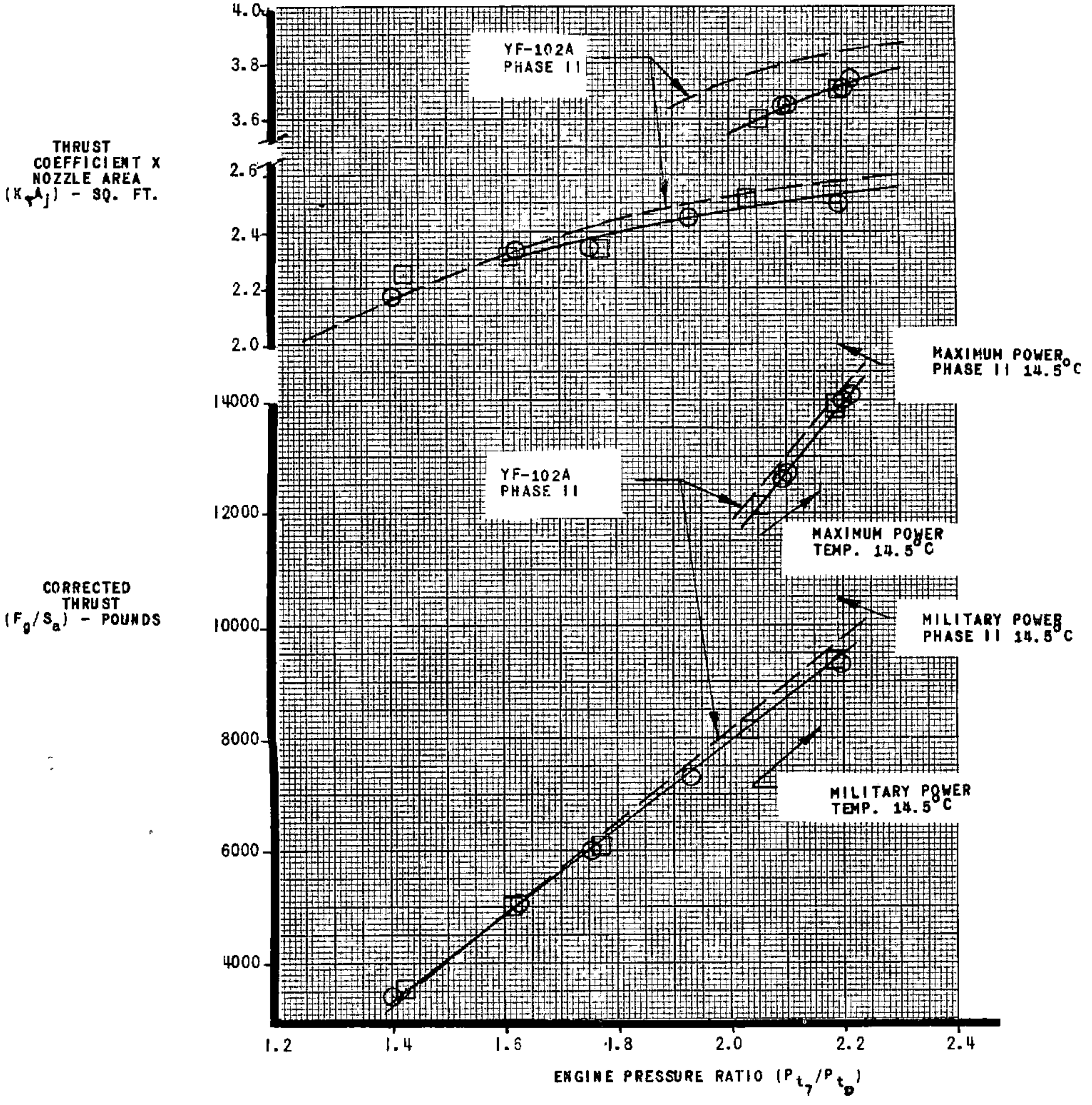


TABLE I

ROLLS FROM 60 DEGREE BANK TO 60 DEGREE BANK

12,000 feet, 180 knots IAS. (Power approach configuration, gear down.)

Total Aileron (deg)	Roll Rate (deg/sec)	Maximum Sideslip Angle (deg)	Maximum Normal Acc. (g)	Yaw Damper and Turn Coordinator
3.8 L	40 L	1.5 L	1.4	On
3.3 R	20 R	1.5 R	1.0	On
7.5 L	50 L	2.5 L	1.2	On
7.0 R	60 R	3.5 R	1.2	On
7.5 L	60 L	2.0 L	1.4	On
8.0 R	70 R	3.0 R	1.3	On
9.0 L	60 L	3.0 L	1.5	On
12.3 R	80 R	4.0 R	1.4	On
11.2 L	70 L	4.0 L	1.6	On
14.3 R	85 R	4.0 R	1.4	On
14.0 L	70 L	5.5 L	1.8	On
14.3 R	90 R	4.0 R	1.4	On

TABLE II

ROLLS FROM WINGS LEVEL THROUGH 360 OR MORE DEGREES

Cruise Configuration, 15,000 Feet

	Total Aileron (deg)	Roll Rate (deg/sec)	Maximum Sideslip Angle (deg)	Maximum Normal Acc. (g)	Number of Cont. Rolls	Yaw Damper and Turn Coordinator
Trim: 0.53 Mach number 260 knots IAS 15,000 feet	5 L	55 L	2 R	1.5	1	On*
	12 L	140 L	5.3 L	2.3	1	On
	13.5 L	140 L	5.75 L	2.5	2	On
	12.8 L	140 L	6.25 L	2.3	2	On
	12.8 L	130 L	6.25 L	3.0	2	On
Trim: 0.62 Mach number 310 knots IAS 15,000 feet	3 L	40 L	1.75 R	1.6	1	On
	4.8 L	65 L	1.5 L	1.9	1	On
	13 L	160 L	4.25 L	2.3	1	On
	13 L	160 L	6.5 L	3.3	2	On
	12.5 L	160 L	6.0 L	2.8	2	On
	10.3 R	140 R	3.0 R	2.1	2	On
	13.1 L	175 L	5.0 L	2.2	2	On
Trim: 0.81 Mach number 410 knots IAS 15,000 feet	4.8 R	100 R	2.0 L	1.6	1	On
	2.8 L	60 L	1.5 R	1.3	1	On
	4.2 L	90 L	1.5 R	1.6	1	On
	8 L	140 L	1.25 L	2.0	1	On
	11.2 L	190 L	1.0 L	2.2	1	On
	11.2 L	215 L	2.0 L	2.6	1	On
	9.5 L	175 L	1.75 L	2.3	2	On
	10 L	180 L	2.0 L	2.4	2	On
	10.1 L	180 L	5.0 L	2.8	2	On
	11.5 L	190 L	2.25 L	2.5	2	On
Trim: 0.95 Mach number 480 knots IAS 15,000 feet	8.5 L	140 L	1.5 L	1.9	60°—60°**	On
	8.75R	160 R	1.5 R	1.65	60°—60°**	On
	9.0 L	150 L	2.5 L	2.0	60°—60°**	On
	8.25R	135 R	1.5 R	2.4	60°—60°**	On
	7.25L	120 L	1.5 L	2.0	60°—60°**	On
	8.75R	130 R	2.25 R	2.0	60°—60°**	On
	7.5 L	135 L	2.0 L	1.8	1	On
	8.75R	140 R	2.0 R	1.3	1	On
	10.5 L	170 L	4.75 L	2.9	1	Off*

** These rolls were made with only 60 degree to 60 degree change in bank angle.

* The yaw damper and turn coordinator were both tied in to one switch in the cockpit and both were either on or off.

TABLE III

ROLLS FROM WINGS LEVEL THROUGH 360 OR MORE DEGREES

Cruise Configuration, 35,000 Feet

	Total Aileron (deg)	Roll Rate (deg/sec)	Maximum Sideslip Angle (deg)	Maximum Normal Acc. (g)	Number of Cont. Rolls	Yaw Damper and Turn Coordinator
Trim: 0.77 Mach number 255 knots IAS 35,000 feet	7.6 L	130 L	1.0 L	1.9	1	On
	7.5 R	120 R	.7 R	2.2	1	On
	10 L	175 L	1.5 L	2.1	1	On
	9.9 R	150 R	1.4 L	2.5	1	On
	13 L	185 L	2.5 L	2.4	1	On
	10 R	175 R	1.5 R	2.1	1	On
	11.5 L	180 L	5.0 L	2.2	1	On
	10 R	180 R	3.5 R	2.0	1	On
	9.2 L	125 L	5.7 L	3.1	2	On
	8.7 L	135 L	5.8 L	3.2	2	On
	9.5 L	150 L	3.7 L	2.5	3	On
	10.7 L	170 L	4.0 L	3.0	4	On
	10 L	150 L	3.0 L	3.1	3	On
	10.2 L	130 L	3.6 L	3.2	3	On
	11.2 L	150 L	3.0 L	2.2	1	On
	11.3 L	140 L	2.2 L	2.2	1	Off
	10 L	135 L	3.0 L	2.1	1	On
	10.5 L	135 L	3.5 L	2.2	1	Off
Trim: 0.92 Mach number 314 knots IAS 35,000 feet	7.5 L	150 L	.6 L	1.7	1	On
	8.5 L	150 L	1.5 L	2.0	1	On
	7.0 R	130 R	1.7 L	1.6	1	On
	9.2 L	180 L	1.5 L	1.8	1	On
	8.8 R	160 R	2.0 L	1.8	1	On
	10.7 L	180 L	2.8 L	2.0	1	On
	13.7 R	215 R	2.0 L	2.3	1	On
	12.2 L	210 L	3.0 L	2.0	1	On
	13 R	210 R	2.0 L	2.1	1	On
	12.6 L	215 L	3.4 L	2.3	1	On
	10.2 L	175 L	2.5 L	2.6	1	On
	9 L	160 L	3.5 L	3.3	1	On
	9 L	150 L	3.5 L	4.5	1	On
	9 R	175 R	.7 R	2.8	1	On
	9.2 L	185 L	1.2 L	3.0	3	On
	11 L	175 L	2.0 L	3.2	3	On
	8.8 L	160 L	2.0 R	2.0	1	On
	7 L	170 L	3.0 R	2.7	1	Off
Trim: 1.12 Mach number 400 knots IAS 35,000 feet	5.3 L	80 L	.8 R	1.1	1	On
	4.7 R	75 R	1.5 L	1.5	1	On
	7.5 L	110 L	1.5 R	1.5	1	On
	7.5 R	110 R	1.5 L	1.2	1	On
	11.5 L	175 L	2.0 R	2.2	1	Off
	10.7 R	160 R	2.0 L	1.2	1	On
	12.2 L	185 L	2.3 R	2.4	1	On
	10.7 L	160 L	1.8 R	2.2	2	On
	12.2 L	200 L	2.5 R	2.4	2	Off
	10.5 L	180 L	2.3 R	2.3	2	Off
	12.0 L	200 L	2.5 R	2.8	2	On
	12.7 L	200 L	2.0 R	2.9	1	On

appendix II flight log

FLIGHT	DATE	TIME	PILOT
1	11 Jan 56	1:20	Stephens, R. L., Major, USAF
2	13 Jan 56	:55	Childs, S. R., Major, USAF
3	13 Jan 56	:50	Childs, S. R., Major, USAF
4	18 Jan 56	:35	Childs, S. R., Major, USAF
5	18 Jan 56	:35	Childs, S. R., Major, USAF
6	18 Jan 56	:40	Childs, S. R., Major, USAF
7	19 Jan 56	:55	Hanes, H. A., Col., USAF
8	20 Jan 56	:35	Baker, R. M., Col., USAF
9	20 Jan 56	:55	Holtoner, J. S., Brig. Gen., USAF
10	21 Jan 56	:35	Everest, F. K., Lt. Col., USAF
11	21 Jan 56	:45	White, R. M., Capt., USAF
12	21 Jan 56	:55	Stephens, R. L., Major, USAF
13	22 Feb 56	1:05	Stephens, R. L., Major, USAF
14	23 Feb 56	:35	Childs, S. R., Major, USAF
15	24 Feb 56	:50	Stephens, R. L., Major, USAF
	Total	**12:05**	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A USAF NO. 53-1813
EVALUATION OF ENLARGED VERTICAL STABILIZER

TEST: Check Climb (Max. Power) Weight at Engine Start 27,640 lb.

Alt. Ft.	IAS Knots	Time Min:Sec	P_{t_7}/P_{t_0}	t_a °C	Gross Wt. Lb.	N_I %RPM
40,000	----	----	2.60	----	25,500	----
41,740	257	27:15	----	-11	----	6260
43,710	249	27:36	----	-12	----	----
44,780	247.5	27:48.2	----	-12	----	6160
45,000	----	----	2.62	----	25,300	----
45,830	240.5	28:02	----	-12	----	6160
46,810	243	28:20	----	-13	----	6150
47,790	224	28:43.8	----	-14	----	6150
48,810	210	28:57.5	----	-15	----	6150
49,780	210	29:17.3	----	-16	----	6105
50,000	----	----	2.67	----	25,000	----
50,320	210	29:35	----	-18	----	6105
50,800	209	29:51	----	-20	----	----
51,330	208.5	30:13	----	-20	----	6120
51,800	210	30:48	----	-21	----	----
52,000	----	----	2.72	----	24,700	----
52,325	210	31:29.7	----	-23	----	6130
52,585	205.5	31:59.2	----	-23	----	----
52,835	204.5	32:21	----	-24	----	6130
53,000	----	----	2.74	----	24,500	----
53,085	201	33:14.5	----	-25	----	----
39,690	284.5	57:01	----	-11	----	
40,000	----	----	2.57	----	25,600	
41,680	269	57:29	----	-11	----	
43,690	256.5	58:02	----	-10	----	
44,760	251	58:26	----	-10	----	
45,000	----	----	2.58	----	25,400	
45,730	243.5	58:42	----	-10	----	
46,780	234	59:02	----	-12	----	
47,810	229	59:21.5	----	-13	----	
48,820	221.5	59:39.5	----	-14	----	
49,830	211	59:55	----	-15	----	
50,000	----	----	2.61	----	25,100	
50,240	210	00:05	----	-15	----	
50,690	211	00:25	----	-16	----	
51,000	----	----	2.62	----	25,050	
51,330	206	00:50.5	----	-16	----	
51,800	203	01:14.5	----	-18	----	
52,310	200.5	02:45	----	-20	----	
52,525	199	03:23	----	-22	----	
52,800	----	----	2.63	----	24,700	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A USAF NO. 53-1813
EVALUATION OF ENLARGED VERTICAL STABILIZER

TEST: Level Flight Data — Weight at Engine Start 27,640 lb.

Alt. Ft.	IAS Knots	t_a °C	P_{t_7}/P_{t_0}	Gross Wt. Lb	N_1 RPM	A/B
50,630	232.5	-23	2.67	23,800	6170	Off
37,210	374.5	+1	2.41	22,500	5970	On(Min.)
43,910	300.5	-2	2.47	24,050	6120	On
37,020	412.0	+12	2.47	23,000	6180	On

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A USAF NO. 53-1813
EVALUATION OF ENLARGED VERTICAL STABILIZER

TEST: Static Thrust Calibration Weight at Engine Start 27,640 lb.

Run No.	Fg Lb	P_{t_7}/P_a	N_2 % RPM	EGT °C	A/B	
First Run	$P_a = 27.65$	$t_a = 9^\circ C$				
1	5540	1.755	89.5	415	Off	
2	13,030	2.215	97.15	590	On	
3	4695	1.625	88.85	400	Off	
4	11,755	2.103	95.35	540	On	
5	3155	1.405	85.15	355	Off	
6	6790	1.93	93.0	490	Off	
7	11,660	2.095	95.15	540	On	
8	435	----	55.90	220	Off	
9	8620	2.195	97.05	585	Off	
10	12,897	2.20	97.15	585	On	
Second Run	$P_a = 27.59$	$t_a = 12^\circ C$				
1	5635	1.77	90.2	425	Off	
2	8670	2.185	97.9	600	Off	
3	12,735	2.192	98.0	600	On	
4	4640	1.62	89.7	405	Off	
5	7525	2.035	95.7	540	Off	
6	11,160	2.055	95.8	590	On	
7	3260	1.425	86.2	380	Off	
8	8660	2.185	98.0	600	Off	
9	12,795	2.198	98.05	600	On	

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TF-102A

PHASE II FLIGHT TESTS

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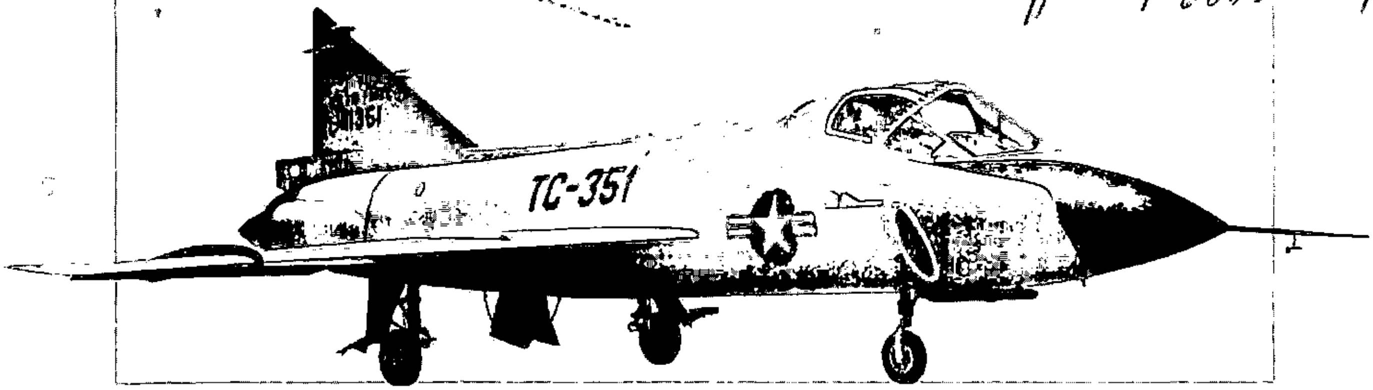

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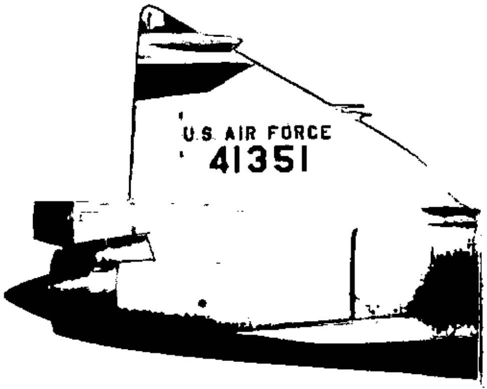

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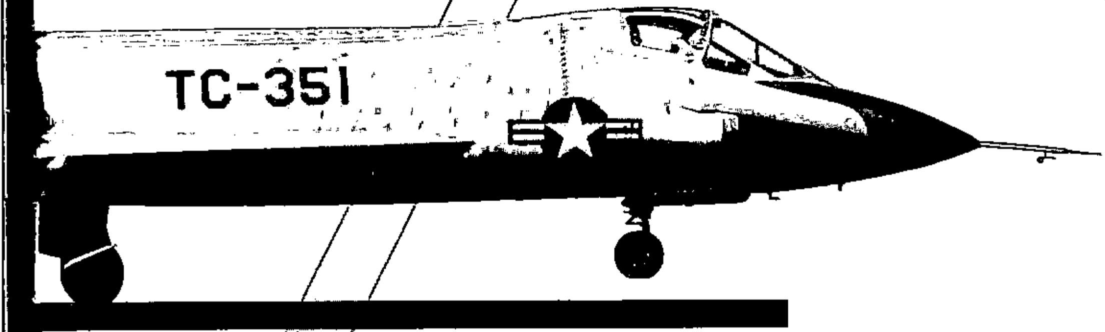

TF-102A Phase II

AFFTC-TR-56-26 • OCTOBER 1956
ASTIA DOCUMENT NO. AD 098040

JESSE B. HOUSTON, JR., 1/LT., USAF, PROJECT ENGINEER
ROBERT D. HIPPERT, CAPTAIN, USAF, PROJECT PILOT

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ABSTRACT

THE TF-102A *is acceptable as a transitional trainer for delta-wing aircraft. The airspeed-altitude performance envelope is significantly inferior to performance characteristics of the F-102A. This performance degradation combined with airframe buffet encountered in the .8 to .9 Mach number region will tend to restrict the utility of the aircraft as a weapon system proficiency trainer.*

Contractural guarantees for take-off, maximum power climb performance, and service ceiling are met or exceeded, but the aircraft is deficient with respect to military power climb performance and maximum speed.

In general, handling characteristics are acceptable.

THIS REPORT
HAS BEEN
REVIEWED
AND APPROVED
18 OCTOBER 1956

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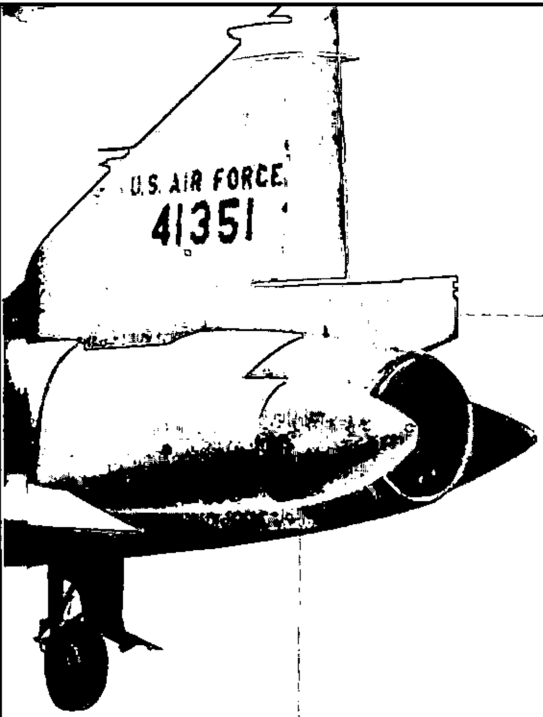

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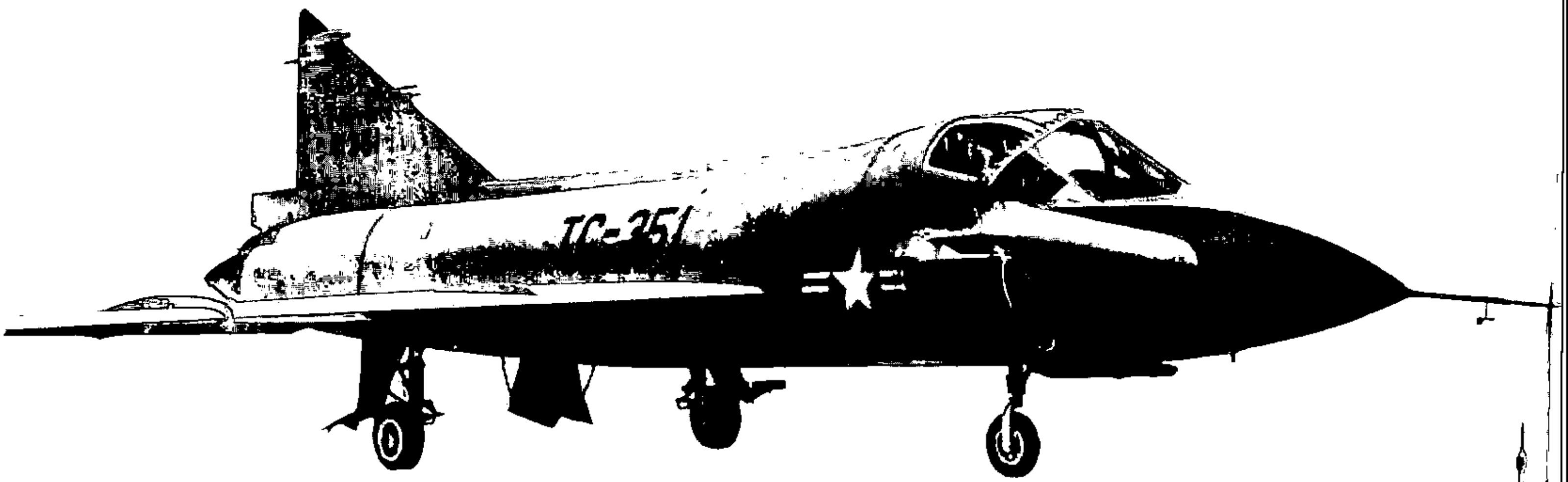
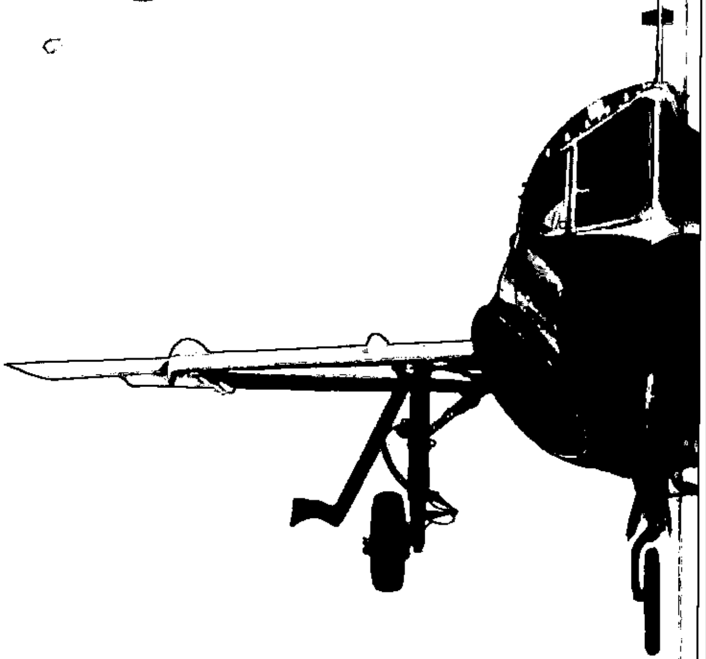
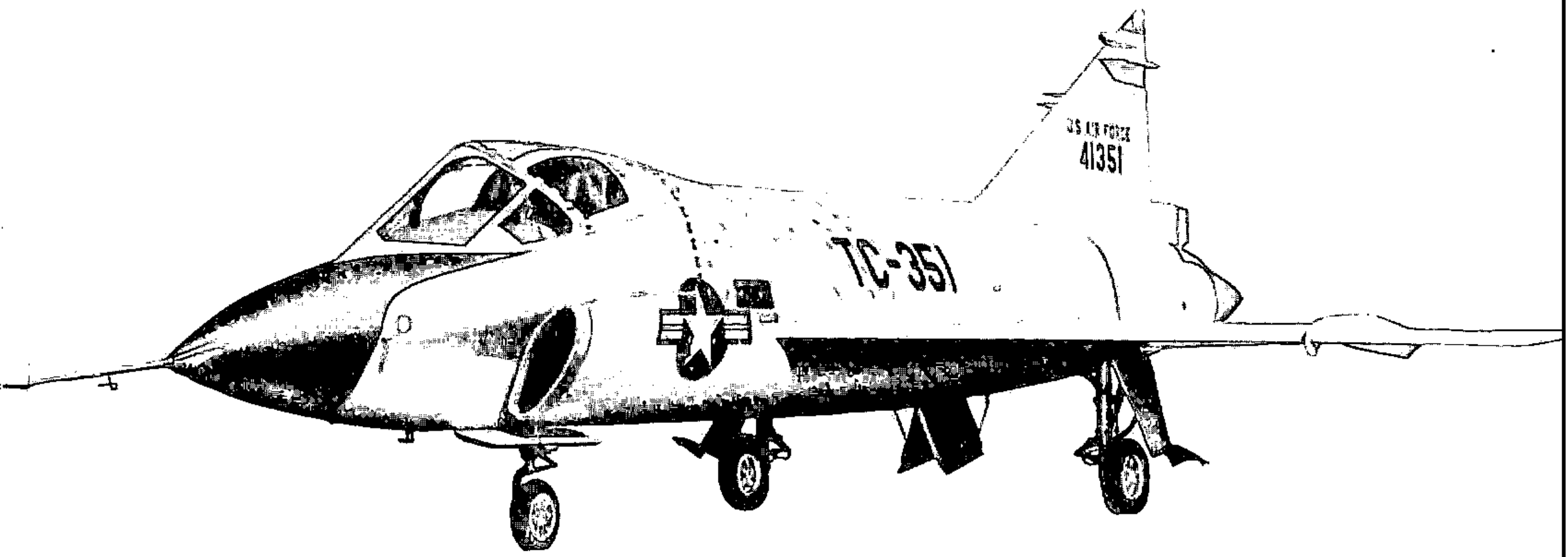
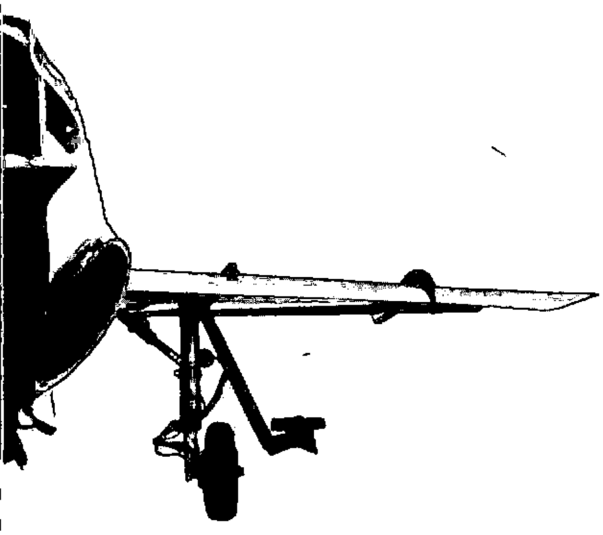




INTRODUCTION



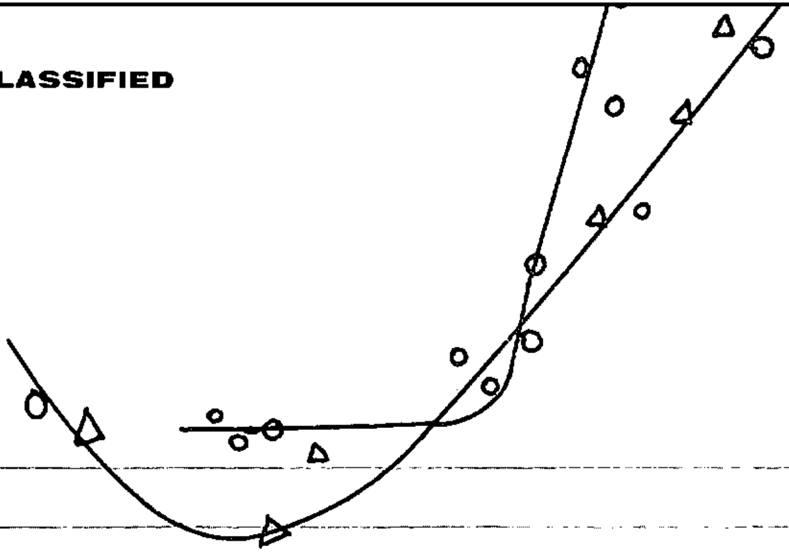


This report presents the performance and stability and control characteristics of the TF-102A airplane as determined from the Phase II flight evaluation. These tests were conducted to determine if contractural guarantees were met and to evaluate the aircraft at an early stage of development. The test program was conducted under the authority of the Commander, Air Force Flight Test Center at the request of the Air Research and Development Command. The tests were accomplished at the Air Force Flight Test Center, Edwards Air Force Base, California, between 11 June and 19 June 1956, and required nine flights totaling 10 flying hours. The test aircraft was maintained by the contractor at the Edwards AFB facility.

This report also includes quantitative performance data obtained during prior tests of two TF-102A aircraft conducted by a joint AFFTC-ADC team for the purpose of making a comparative evaluation of two canopy configurations.

The TF-102A airplane is the two place trainer version of the F-102A, featuring a side-by-side seating arrangement for instructor and student. The trainer aircraft is identical to the F-102A, except that the cockpit, nose section and intake ducts are modified to accommodate the side-by-side seating arrangement. The airplane is powered by a J57-P-23 turbo-jet engine with afterburner (flap type nozzle, tandem cylinder, forged flap) rated at 16,000 pounds sea level static thrust.

The gross weight of the aircraft at engine start is 28,300 pounds, including the pilots, 7098 pounds of fuel, and full oil supply. The center of gravity varies with fuel consumption from 26.1 percent MAC at engine start, to 19.7 percent MAC when the fuel load is completely expended.



TEST RESULTS

cockpit

Entrance to the cockpit is made with the use of external ladders that rest against the side of the fuselage. Since the ladders are not attached to the aircraft and are free to slide, they constitute a ground safety hazard.

The cockpit affords ample room and adequate adjustments for comfort. Basic instrument arrangement and cockpit switches are consistant with the F-102A, except that the circuit breakers on the aft portion of the left console are easier to see and reach than they are on the F-102A. The rudder trim switch, located on the center pedestal in front of the stick, is difficult to reach under all flight conditions. The switch should be placed in a more convenient location: perhaps on the left console just aft of the throttle. The ram air turbine handle is difficult to actuate with the throttle in the idle cut-off position as it would be for a flame-out landing. This could be corrected by placing the turbine handle beside the emergency gear down handle.

The detailed production cockpit arrangement was altered for test purposes. The most noticeable difference is the omission of the radar control console and presentation screen.

taxiing and ground handling

Control of the aircraft during taxi operations is satisfactory. Good directional control can be maintained with nose wheel steering. Directional control utilizing differential braking is possible but is not recommended due to the loss in taxi speed which results. Rudder forces during taxi operations are high but not excessive.

Visibility forward and to the sides is satisfactory for both the pilot and instructor. Due to the seating, the student pilot is completely dependent on the instructor for visibility to the right rear, and the instructor is dependent on the pilot for visibility to the left rear. With the canopy open, as often is the case in taxiing, the pilot's visibility to the left rear (or to the right rear for the instructor) is satisfactory. With the canopy closed, however, visibility to the rear is impaired to the extent that only about 1 foot of the elevon can be seen. With the minimum crew of one, as in ferry flights or functional check flights, the visibility to the right rear is non-existent.

The canopy hold-open device is excellent, but the force required to lower the canopy is excessive. The efforts of both aircraft occupants are required to close the canopy.

Idle power is sufficient for all taxi operations except initial movement after engine start.

take-off and initial climb

Adequate braking is available to hold the aircraft during military power run-up prior to the take-off roll. The brakes should be released and take-off roll begun when the afterburner is lighted.

Nose wheel steering is adequate for directional control until the rudder becomes effective at approximately 70 knots IAS. Nose wheel steering is not recommended above 90 knots IAS due to a tendency to overcontrol while making directional corrections.

Maximum power take-off speeds closely approximate those of the F-102A. Minimum nose wheel lift-off speed is 125 knots IAS. Recommended take-off indicated airspeed is 140 knots (2100 feet ground roll). A higher speed requires excessive nose high attitude after take-off to prevent exceeding gear down limit speeds. A total distance of 3500 feet is required to reach 50 feet with a speed of 160 knots IAS. The nose attitude appears slightly higher in the trainer than in the F-102A due to the short nose, but visibility over the nose is satisfactory during lift-off and gear retraction. Speed increase is moderate, and 1 minute is required to reach the climb schedule.

Military power take-offs are satisfactory, although the take-off roll is long and acceleration to take-off speed is slow. A minimum nose-wheel-off speed of 130 knots IAS and lift-off speed of 140 knots IAS are recommended for positive control. The take-off roll requires approximately 5000 feet at 150 knots and a total distance of approximately 8000 feet is required to reach 50 feet at 165 knots. Two minutes are required to accelerate from brake release to the climb schedule.

Immediately after take-off a pilot-induced lateral oscillation is generally encountered. High aileron breakout forces and relatively large aileron control movements which follow break-out create a tendency to over control a correction for wing drop. The landing gear handle requires excessive force to actuate and the landing gear warning light is difficult to see. Take-off performance for standard sea level conditions is presented in Appendix I, Figure 1.

SUMMARY OF TAKE-OFF PERFORMANCE Weight 28,300 lbs.

TEST CONDITIONS	INDICATED AIRSPEED AT TAKE-OFF — KTS.	GROUND ROLL FT.	INDICATED AIRSPEED AT 50 FT. HEIGHT — KTS.	TOTAL DISTANCE TO CLEAR 50 FEET FT.
Maximum power	**140**	**2100**	**160**	**3500**
Maximum power	**160**	**2600**	**175**	**3900**
Military power	**140 (minimum)**	**4500**	**160**	**7650**
Military power	**160**	**6000**	**175**	**8600**

While the climb potential with maximum power is considerably less than that of the F-102A, it does exceed the contractural guarantees. The service ceiling of 48,800 feet is 1000 feet greater than guaranteed. The time to climb to 48,000 feet is 7½ minutes (including 1 minute for take-off and acceleration), which is 3.3 minutes faster than guaranteed. Sea level rate of climb with military power is 4500 feet per minute, 1500 feet less than guaranteed, and the aircraft requires 21 minutes to climb to 40,000 feet, 4½ minutes more than specified.

Visibility over the nose is satisfactory during a maximum power climb and is good during a military power climb. In all climbs, visibility to the left is good, to the left rear poor, and to the right rear entirely dependent on the instructor.

Maintaining the proper climb schedule is very difficult due to the high longitudinal break-out force (the production control system was not installed) and the large change of airspeed which results from minor pitch corrections. Trim must be constantly monitored during the climb due to the high elevator forces and the lateral unbalance which occurs rapidly with asymmetric fuel feeding. The best climb speed schedule above 35,000 feet is in the region of airframe buffet created by the large canopy. Climb performance is sacrificed in order to fly the aircraft at speeds below the buffet region.

Climb performance is presented in Appendix I, Figures 2 and 3.

Take-off and acceleration to climb schedule requires 1 minute and 450 pounds of fuel with maximum power and 2 minutes and 300 pounds of fuel with military power.

SUMMARY OF CLIMB PERFORMANCE

	MAXIMUM POWER			MILITARY POWER		
ALTITUDE FT.	WEIGHT LBS.	RATE OF CLIMB FT/MIN.	TIME TO CLIMB MIN.	WEIGHT LBS.	RATE OF CLIMB FT/MIN.	TIME TO CLIMB MIN.
10,000	28,000	12,000	0.8	28,000	4300	2.4
20,000	27,700	11,000	1.7	27,500	3400	5.0
30,000	27,500	8,800	2.8	27,200	2200	8.6
40,000	27,000	5,600	4.0	26,700	200	19.0
45,000	26,700	3,000	5.0			

■ level flight performance

Maximum Speeds: The TF-102A has a maximum speed capability of 0.975 Mach number at 35,000 feet and 24,000 pounds gross weight. This is considerably below the maximum speed of the F-102A (1.2 Mach number at 35,000 feet altitude at 23,000 pounds gross weight). The maximum speed capability decreases with an increase or decrease of altitude from 35,000 feet so that the performance guarantees are not met. The true speed of 550 knots (0.955 Mach number) at 40,000 feet does not meet the requirement of 568 knots (0.988 Mach number), nor does the 512 knots true speed (0.89 Mach number) at 47,000 feet fulfill the 534 knots (0.93 Mach number) guarantee.

The maximum Mach number with military power is 0.88 at 30,000 feet, and a decrease in speed occurs with higher or lower altitudes. Above 25,000 feet all maximum speeds occur in a region of buffet (reference page 7 "Canopy Vibration and Buffet").

The maximum speeds of the TF-102A are presented in Figure 5, Appendix I.

		MAXIMUM POWER		MILITARY POWER	
ALTITUDE FT.	WEIGHT LBS.	MACH NUMBER	TRUE SPEED KTS.	MACH NUMBER	TRUE SPEED KTS.
25,000	24,000	0.955	550	0.875	504
30,000	24,000	0.975	560	0.880	507
35,000	24,000	0.975	560	0.875	504
40,000	24,000	0.955	550	0.850	493
45,000	24,000	0.910	522	0.820	472

- *Maneuvering Capabilities:* The maneuvering capabilities of this aircraft are considerably inferior to those of the F-102A. The maximum load factor obtainable at 36,000 is approximately 1.7g at the optimum speed of 0.85 true Mach number (0.82 indicated Mach number). The maneuvering capabilities of the TF-102 are further impaired by the canopy buffet. The maximum turn rate at altitude is accomplished at a Mach number where buffet is evident in straight and level flight. Any addition in load factor is accompanied by an increase in buffet. (Reference page 7 "Canopy Vibration and Buffet".)

Thurst-limited turns at constant speed and altitude were accomplished at 36,000 feet. The data, corrected to standard conditions, is presented in Figure 4, Appendix I and summarized in the following table:

MANEUVERING CAPABILITIES

ALTITUDE: 36,000 FEET GROSS WEIGHT: 25,000 POUNDS

MACH NUMBER	MAXIMUM LOAD FACTOR - g	MAXIMUM TURN RATE-DEG/SEC.	MINIMUM TURN RADIUS - N. M.
0.70	1.40	2.3	2.6
0.85	1.67	2.9	2.5
0.95	1.20	0.7	3.2

- *Cruise and Power Required:* Cruise performance at recommended cruise speeds increases with altitude from 0.10 nautical air miles per pound of fuel at 10,000 feet to the optimum cruise conditions of 0.157 NAMPP in the region of 35,000 feet and 26,000 pounds gross weight. Cruise performance of the trainer is considerably less than that of the F-102A. The speed for recommended cruise in the trainer is 0.78 Mach number (at approximately 35,000 feet) compared to 0.905 Mach number for the F-102A, while the specific range of the F-102A exceeds that of the TF-102A by .083 NAMPP. Cruise data is presented in Appendix I, Figures 6, 7 and 8.

BEST CRUISE

ALTITUDE FT.	WEIGHT LBS.	MACH NUMBER	SPECIFIC RANGE NAMPP	P_{t_7}/P_{t_0}
10,000	27,597	0.570	0.100	1.67
25,000	26,552	0.680	0.144	1.96
35,000	26,076	0.775	0.157	2.34

Maximum endurance increases with altitude to approximately 25,000 feet. At this altitude a minimum fuel flow of approximately 2430 pounds per hour can be maintained in stabilized flight, giving an endurance of 2.47 minutes for each 100 pounds of fuel. Fuel consumption with Mach number is presented in Appendix I, Figure 18 and summarized in the following table:

MAXIMUM ENDURANCE

ALTITUDE FT.	WEIGHT LBS.	MACH NUMBER	FUEL FLOW LBS/HR.	ENDURANCE MIN/100 LBS FUEL	P_{t_7}/P_{t_0}
10,000	27,597	0.40	2720	2.205	1.55
25,000	26,552	0.56	2430	2.470	1.93
35,000	26,076	0.63	2510	2.390	2.37

- *Drag:* Aerodynamic drag of the TF-102A is significantly higher than for the F-102A, especially in the high speed range and the drag rise is encountered at a lower Mach number. The drag rise is encountered at 0.875 Mach number on the F-102A but is encountered at 0.70 on the trainer. This drag increase, which stems from the wider cockpit and nose section, is the basic cause of the limited performance capabilities of the trainer. Drag data is presented in Appendix I, Figures 9 through 13.

DRAG DATA SUMMARY

CLEAN CONFIGURATION

ALTITUDE FT.	W/δ LBS.	WEIGHT LBS.	MINIMUM DRAG FN/δ LBS.
10,000	40,135	27,597	3,900
25,000	71,570	26,552	7,100
35,000	110,870	26,076	10,900

level flight handling characteristics

Dynamic Stability: The dynamic longitudinal stability of the TF-102A is satisfactory in the level flight speed ranges, both with and without the pitch damper operating. Without the damper, induced oscillations persist for two cycles. These oscillations damp out in one cycle with the pitch damper operating. Time histories of induced oscillations are presented in Figures 26, 27 and 28 of Appendix I.

The dynamic directional stability is satisfactory.

Induced oscillations persist for two and one-half to three cycles throughout the level flight speed range of the basic airplane. With the yaw damper operating, these directional oscillations damp in one cycle. Time histories of these directional oscillations are presented in Figures 29 through 35 of Appendix I.

The dynamic lateral stability was unsatisfactory. Lateral oscillations induced at the same time as directional oscillations persisted after the directional oscillations had damped. In most cases, the oscillations continued until stopped by the pilot. High break-out forces in the lateral control created a tendency to overcontrol the lateral corrections. Use of the yaw damper did not appreciably affect the lateral oscillations.

- *Static Stability:* The static directional stability is positive under all flight conditions tested. The rudder force required to maintain a steady sideslip increases as Mach number is increased. Above 0.8 Mach number and 5 degrees of rudder travel, the sideslip did not appear to be proportional to rudder displacement. A force lightening was not encountered, however. Zero sideslip is slightly difficult to maintain in 45 degree banked turns in the power approach configuration at 180 knots IAS. The rudder power was adequate to reduce the yawing acceleration to zero under all conditions tested.

The static longitudinal stability is positive in the cruise configuration. The canopy buffet is annoying to the pilot but does not affect stability. Quantitative static longitudinal data is presented in Appendix I, Figure 24.

The static lateral stability is basically satisfactory with positive dihedral effect exhibited in all steady state sideslips. Due to high aileron break-out forces, precise lateral control is difficult to maintain. The rapid trim rate causes a tendency to over-trim.

- *Maneuvering Characteristics:* The gradient of elevon position with load factor in maneuvering flight is positive throughout the level flight speed region under the load factors tested. The gradient is linear at low altitudes; at medium and high altitudes (above 30,000 feet) the gradient decreases with increased load factor but does not exceed the requirements of Air Force Specification MIL-F-8785 (ASG). Force instrumentation was not available during Phase II tests, but a qualitative analysis indicated an excessive stick force gradient.

Precision turns are difficult to maintain due to the tendency to overcontrol the bank angle. Even with the yaw damper operating, constant bank angles are difficult to maintain; with damper off, precise lateral control is not possible.

■ lateral control

The inertial coupling problem does exist for the TF-102A airplane. The present restrictions of limiting roll maneuvers to the positive quadrants and one-half aileron deflection are adequate to avoid any serious inertial coupling effects. However, the high roll rate of the TF-102A makes exceeding these limits easy. The large vertical tail is prescribed as a partial remedy for this problem and will be incorporated on production models. The rate of roll within the restrictions is adequate under all flight conditions.

The rolls accomplished in the cruise configuration from 60 degrees left to 60 degrees right bank did not produce appreciable adverse yaw. These rolls were accomplished at 35,000 feet using aileron angles of as much as 6½ degrees (approximately one-half available aileron) and at various speeds from 0.72 to 0.91 Mach number. The yaw damper and turn coordinator were on and an attempt was made to maintain zero rudder deflection during the roll maneuvers. In the same configuration 360 degree rolls were accomplished at 0.73 Mach number at 35,000 feet with the yaw damper and turn coordinator engaged (Figures 40 and 41). During these rolls the adverse yaw built up to the point where the roll rate was reduced appreciably. In the power approach configuration, with yaw damper and turn coordinator functioning, the sideslip due to adverse yaw was not excessive. With 7 degrees of aileron, the maximum used in this configuration, the roll rate was 70 degrees per second and the sideslip was 4 degrees. The roll potential (helix angle vs aileron deflection) is presented graphically in Appendix I, Figure 53. Roll time histories are presented in Figures 36 through 52.

■ supersonic dive characteristics

The TF-102A airplane will attain supersonic speeds in a slight dive. The mild nose down pitch encountered can easily be trimmed out. A moderate pitch-up occurs at 0.9 indicated Mach number when the aircraft is returned to subsonic flight.

No unusual control problems are encountered during the dive or recovery from the dive. A time history of a high Mach dive is presented in Appendix I, Figure 54.

canopy vibration and buffet

The buffet encountered early in the Phase I program is greatly decreased by a double row of vortex generators on the canopy (reference joint messageforms in Appendix II). However, canopy buffet remains as a definite annoyance. The buffet is encountered in unaccelerated conditions at 0.80 indicated Mach number, increasing to maximum intensity at 0.85 indicated Mach number, and then decreasing in intensity with further increases in speed until it disappears at approximately 0.90 indicated Mach number. This buffet is experienced from the speed for best climb through the maximum speeds with military power at all altitudes above 25,000 feet. Increased load factors aggravate the buffet condition. At low Mach numbers, buffet occurs with slight increases of load factor. At speeds where buffet exists under 1g conditions, it increases rapidly with increases in load factor.

This buffet, created by flow separation over the canopy, is annoying since the normal operating speeds for this aircraft occur in this buffet region. The optimum speeds for turning flight at altitude (0.85 true Mach number) are in the region where buffet is encountered in unaccelerated flight and is magnified by the increased load factor of turning flight. The intensity of this buffet is sufficient to distract the student pilot from the primary flight problem.

approach and landing

The handling characteristics of the TF-102A during approach and landing differ only slightly from the F-102A. The power settings required in the pattern are approximately 2 percent RPM higher than for the F-102A, and all pattern speeds, except gear extension speed, should be increased 5 knots to provide good control and reduce rate of sink. The buffet created by gear extension is more pronounced than in the F-102A. Due to the shorter nose section, the nose high attitude appears exaggerated at touchdown. Visibility over the nose is good. Maximum braking is not recommended because of the delay between pedal release and brake release. This delay of 1 to 1½ seconds does not allow the pilot to adequately control the aircraft and correct for skidding tires.

The recommended approach and landing technique, with the exception of the variations noted, is the same as for the F-102A. Landing performance is presented graphically in Appendix I, Figure 14 and summarized in the following table:

LANDING PERFORMANCE

GROSS WEIGHT 23,000 POUNDS

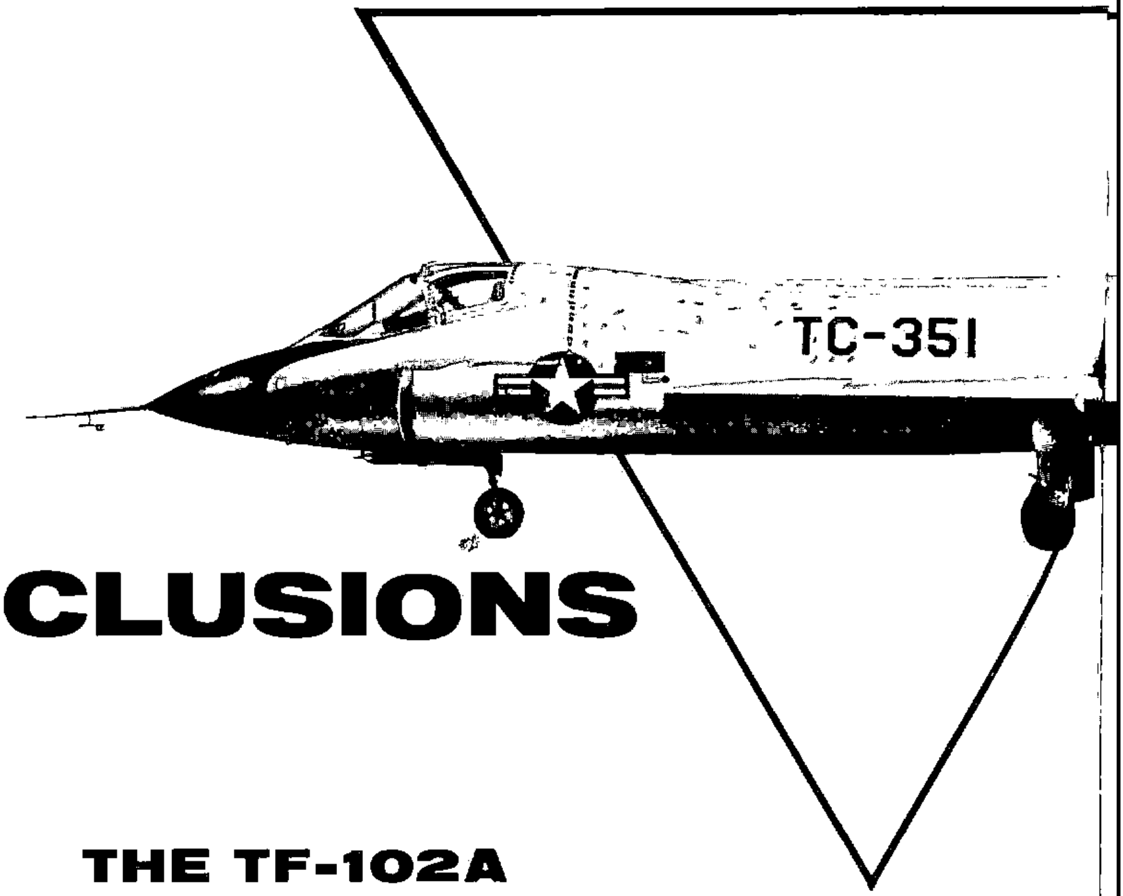
BRAKING CONDITIONS	INDICATED AIRSPEED AT 50 FEET - KTS.	DISTANCE FROM 50 FEET TO STOP - FT.	INDICATED AIRSPEED AT TOUCH-DOWN - KTS.	GROUND ROLL FROM TOUCH-DOWN - FT.
NORMAL BRAKING WITHOUT DRAG CHUTE	**150**	**7800**	**130**	**6000**
NORMAL BRAKING WITH DRAG CHUTE	**150**	**5500**	**130**	**4100**
MAX BRAKING WITH DRAG CHUTE	**150**	**4500**	**130**	**3000**

aircraft systems

Airspeed System: The position error associated with the airspeed system of the TF-102A is inordinately high in the speed range from 0.85 to 1.0 indicated Mach number, requiring a maximum airspeed correction of approximately 20 knots and a maximum altitude correction of approximately 2500 feet at 35,000 feet altitude.

The airspeed calibration was obtained from stabilized pacer points and is presented (with a description of the nose boom) in Appendix I, Figures 22 and 23.

- *Engine and Afterburner:* Operation of the J57-P-23 turbo-jet engine and afterburner was satisfactory. Compressor stalls were not experienced; however, during ground operation mild pulsations similar to compressor stalls were frequently experienced when advancing the throttle from idle position. This condition does not require throttling back as a corrective measure.



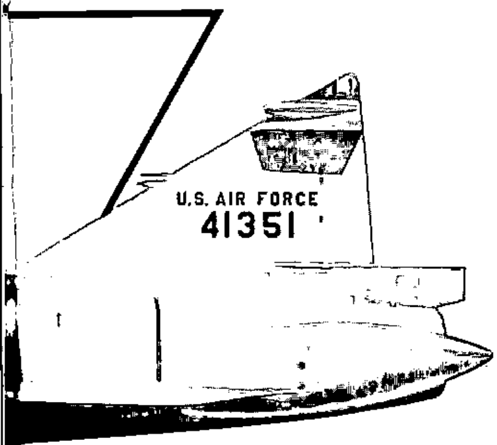
CONCLUSIONS

THE TF-102A

satisfactorily fulfills the requirements of a transitional trainer for delta wing aircraft. The aircraft handling qualities closely approximate those of the F-102A in take-offs, climbs, slow speed characteristics, instrument flying, and landing. The performance capabilities, however, differ substantially from those of the F-102A aircraft. The canopy design and wider fuselage result in increased drag and a one g buffet that is intensified by maneuvering flight. This causes the TF-102A speed-altitude performance to be significantly inferior to that of the F-102A. The buffet will tend to limit the utility of the TF-102A as a weapon system proficiency trainer to practice intercept missions below 40,000 feet and 0.8 Mach number. The performance capabilities of the TF-102 are not compatible with design concepts of the modern tactical weapons system. Consequently, the aircraft should receive only supernumerary status in the tactical inventory.

The performance guarantees are met or exceeded in take-off and climbs with maximum power. The climb capability and the maximum level flight speeds with military power fall short of guarantees. Maximum power will be required for most operations above 40,000 feet, resulting in short flights.

Handling characteristics are generally satisfactory, but minor control problems are caused by the excessive break-out forces. Within the present flight restrictions, the inertial coupling tendencies are not dangerous, but the high roll rate of the aircraft makes it easy to exceed these limits.



RECOMMENDATIONS

Efforts to reduce canopy buffet should be continued. Close quality control must be maintained to insure that aircraft are not delivered with buffet worse than that encountered on the Phase II test aircraft (No. 1351) with vortex generators installed. The following recommendations should be accomplished as soon as possible:

1. *Reduce the problem of inertial coupling during roll maneuvers.*
2. *Provide for better damping of the lateral oscillations resulting from abrupt lateral control displacements.*
3. *Reduce the control break-out forces in the lateral and longitudinal directions.*
4. *Improve main gear braking control to provide brake release simultaneously with pedal release.*
5. *Reduce the force required to lower the canopy.*
6. *Reduce the force necessary to operate the landing gear handle.*
7. *Relocate the rudder trim switch to improve accessibility.*
8. *Improve ease of operation of the ram air turbine handle.*
9. *Provide entrance ladders that attach to the aircraft to eliminate a ground safety hazard.*

APPENDIX I

data analysis methods and performance and stability curves

drag computations

The manufacturer's nozzle coefficient curve was used as a basis for extrapolating measured thrust data obtained on the thrust stand. Using this curve and the expressions referred to in the thrust calibration discussion of this Appendix, a plot of corrected gross thrust versus engine pressure ratio was made encompassing the complete range of pressure ratios encountered in flight. This curve was then utilized for drag computations.

Net thrust as used in this report is defined as gross thrust less ram drag. Engine airflow for the ram drag computations was obtained from Pratt and Whitney's curve No. 17145, dated 9 May 1955. The ram efficiency curve is presented in Appendix I, Figure 21.

The drag polars presented in Figures 12 and 13 were obtained from stabilized level flight data flown at a constant weight-pressure ratio for each altitude so that weight corrections were not necessary. The elevon trim positions are indicated for each test point in the above mentioned Figures.

No attempt was made to define or compute detailed drag items as trim drag.

standard day power available

An engine pressure ratio bias curve (P_{t_7}/P_{t_0} versus t_{ic}) was used to determine power available with military or maximum power on a standard day. Due to the pressure and temperature lags during climbs, two bias curves were used; one for stabilized flight conditions and the other for climbs. The two curves are presented in Figure 17 of this Appendix. Power available for climbs was obtained by entering the bias curve with standard day temperature to obtain standard day pressure ratio (P_{t_7}/P_{t_0}). Thus, using test and standard day values of P_{t_7}/P_{t_0}, an incremental thrust was obtained from the thrust calibration curves.

For stablized data, such as maximum speeds and turn capabilities, the bias curve was entered with standard day temperature to obtain standard P_{t_7}/P_{t_0} which was then converted to P_{t_7}/P_0 and used to obtain standard gross thrust. The standard gross thrust was subsequently used in conjunction with a plot of corrected gross thrust versus Mach number to determine the maximum level flight speed at the standard gross thrust computed.

The level flight speed power data was corrected to a standard weight to obtain thrust required versus equivalent airspeed by using a plot of $\triangle C_L^2/\triangle C_D$ versus Mach number.

take-offs

Take-off data was corrected to standard day, no-wind conditions at sea level as outlined in AFFTC Technical Note R-12 with the use of equations (4-1) and (4-4) on pages 7 and 8 respectively.

continuous climbs

Continuous climbs were corrected to standard conditions at test gross weight (weight variations were negligible) as outlined in AF Technical Report No. 6273, pages 5-1 through 5-17 and using equations (5.203) and (5.206).

maneuvering capabilities

Maneuvering capabilities were determined by using true airspeed and normal acceleration obtained from maximum power turns (constant speed and altitude) in conjunction with the following expression:

$$n = \sqrt{0.1087 \left(\frac{V_T}{t}\right)^2 + 1}$$

n = load factor (normal acceleration), g
V_T = true airspeed, knots
t = time to turn 360 degrees, seconds

This data was corrected to standard conditions with the use of the drag polar, test and standard day drag coefficients and test lift coefficients. The gross thrust, as discussed in paragraph 2, was corrected to net thrust using the engine manufacturer's air flow curves.

level flight

Stabilized speed-power data was obtained at a constant weight-pressure ratio thus eliminating the need for weight corrections. The data is presented with the engine pressure ratio (turbine discharge pressure/free stream total pressure) as the power parameter. A curve of pressure ratio versus engine speed is presented for the convenience of those who wish to make the conversion to rpm.

landings

The landing data was corrected to standard conditions as outlined in AF Technical Report No. 6273, pages 6-17 through 6-19, using equations (6.407) and (6.408).

thrust calibration

Engine thrust calibration data was corrected to standard, sea-level conditions as outlined in AF Technical Report No. 6273, pages 3-32 through 3-35 and using equations (3.10.06) and (3.10.07).

temperatures

The recovery factor of the temperature sensing element was found to be 0.95 under stabilized conditions.

performance and stability curves

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4	TURN CAPABILITIES	18
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FIGURE

TAKE-OFF PERFORMANCE

TF-102A USAF S/N 54-1351

SEA LEVEL STANDARD CONDITIONS

GROSS WEIGHT 23,300 LBS

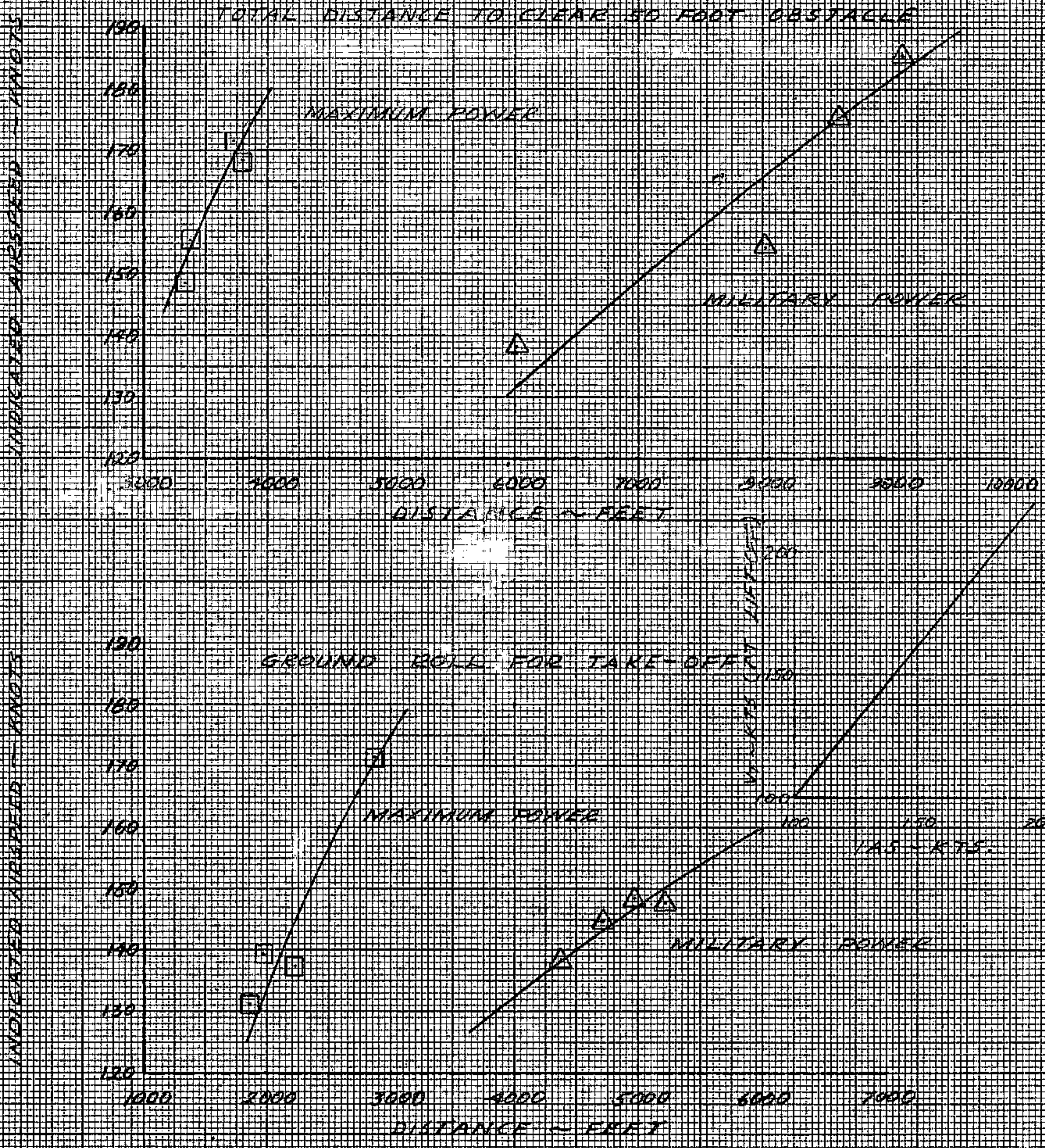
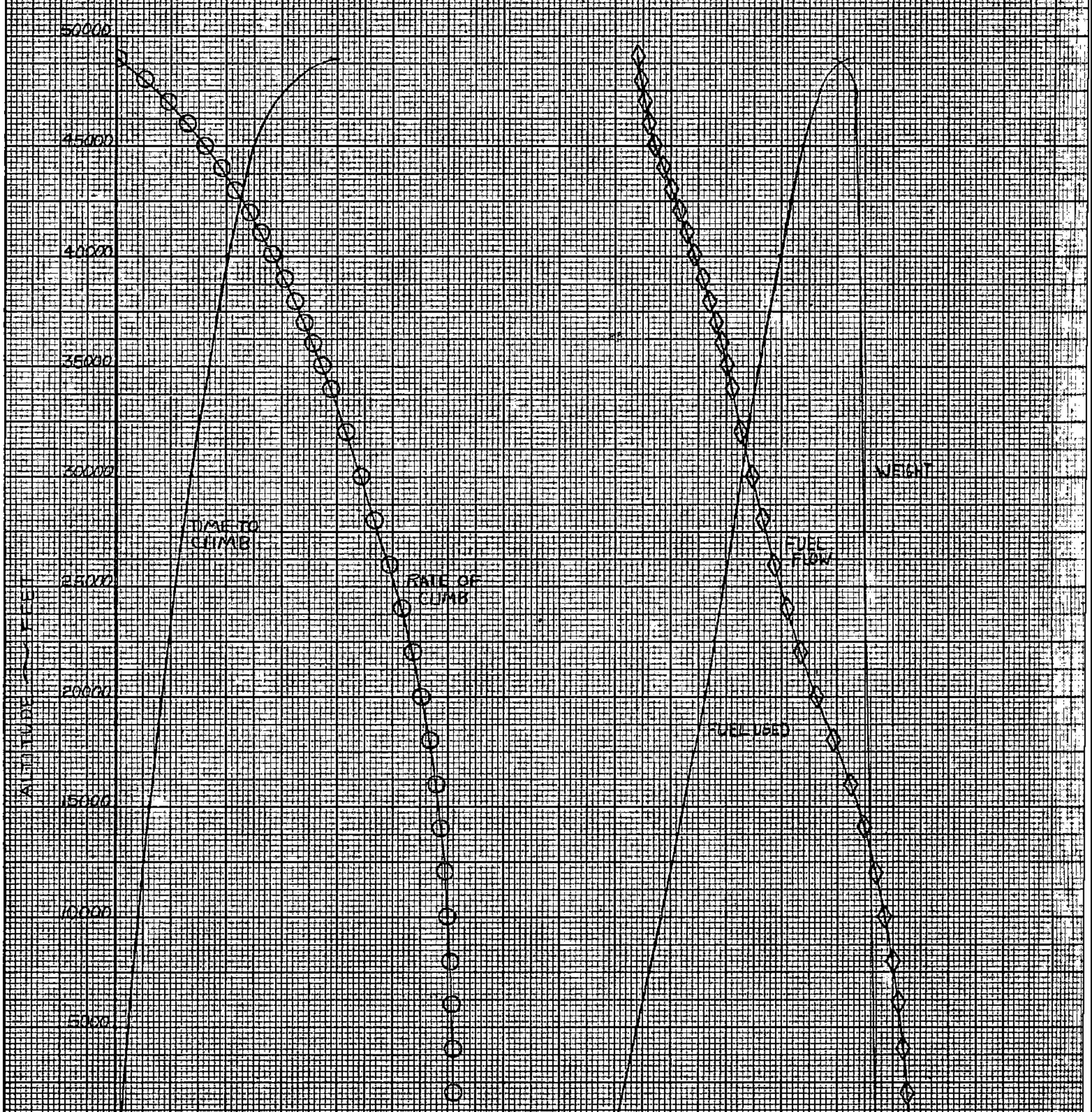
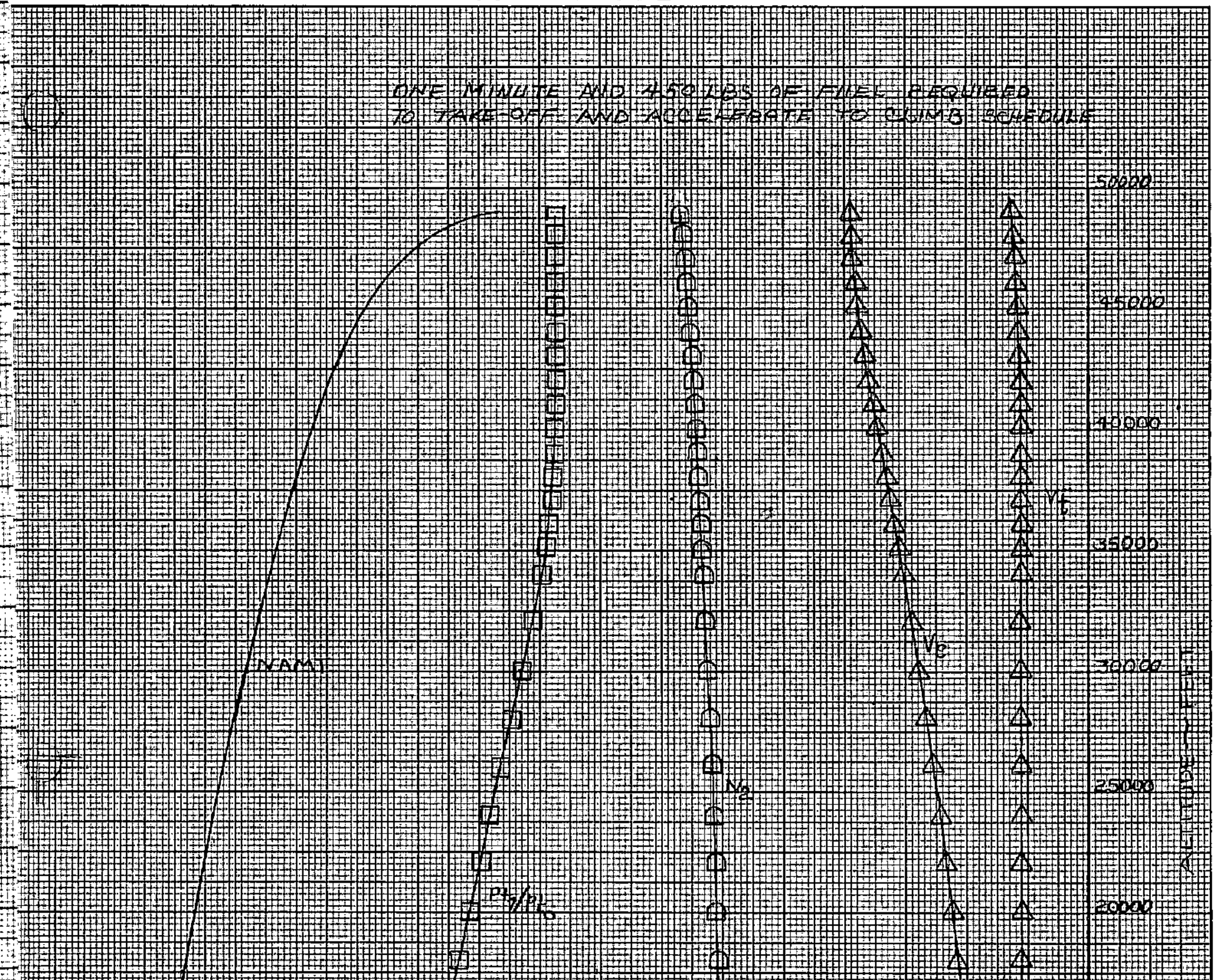


FIGURE 2
CLIMB PERFORMANCE
TF-102A USAF S/N 54-1351
MAXIMUM POWER





ONE MINUTE AND 450 LBS OF FUEL REQUIRED TO TAKE-OFF AND ACCELERATE TO CLIMB SCHEDULE

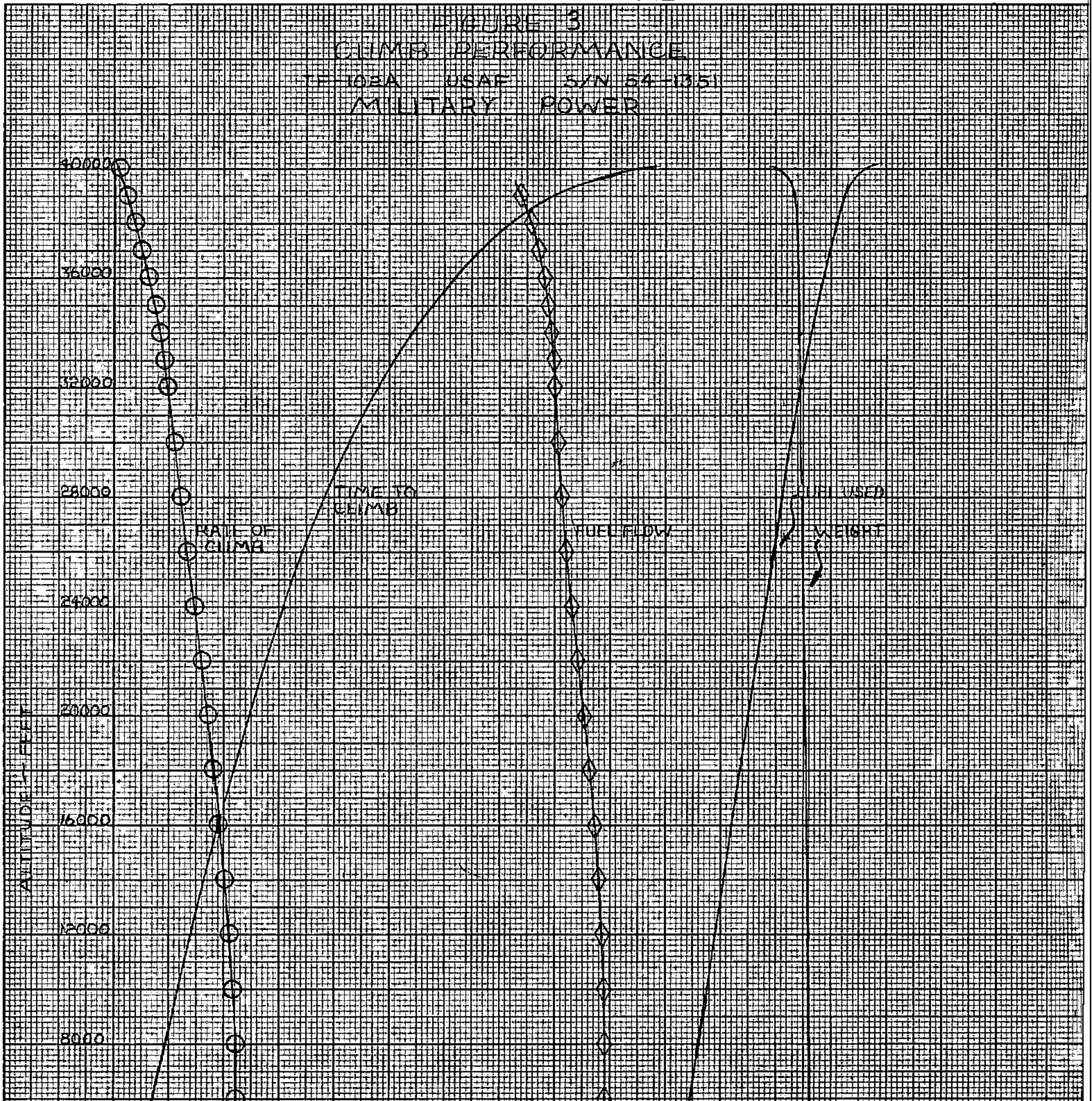
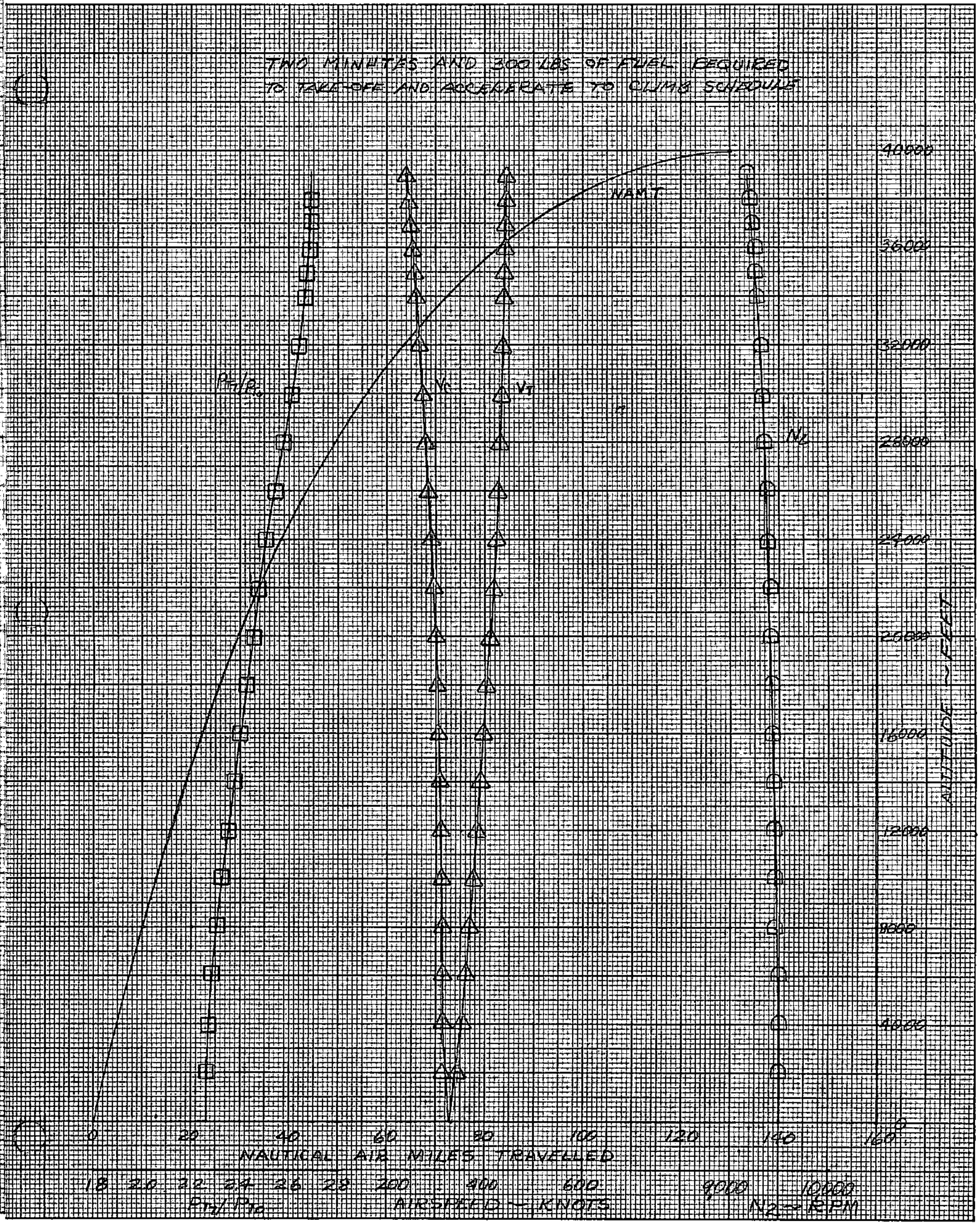


FIGURE 3
CLIMB PERFORMANCE
TF-102A USAF S/N 54-1351
MILITARY POWER



TWO MINUTES AND 300 LBS OF FUEL REQUIRED TO TAKE-OFF AND ACCELERATE TO CLIMB SCHEDULE

FIGURE 4

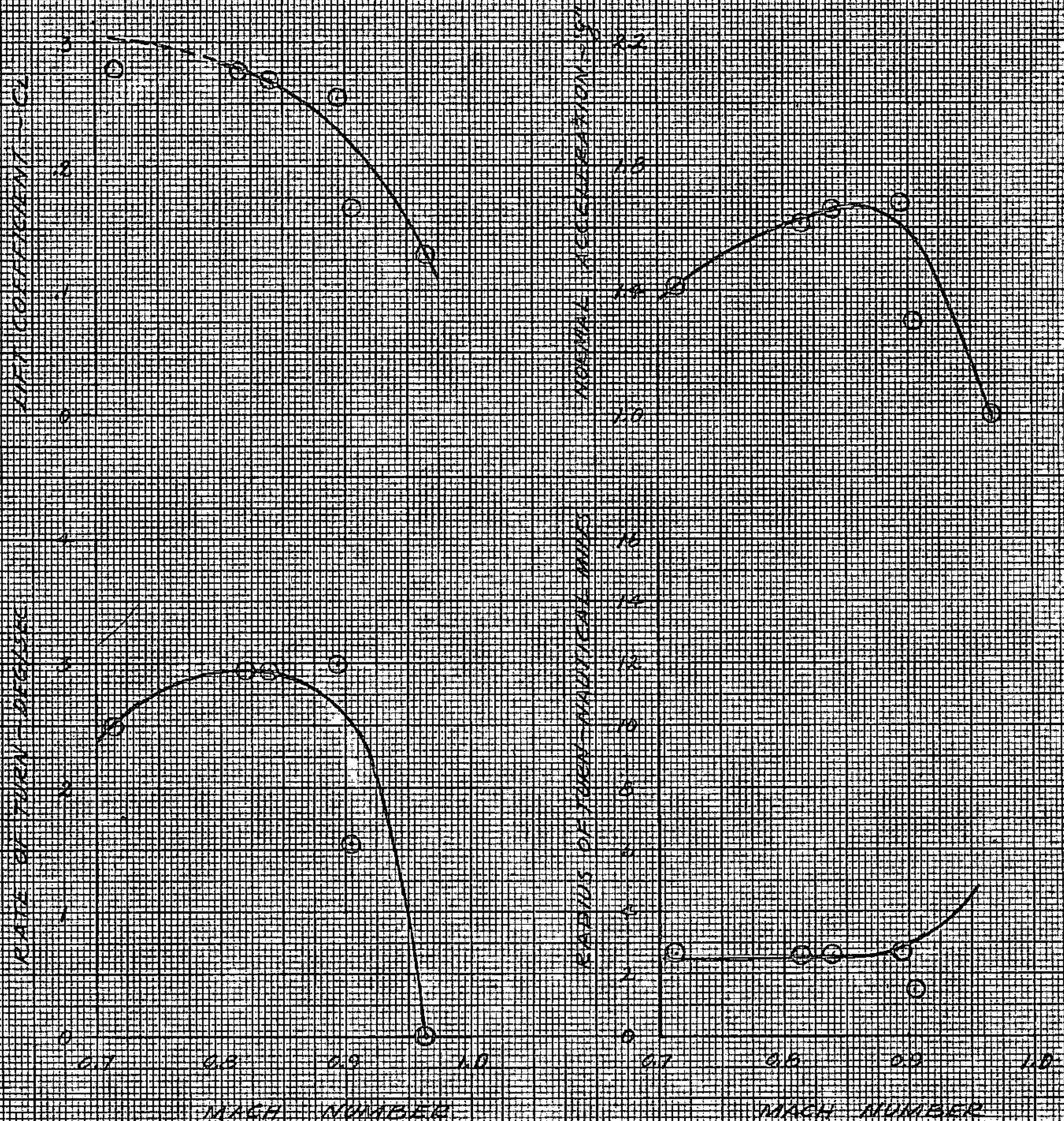
TURN CAPABILITIES

TF-102A USAF 54-1351

ALT. 36000 FT.

WT. 25000 LBS.

NOTE: TURN CAPABILITIES WITH CONSTANT SPEED AND ALTITUDE AT MAXIMUM POWER ON STANDARD DAY



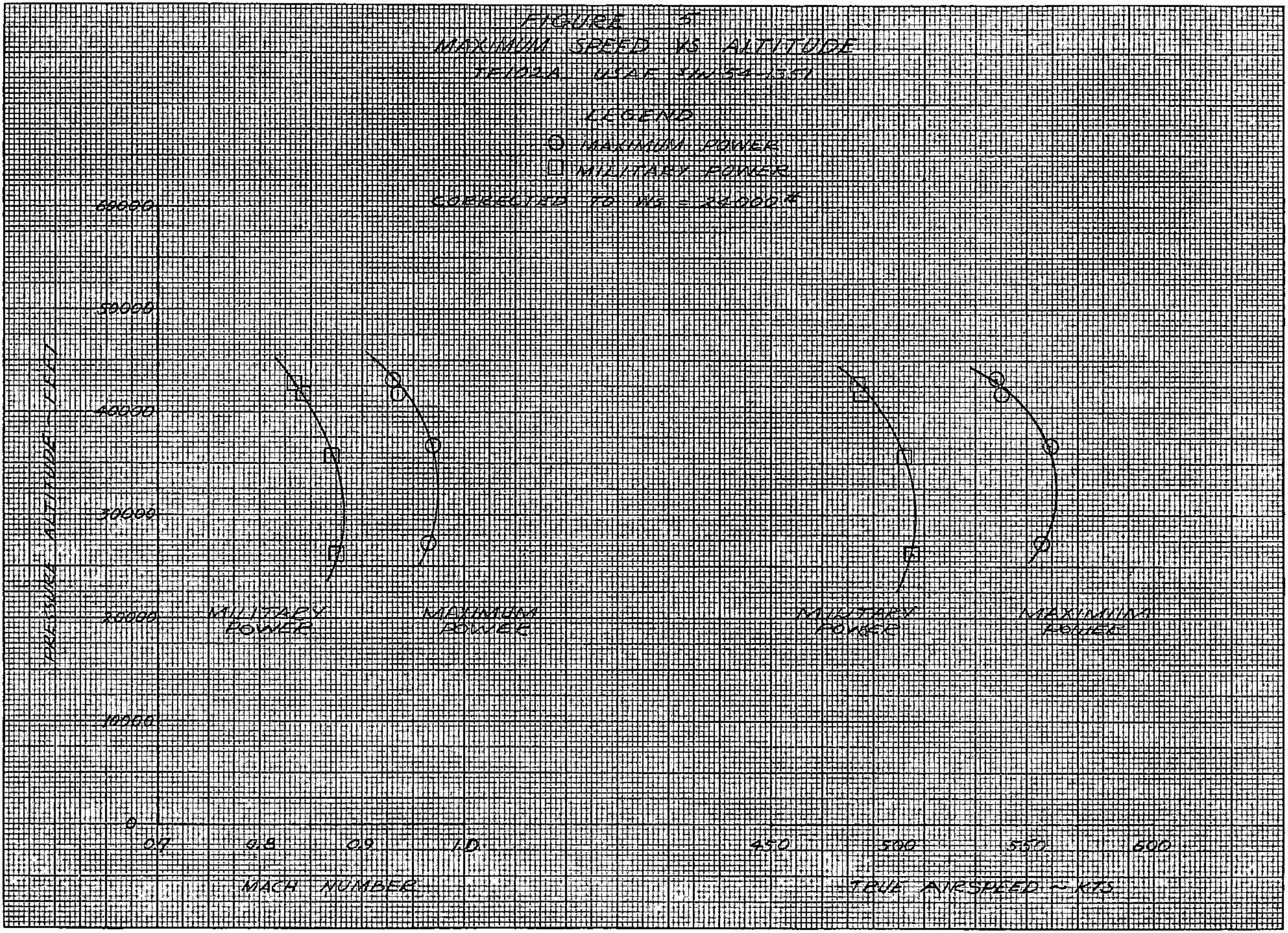


FIGURE 5
MAXIMUM SPEED VS ALTITUDE
TF102A USAF SN 54-1351

FIGURE 6
LEVEL FLIGHT PERFORMANCE
TF-102A USAF S/N 54-1351

SYMBOL	W/δ	ALTITUDE	WT.
○	40,170	10,000	27,600
△	71,570	25,000	26,550
□	110,870	35,000	26,080

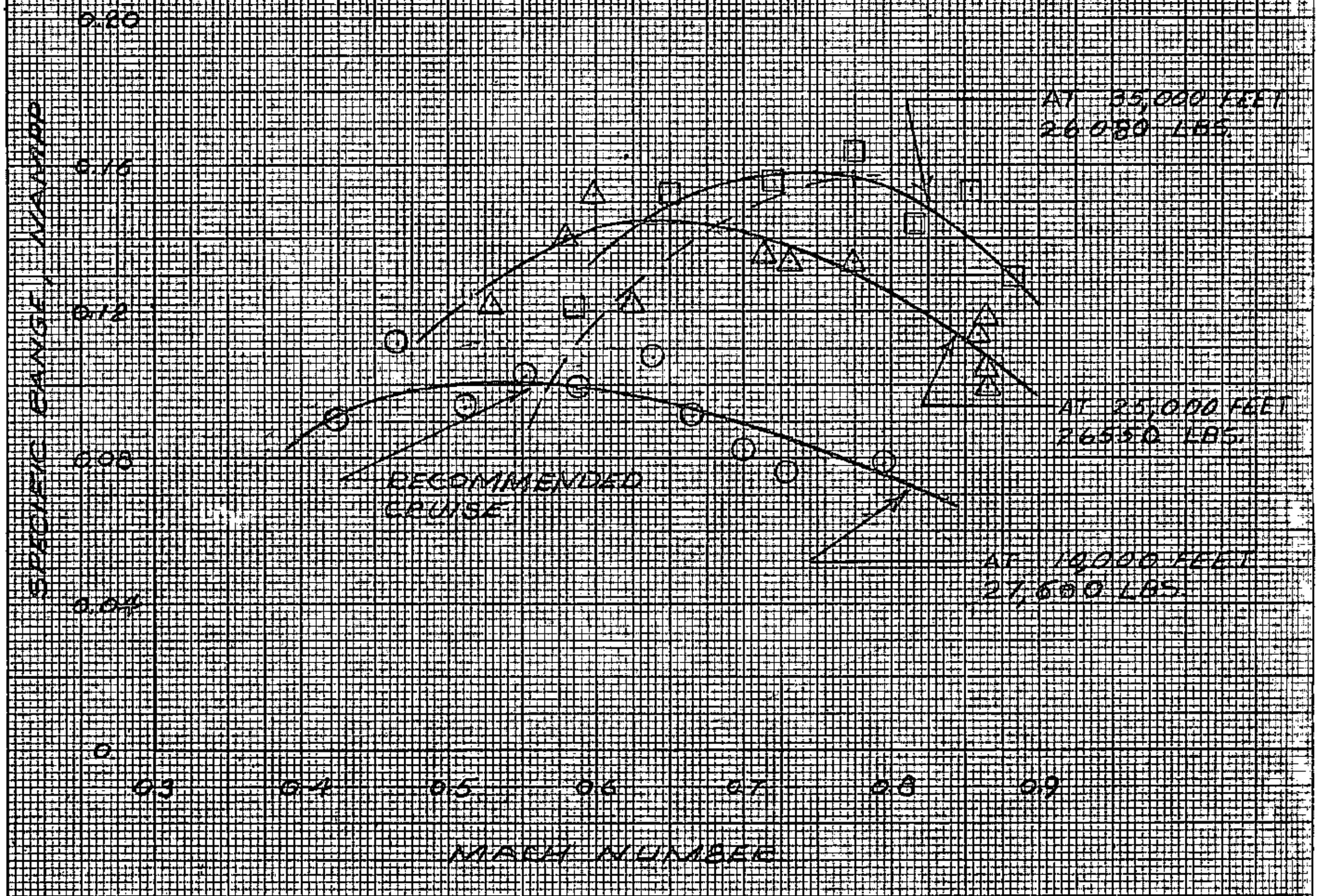
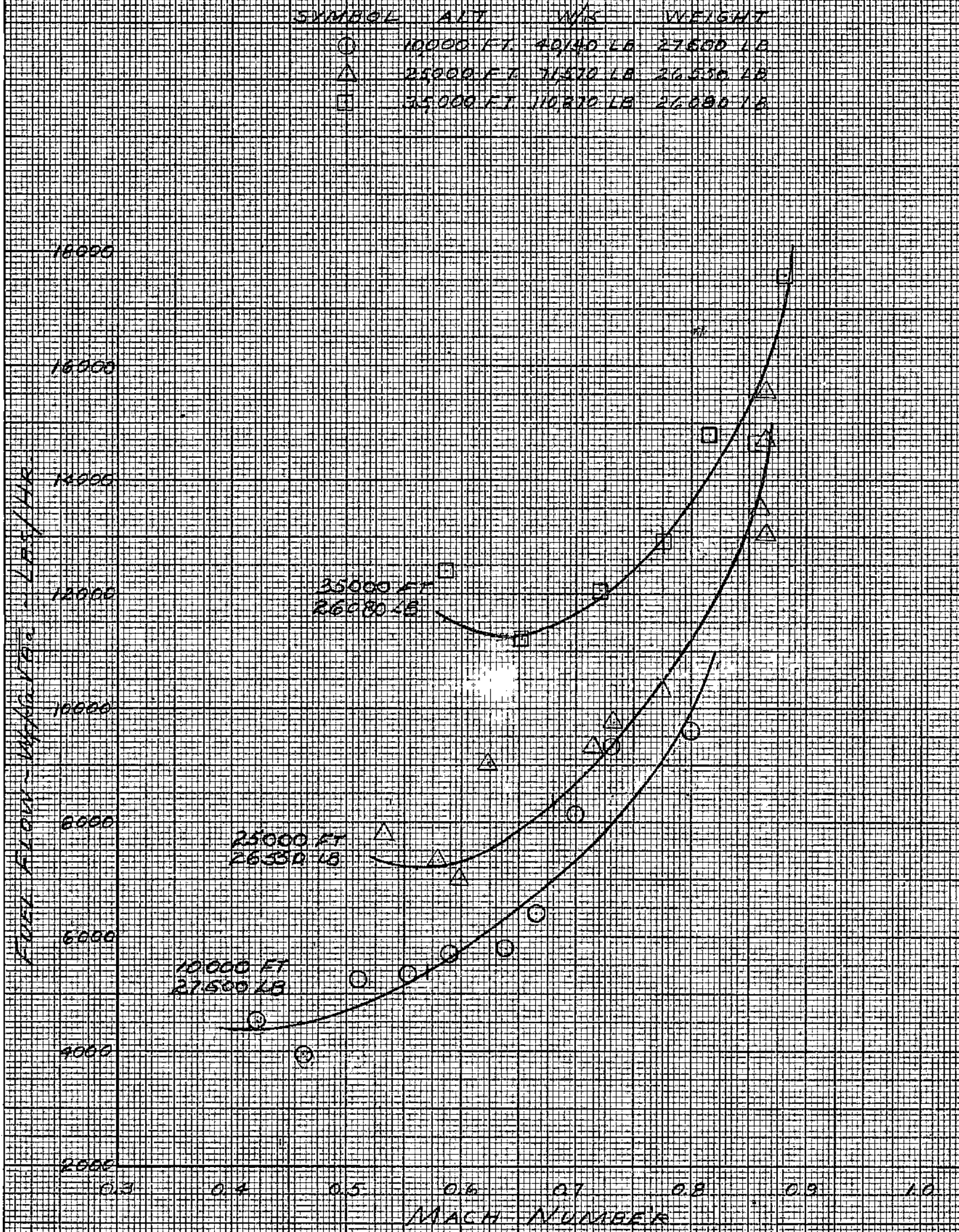


FIGURE 7:
LEVEL FLIGHT PERFORMANCE
TF-102A USAF S/N 54-1351



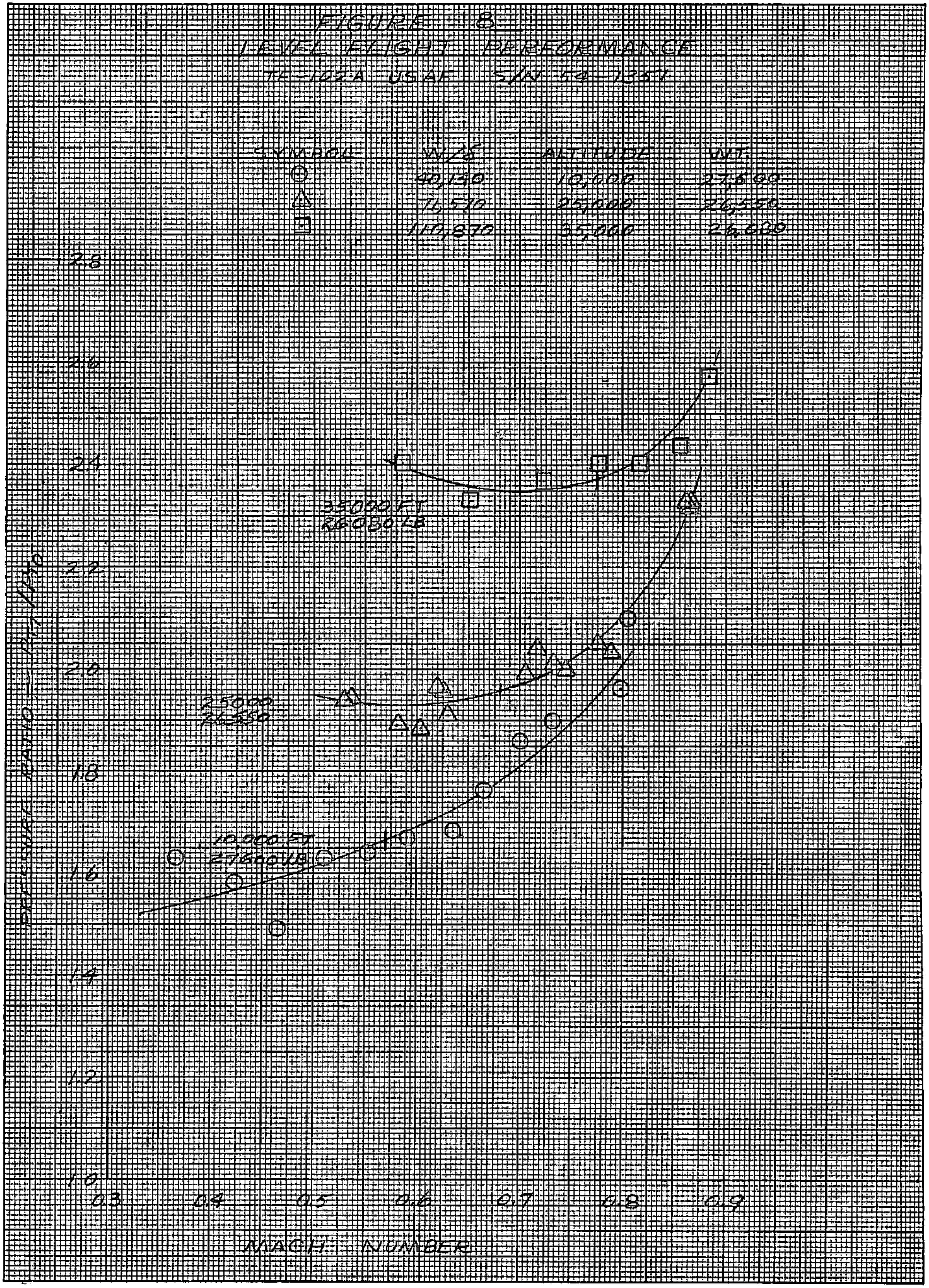


FIGURE 8
LEVEL FLIGHT PERFORMANCE
TF-102A USAF S/N 54-1351

FIGURE 9

LEVEL FLIGHT PERFORMANCE

TF-102A USAF S/N 54-1351

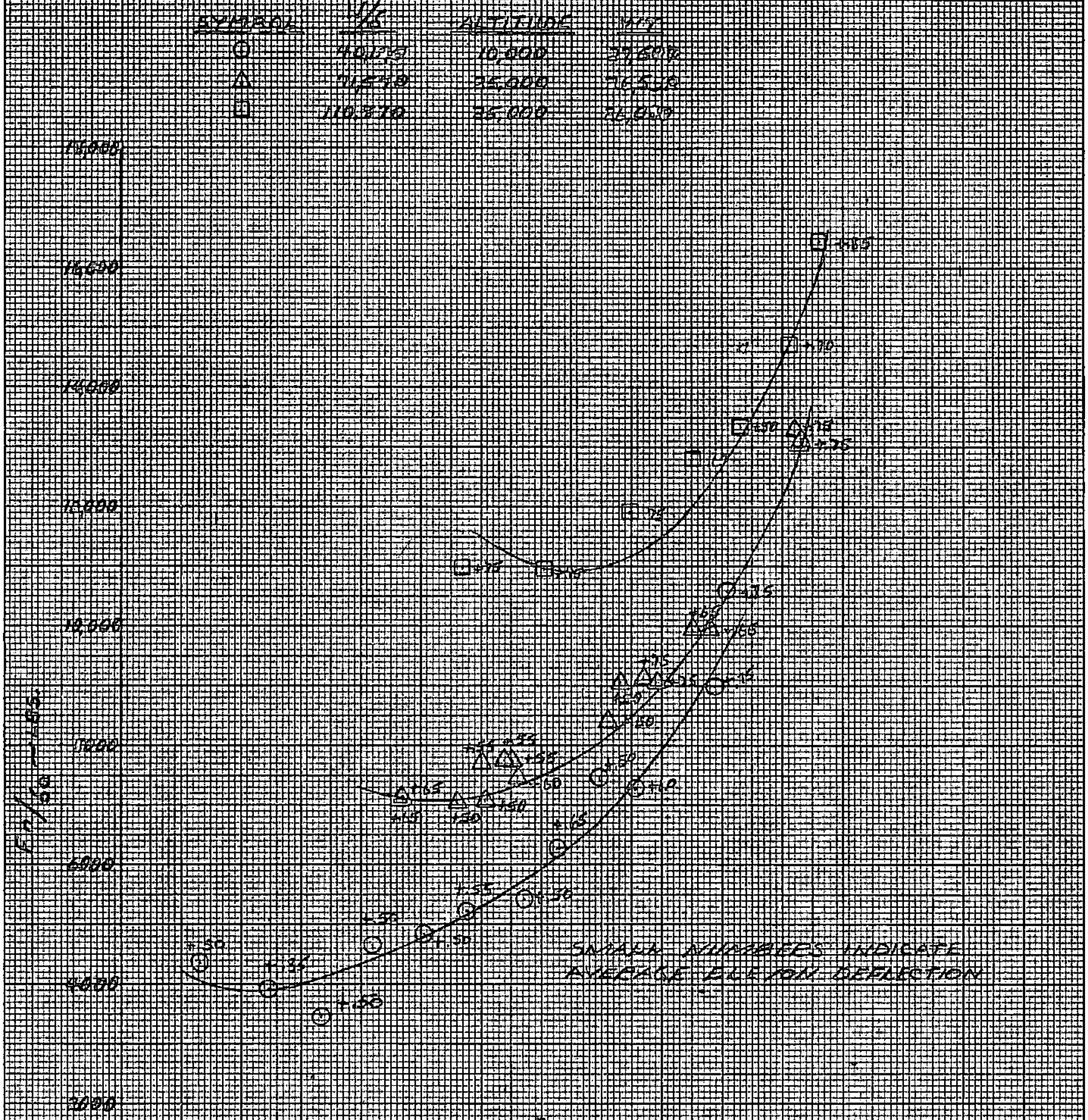


FIGURE 10

LEVEL FLIGHT PERFORMANCE

STANDARD NET THRUST VS EQUIVALENT AIRSPEED

YF102A USAF S/N 54-1351

SYMBOL	W/δ	ALTITUDE	WT
○	40,140	10,000	23,600
△	71,570	25,000	26,550
□	110,870	35,000	26,680

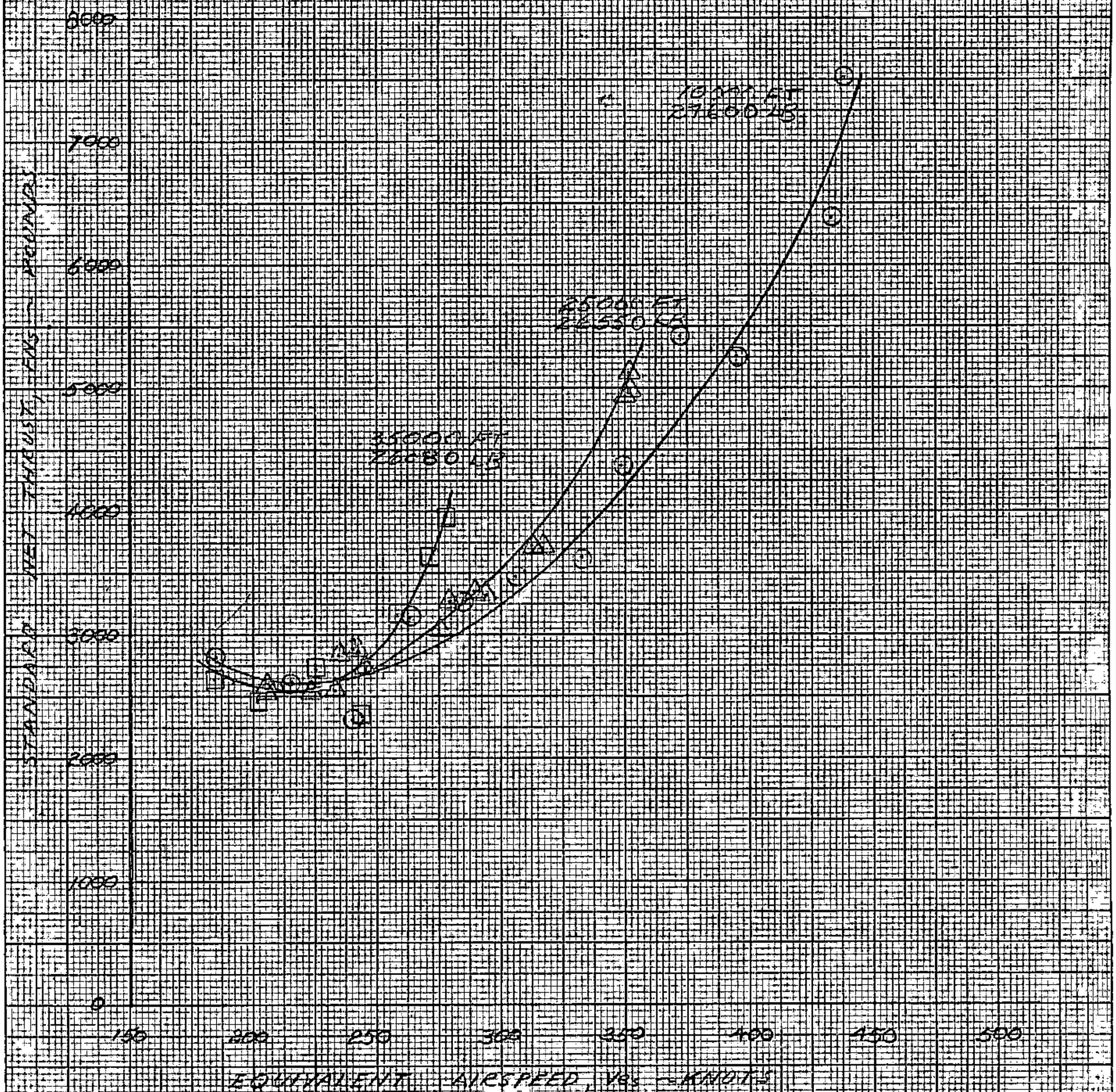


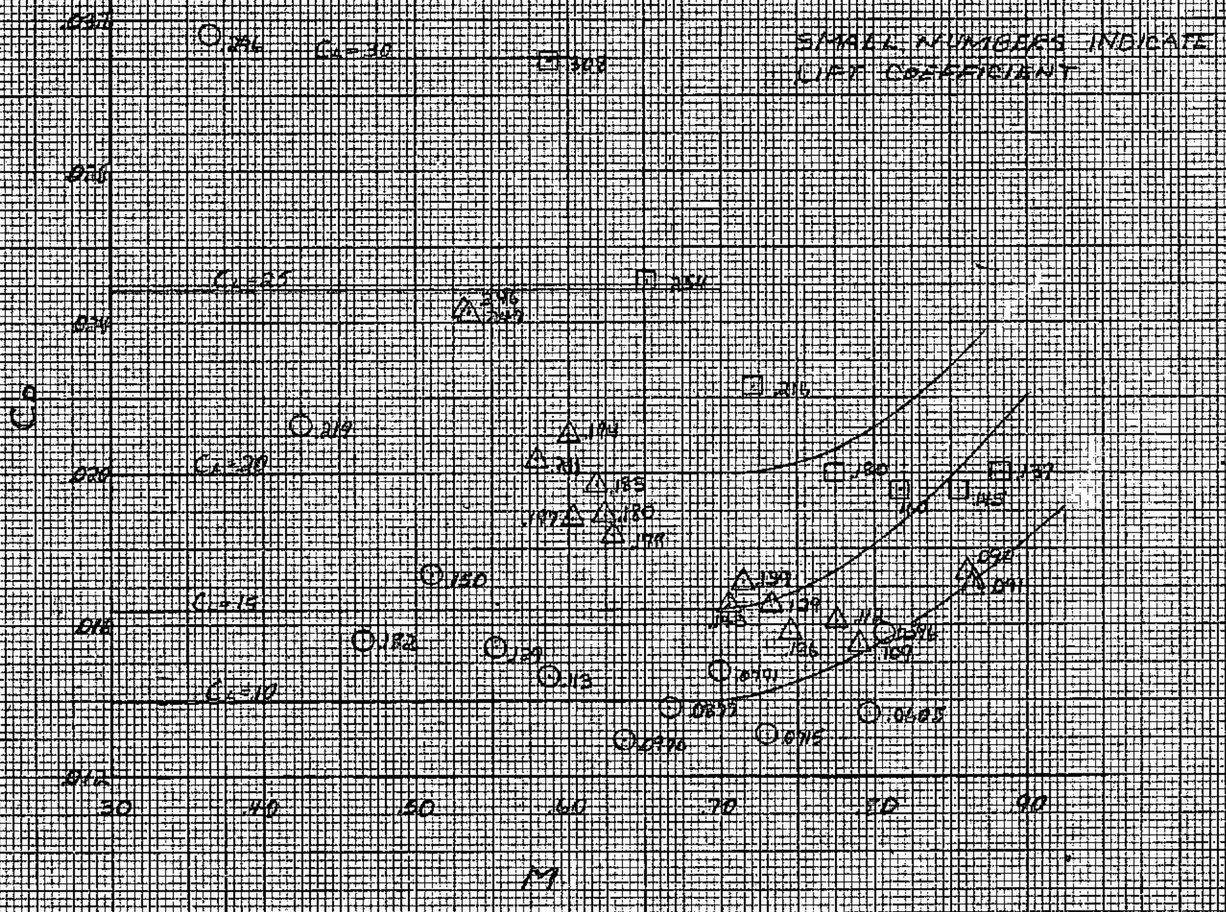
FIGURE 11

LEVEL FLIGHT PERFORMANCE

LIFT COEFFICIENT VS MACH NO.

TF-102A USAF S/N 54-1351

SYMBOL	W/δ	ALTITUDE	WT
○	40,136	10,000	27,600
△	71,570	25,000	26,550
□	110,870	35,000	26,000



SMALL NUMBERS INDICATE LIFT COEFFICIENT

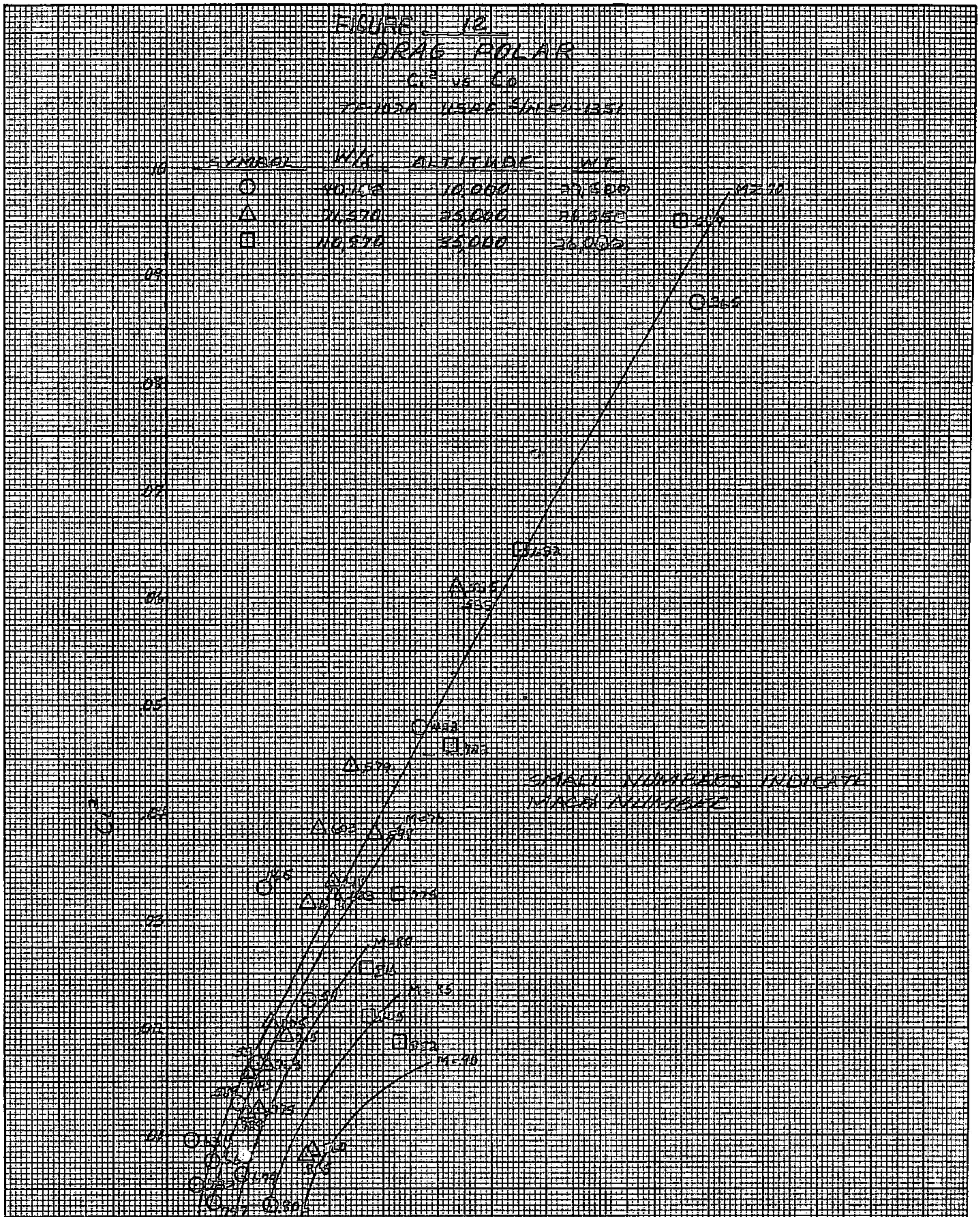


FIGURE 12
DRAG POLAR
C_L^2 vs C_D
TF-102A USAF S/N 54-1351

FIGURE 13

LEVEL FLIGHT PERFORMANCE

LIFT COEFFICIENT VS DRAG COEFFICIENT

TF-102A USAF S/N 54-1351

SYMBOL	W/δ	ALTITUDE	W
○	40,045	10,000	27,550
△	71,570	25,000	26,550
☐	110,770	35,000	26,080

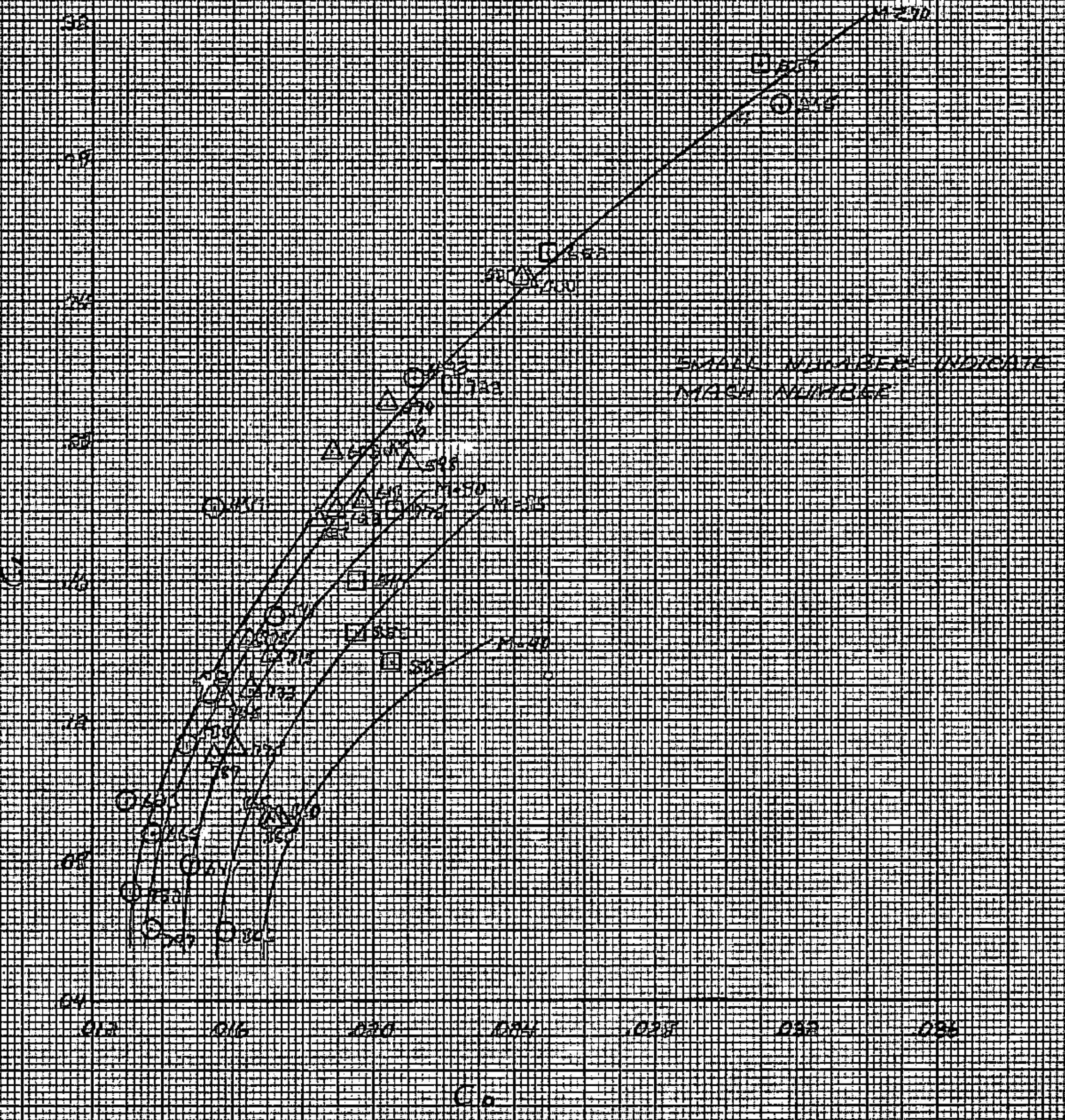
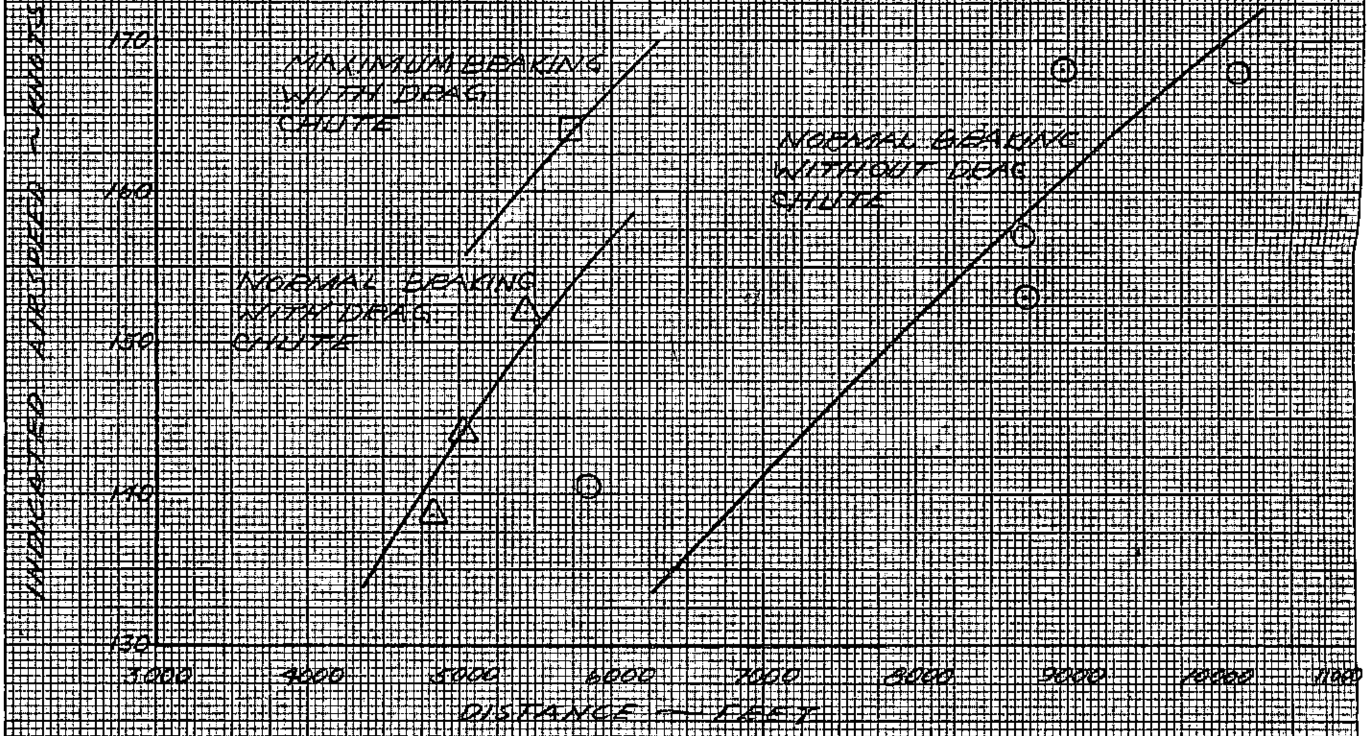


FIGURE 14
LANDING PERFORMANCE
TF-102A USAF S/N 54-1351
SEA LEVEL STANDARD CONDITION
GROSS WEIGHT 23000 LBS

TOTAL DISTANCE TO CLEAR 50 FOOT OBSTACLE
LAND, & STOP



GROUND ROLL FROM TOUCH DOWN TO STOP

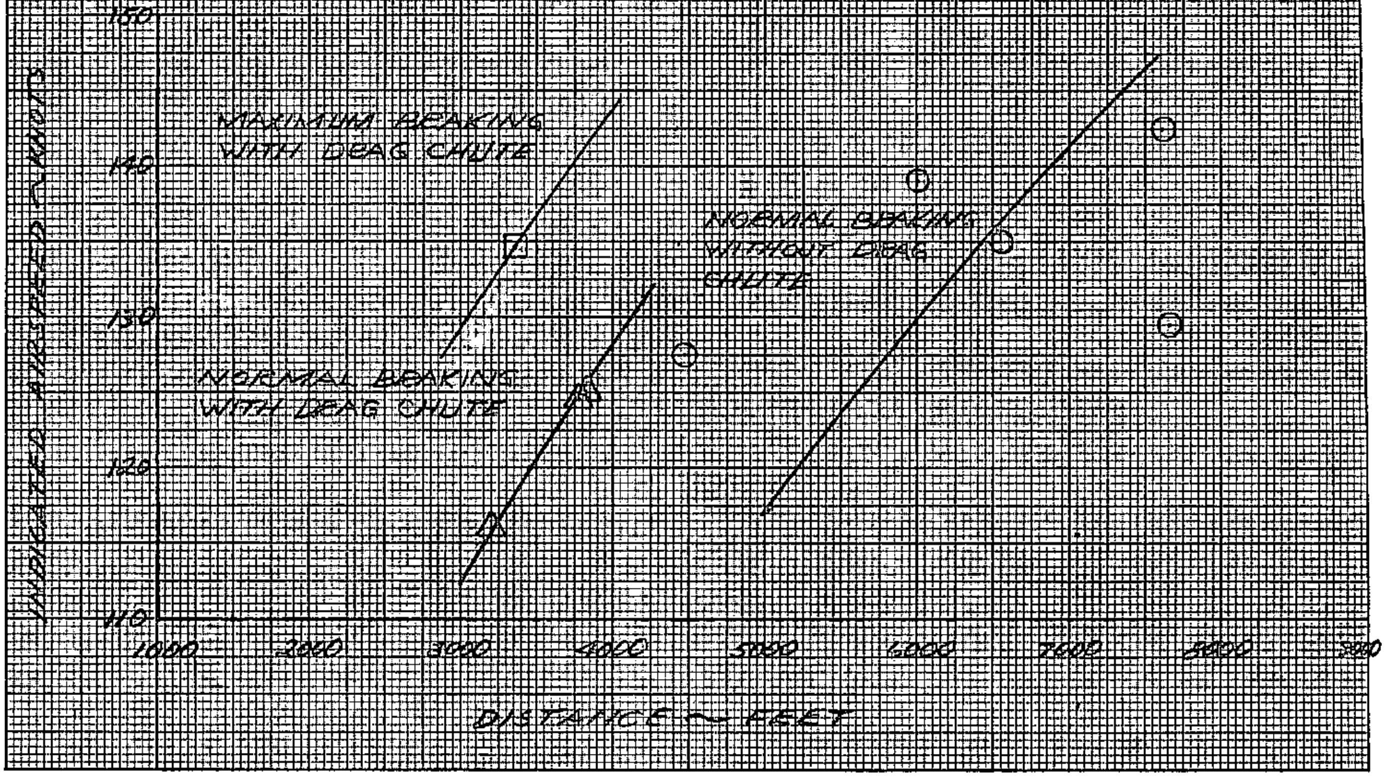


FIGURE 15
THRUST CALIBRATION
TF-102 A USAF S/N 54-1351
ENGINE ~ J-57P23 S/N 604498 AFTERBURNER S/N (SEE BELOW)
FLAP TYPE NOZZLE (TANDEM CYLINDERS FORCED FLAP)
DOWNSTREAM PRESSURE PROBES

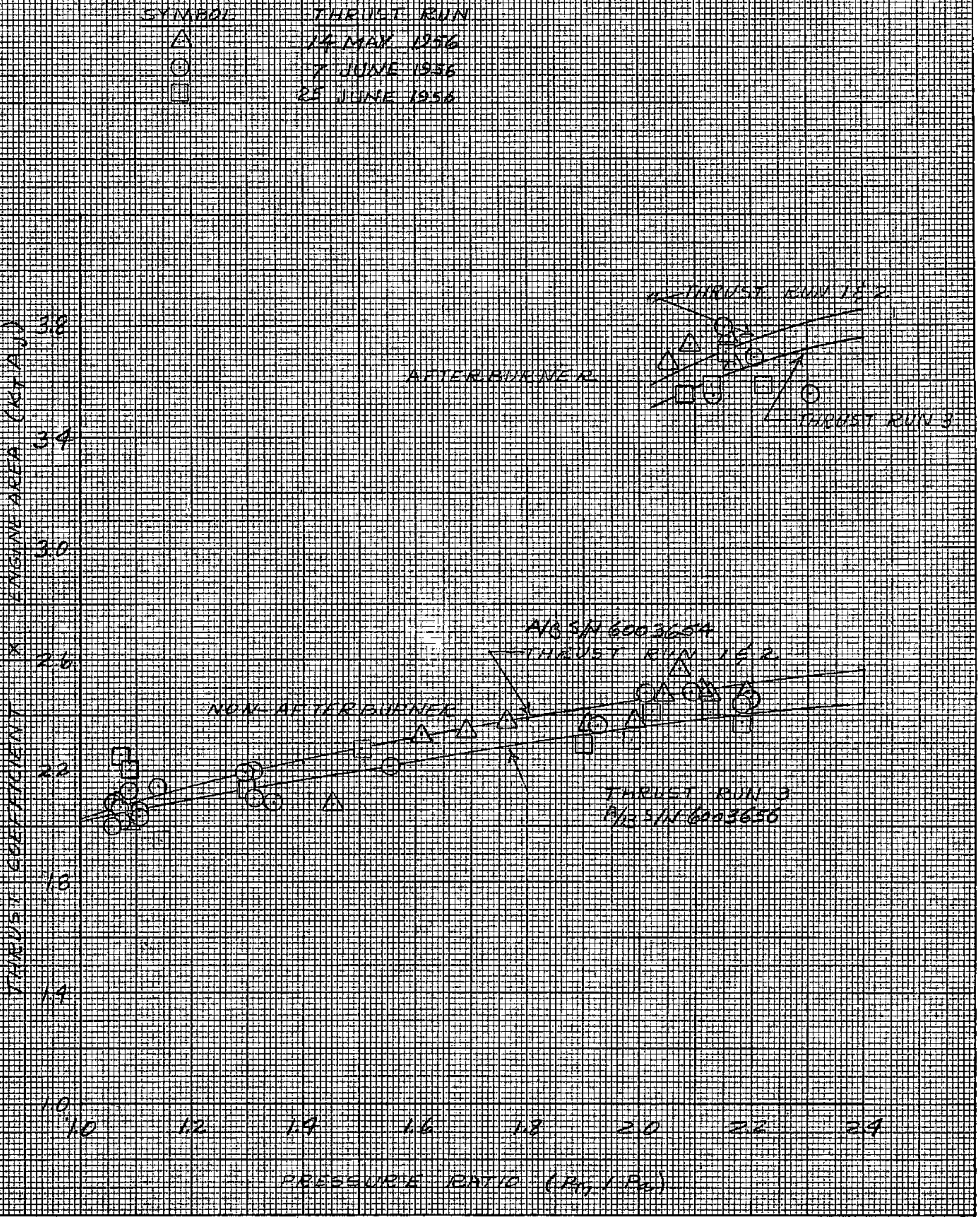


FIGURE 6
ENGINE THRUST CALIBRATION
TF-102A USAF S/N 54-1351

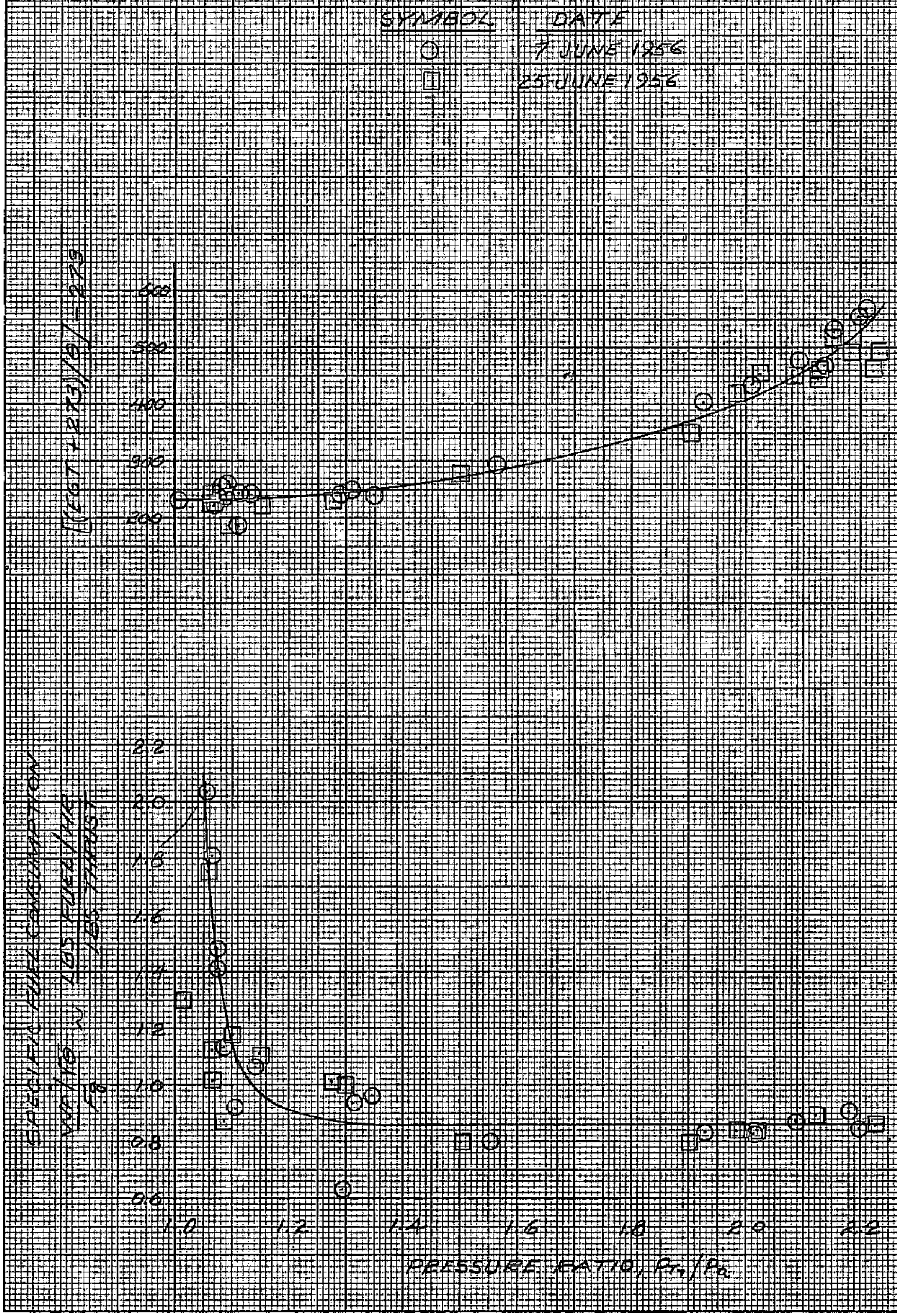


FIGURE 17
STATIC THRUST CALIBRATION
TF-102A USAF S/N 54-1351
ENGINE – J-57 P23 S/N 604498 AFTERBURNER S/N (SEE BELOW)
FLAP TYPE NOZZLE (TANDEM CYLINDERS FORCED FLAP)
DOWN STREAM PRESSURE PROBES

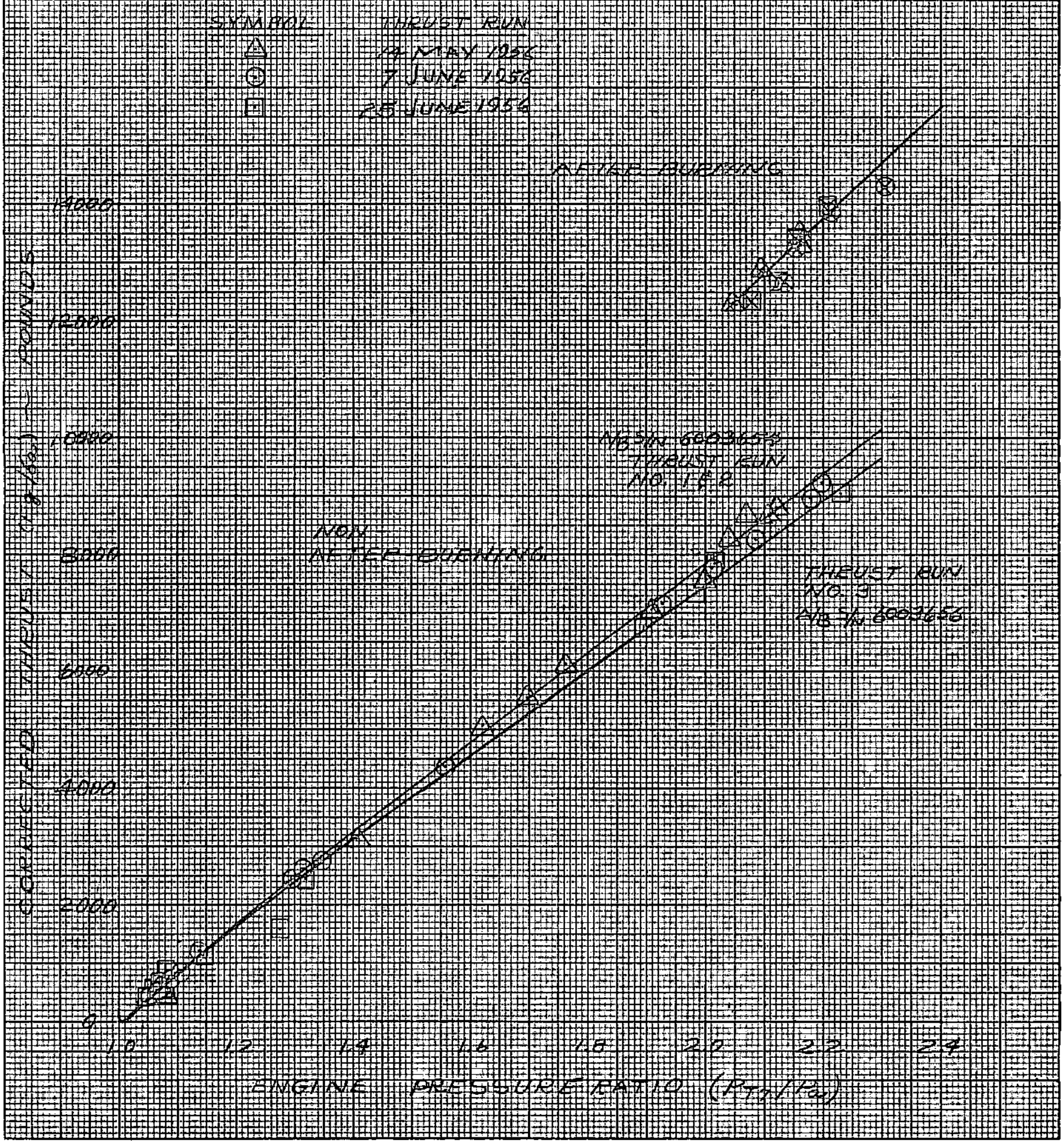


FIGURE 18
PRESSURE RATIO BIAS CURVE
TF-102A S/N 54-1351

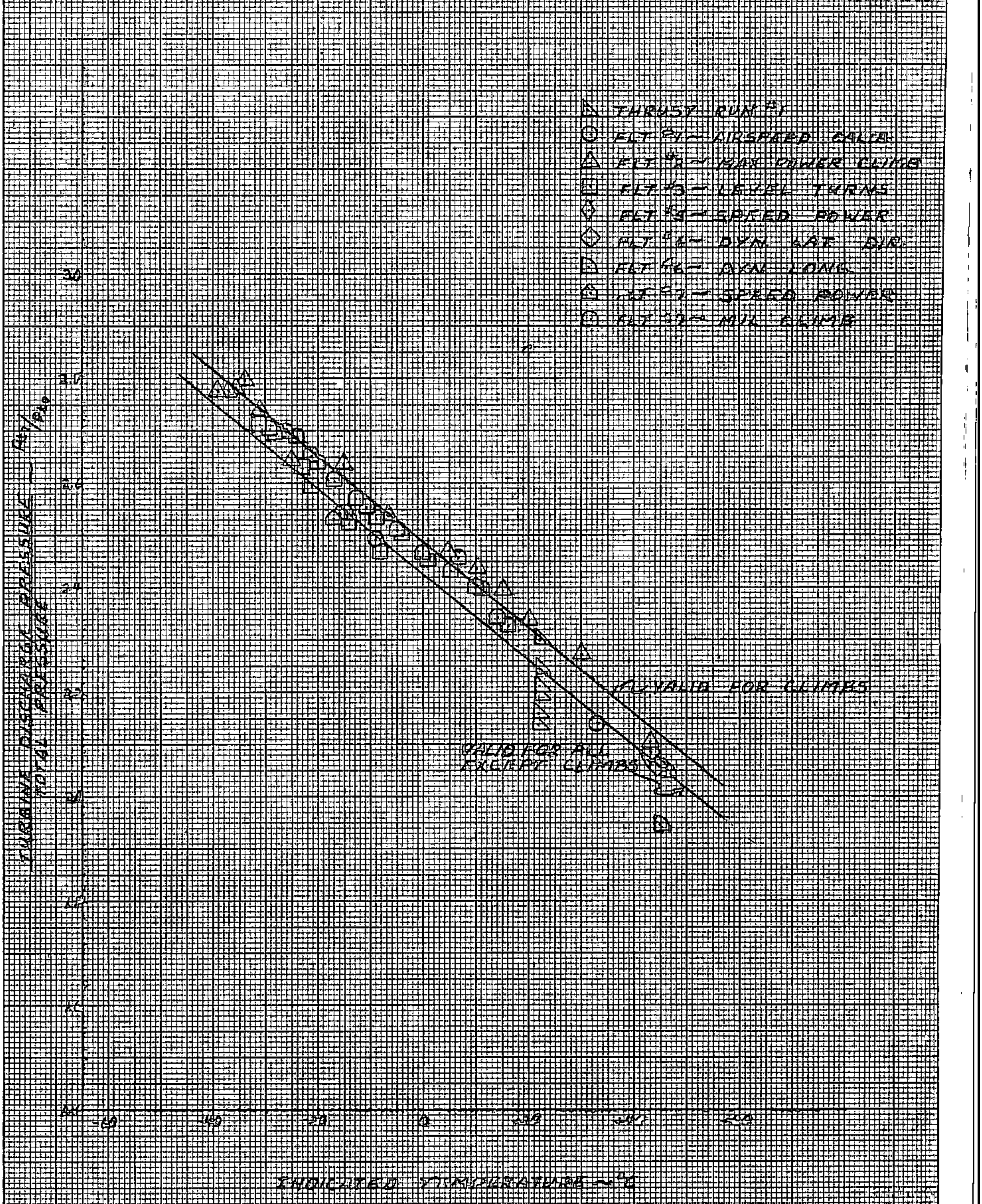
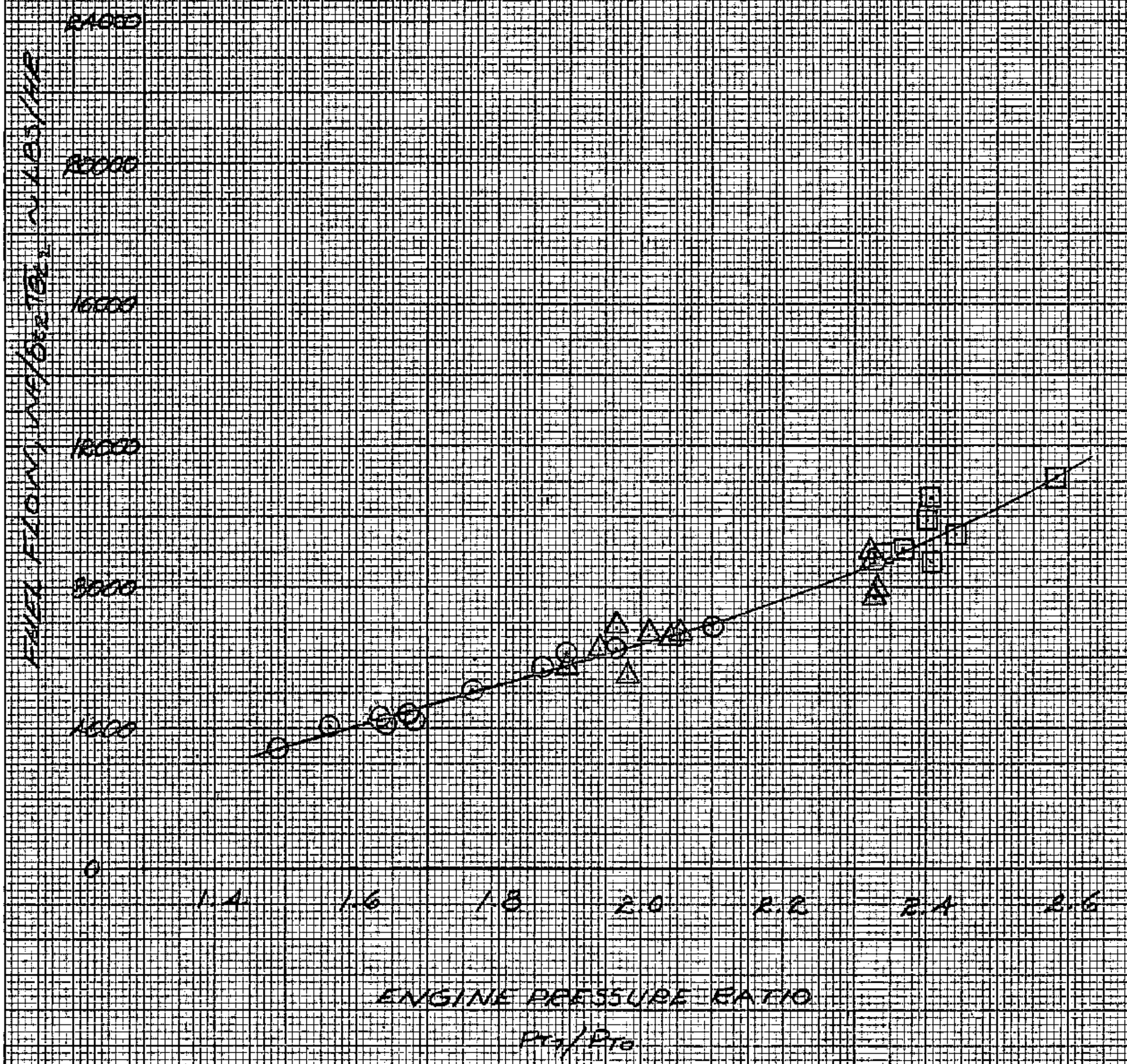


FIGURE 19

LEVEL FLIGHT PERFORMANCE

TF102A USAF S/N 54-1351

SYMBOL	W/δ	ALT.	WEIGHT
○	40140	10,000'	27,600
△	71,570	25,000'	26,550
□	110,870	35,000'	26,080



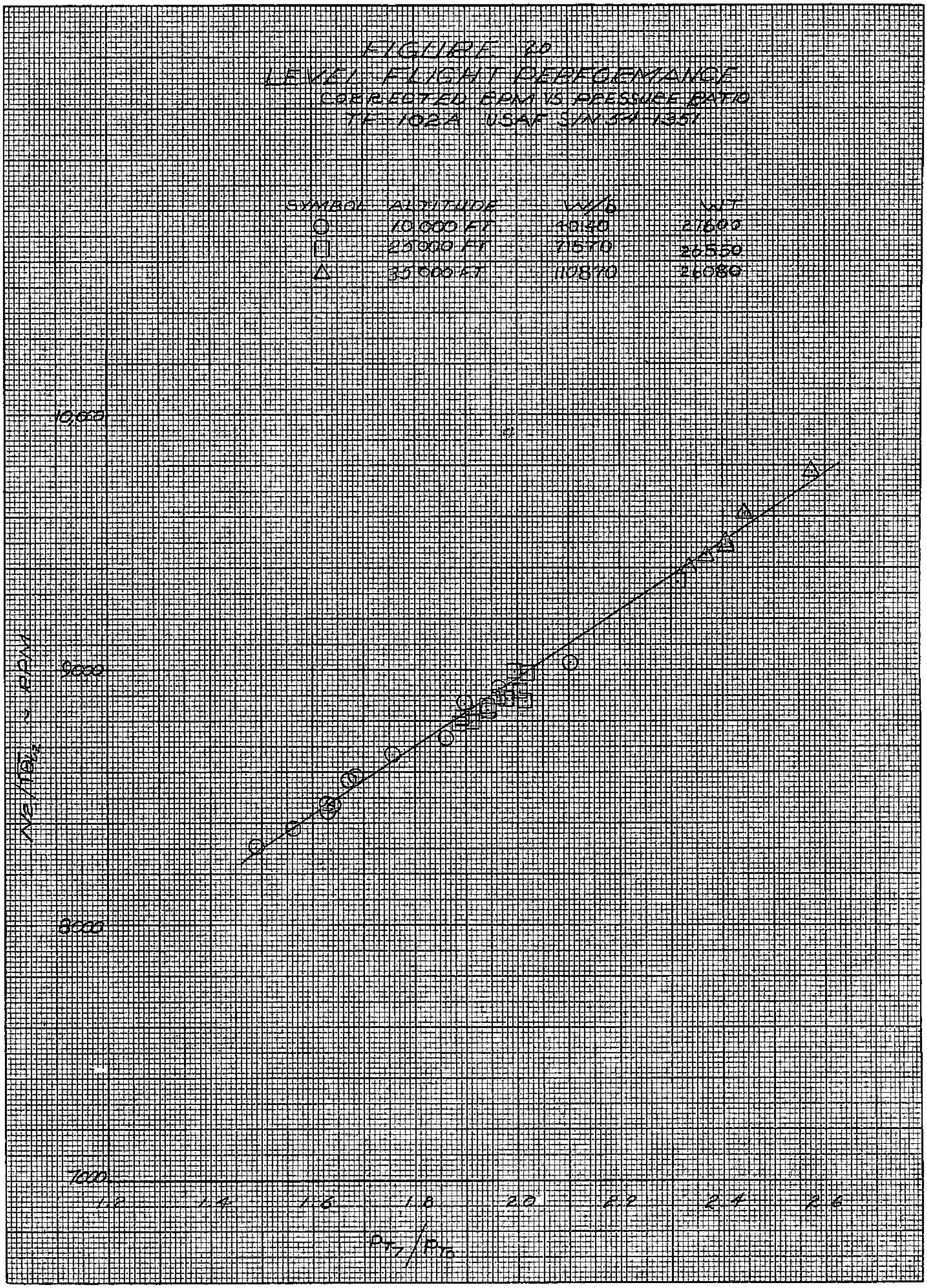


FIGURE 20
LEVEL FLIGHT PERFORMANCE
CORRECTED RPM VS PRESSURE RATIO
TF-102A USAF S/N 54-1351

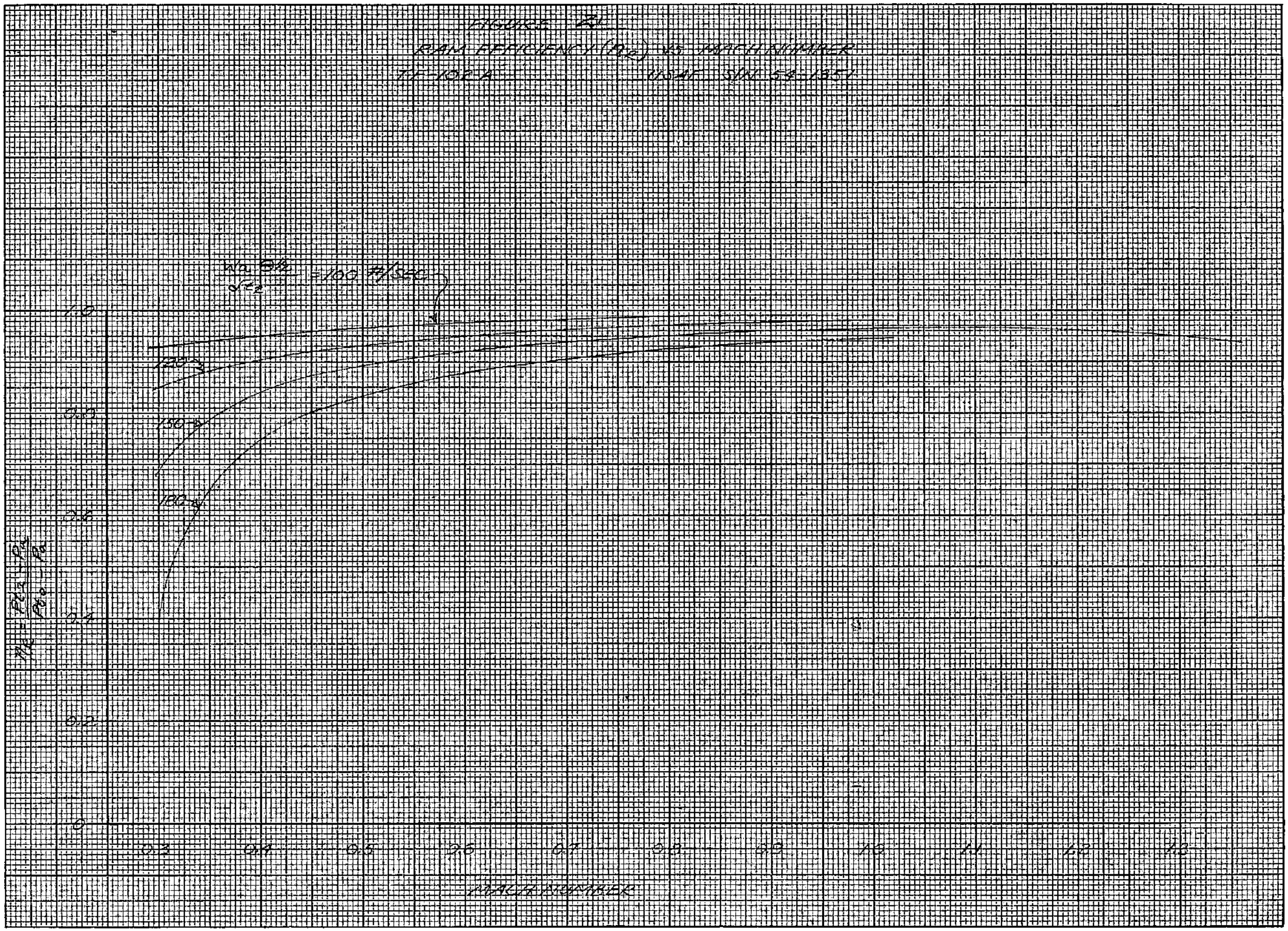
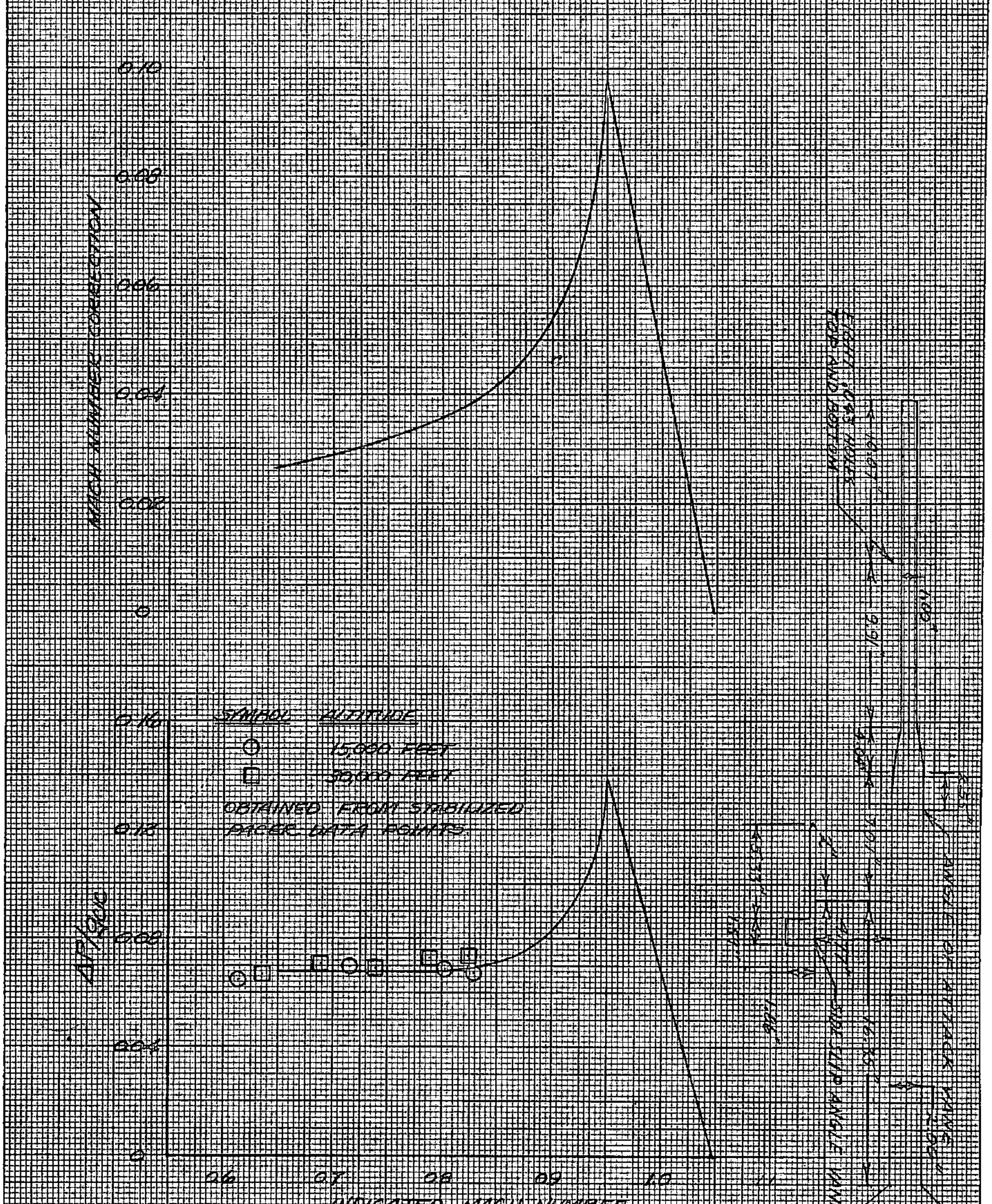


FIGURE 21

RAM EFFICIENCY (η_e) VS MACH NUMBER

TF-102A USAF S/N 53-1351

FIGURE 22
AIRSPEED CALIBRATION
TF102A — USAF S/N 54-1351



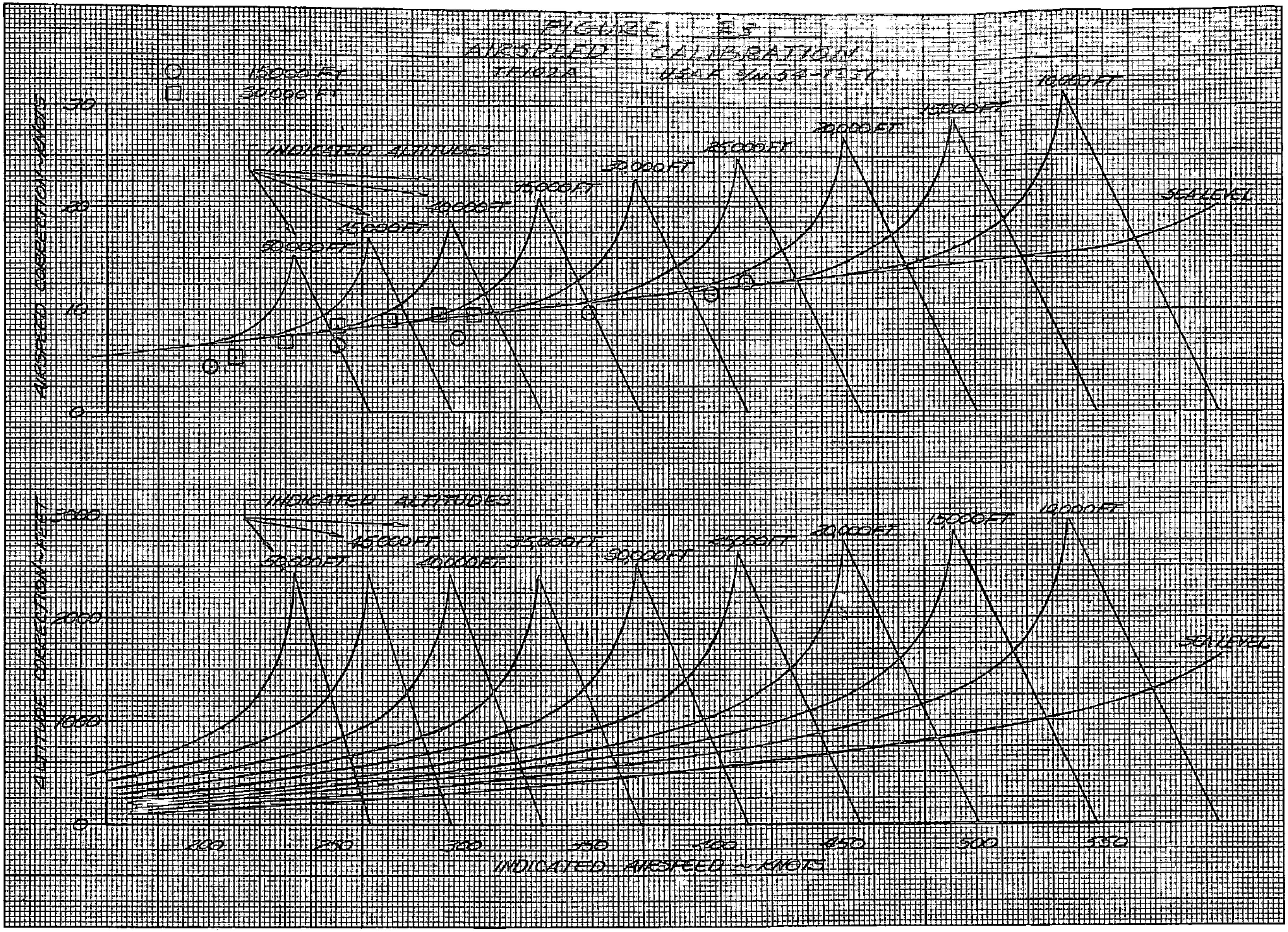


FIGURE E5
AIRSPEED CALIBRATION
TF-102A USAF S/N 54-1357

FIGURE 24

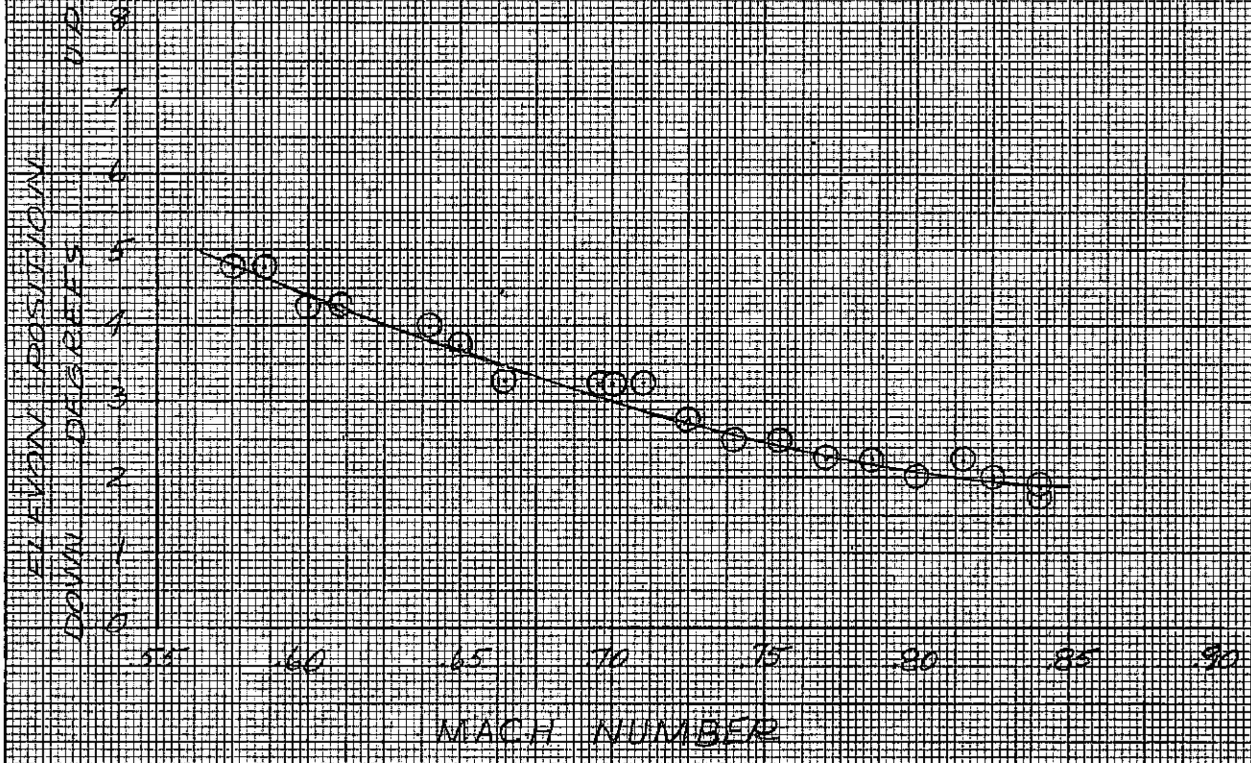
STATIC LONGITUDINAL STABILITY

TF-102A USAF NO 54-1351

CRUISE CONFIGURATION ALTITUDE 35000 FT

WEIGHT 26 835 LBS. C.G. 23.7% MAC.

POWER SLOWLY INCREASED (½% PER SEC) DURING ACCELERATION



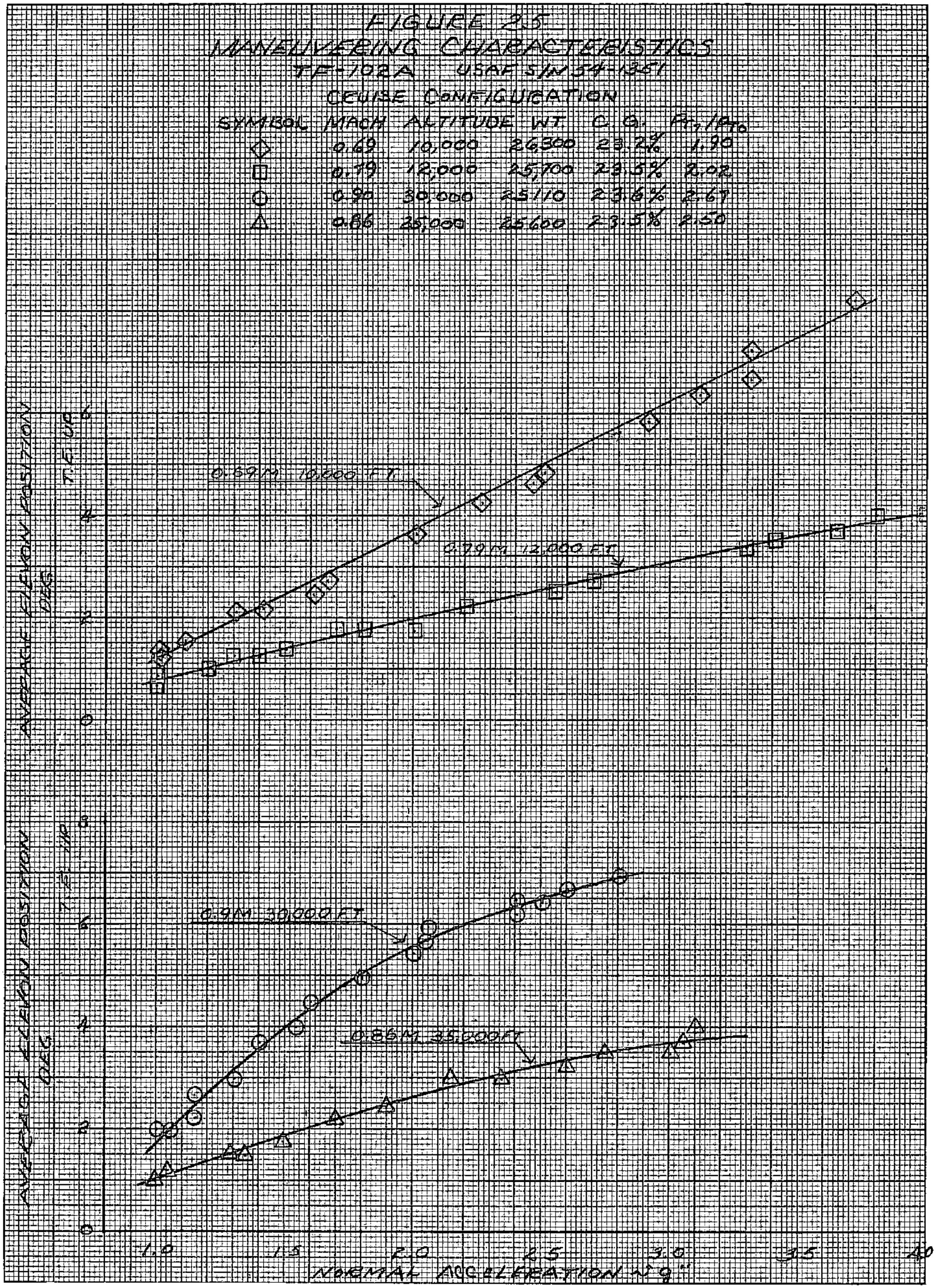


FIGURE 2.5
MANEUVERING CHARACTERISTICS
TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION

FIGURE 26
DYNAMIC LONGITUDINAL STABILITY
TF-102A USAF S/N 54-1351
ALT. 10,000 FT
VIC 325 KTS
WT. 3,000 LBS
C.G. 23.6 % MAC
CRUISE CONFIGURATION
PITCH DAMPER ON

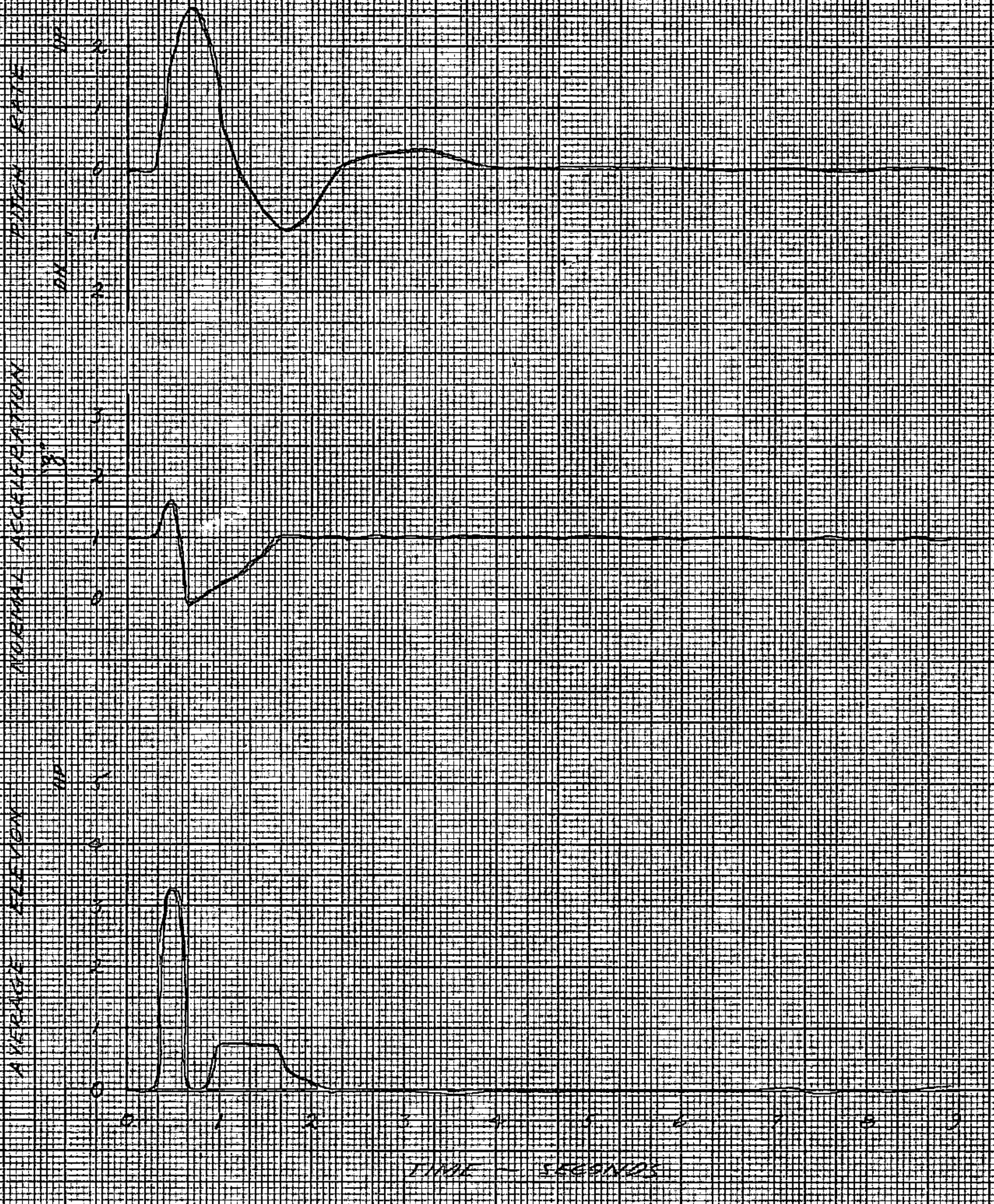


FIGURE 27
DYNAMIC LONGITUDINAL STABILITY
TF-102A USAF S/N 54-1351
ALT 15,000 FT
Vic 180 KTS
WT 26800 LBS
C.G. 23.8 % MAC
POWER APPROACH CONFIGURATION
PITCH DAMPER ON

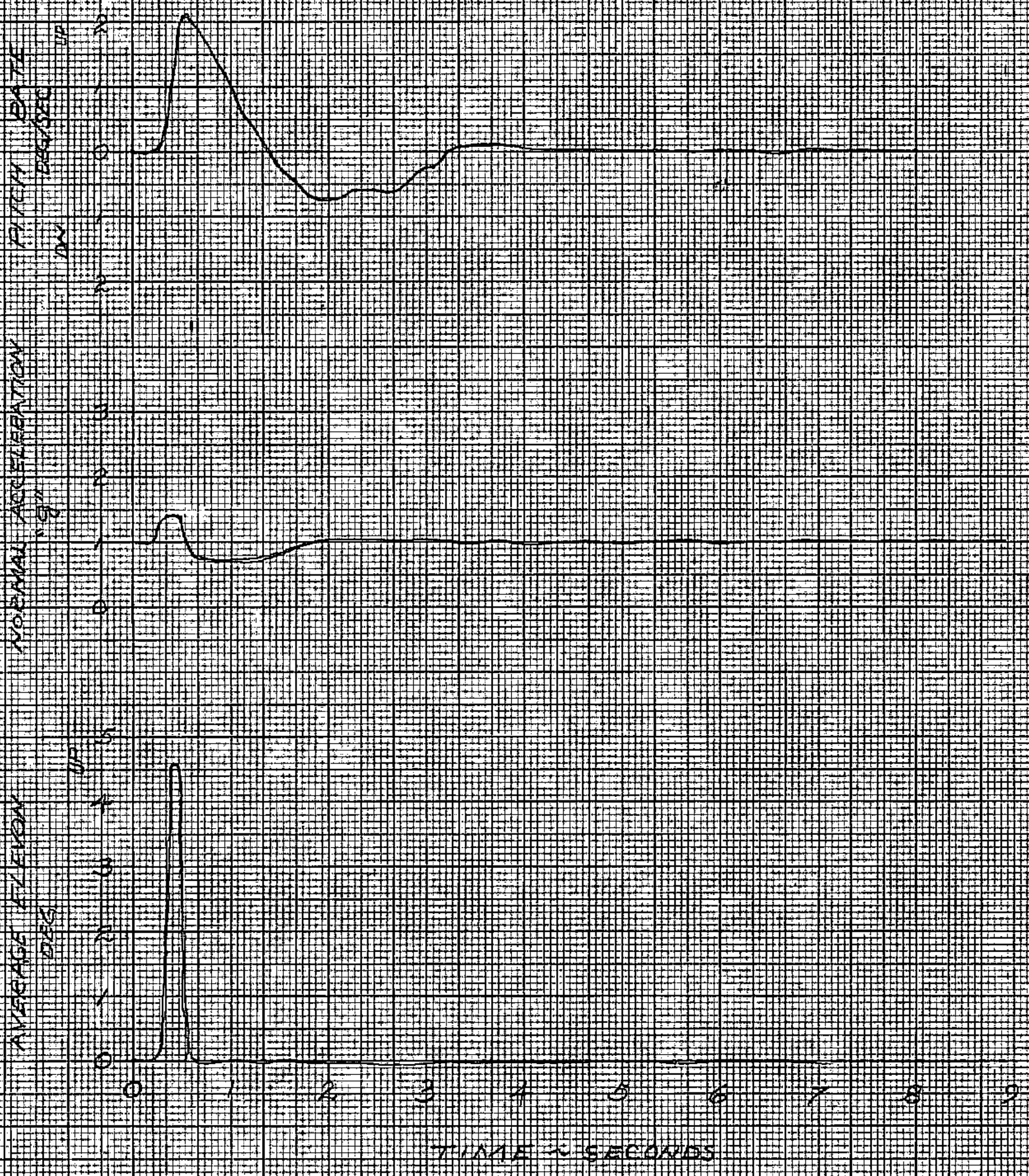


FIGURE 28

DYNAMIC LONGITUDINAL STABILITY

TF-102A USAF S/N 54-1351

ALT 15000 FT.
V_c 180 KTS
WT. 26300 LBS
C.G. 23.8 % MAC
POWER APPROACH CONFIGURATION
PITCH DAMPER OFF

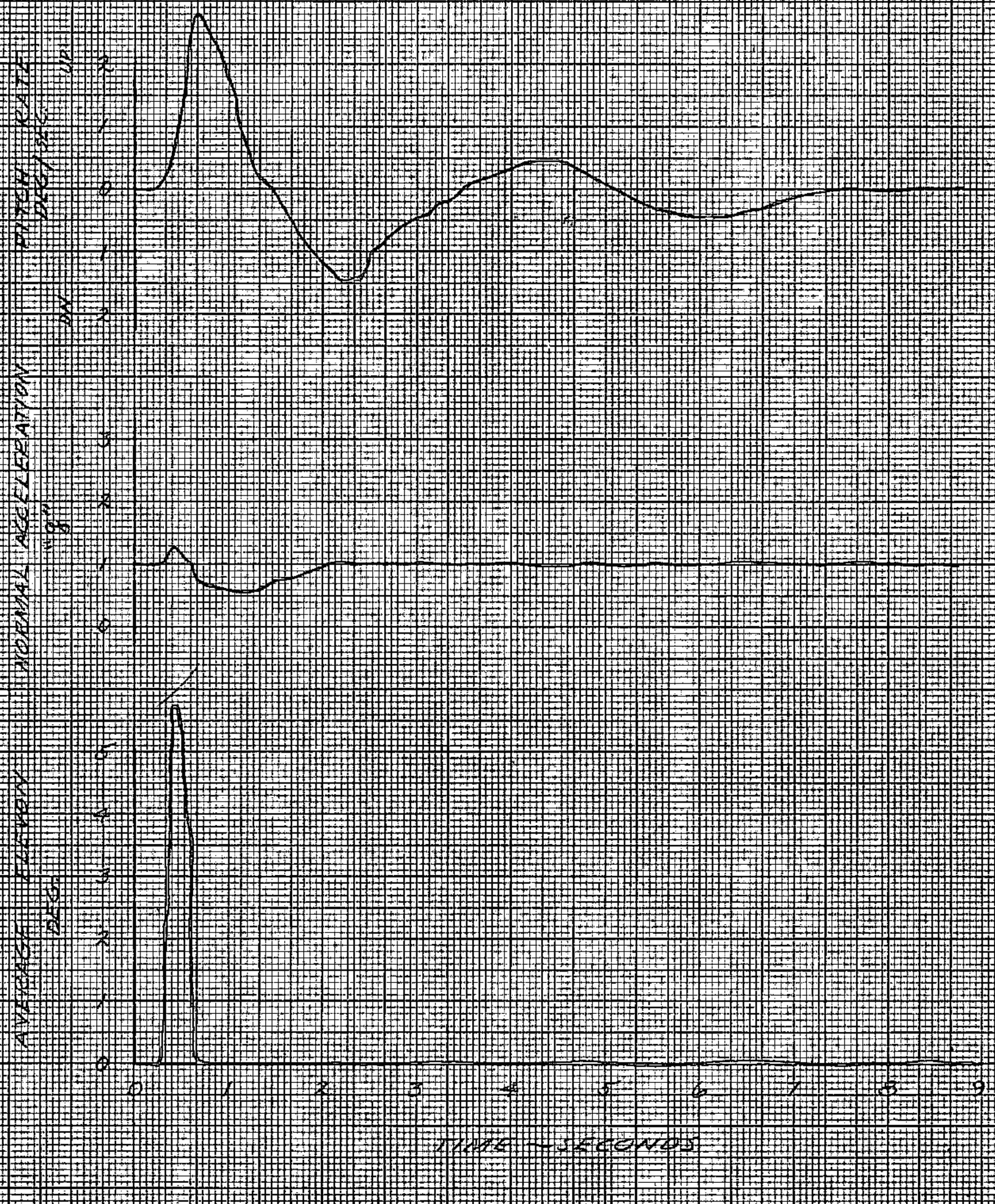


FIGURE 29
DYNAMIC LATERAL-DIRECTIONAL STABILITY
TF-102 USAF S/N 54-1351
CRUISE CONFIGURATION
ALT. 10,000 FT
MACH NO. 0.8
WT. 25000 LBS
C.G. 23.5 %MAC

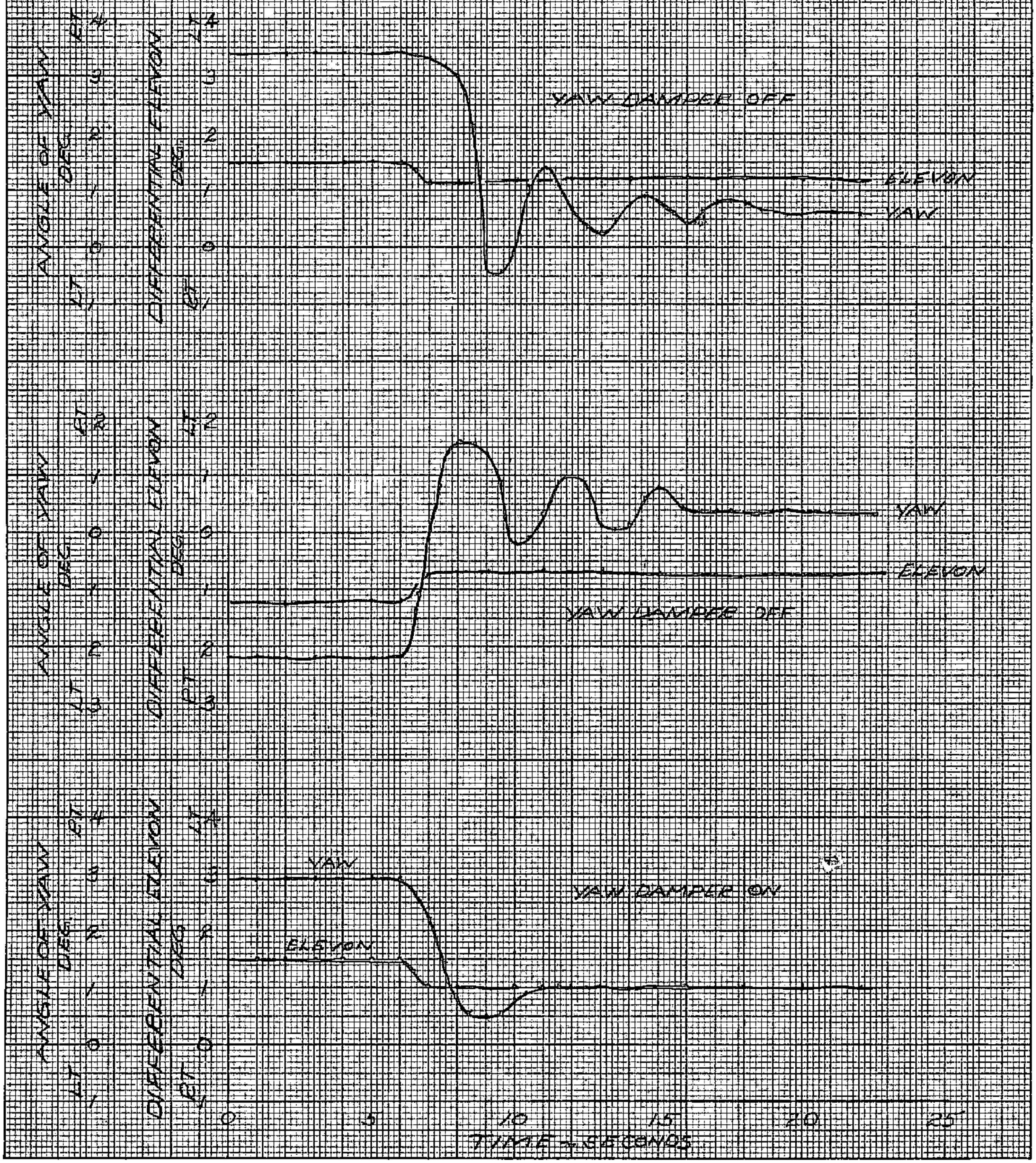


FIGURE 30
DYNAMIC LATERAL-DIRECTIONAL STABILITY

TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
ALT. 11,000 FT
MACH NO. 0.6
WT 25169.1 LBS
C.G. 25.8 %MAC

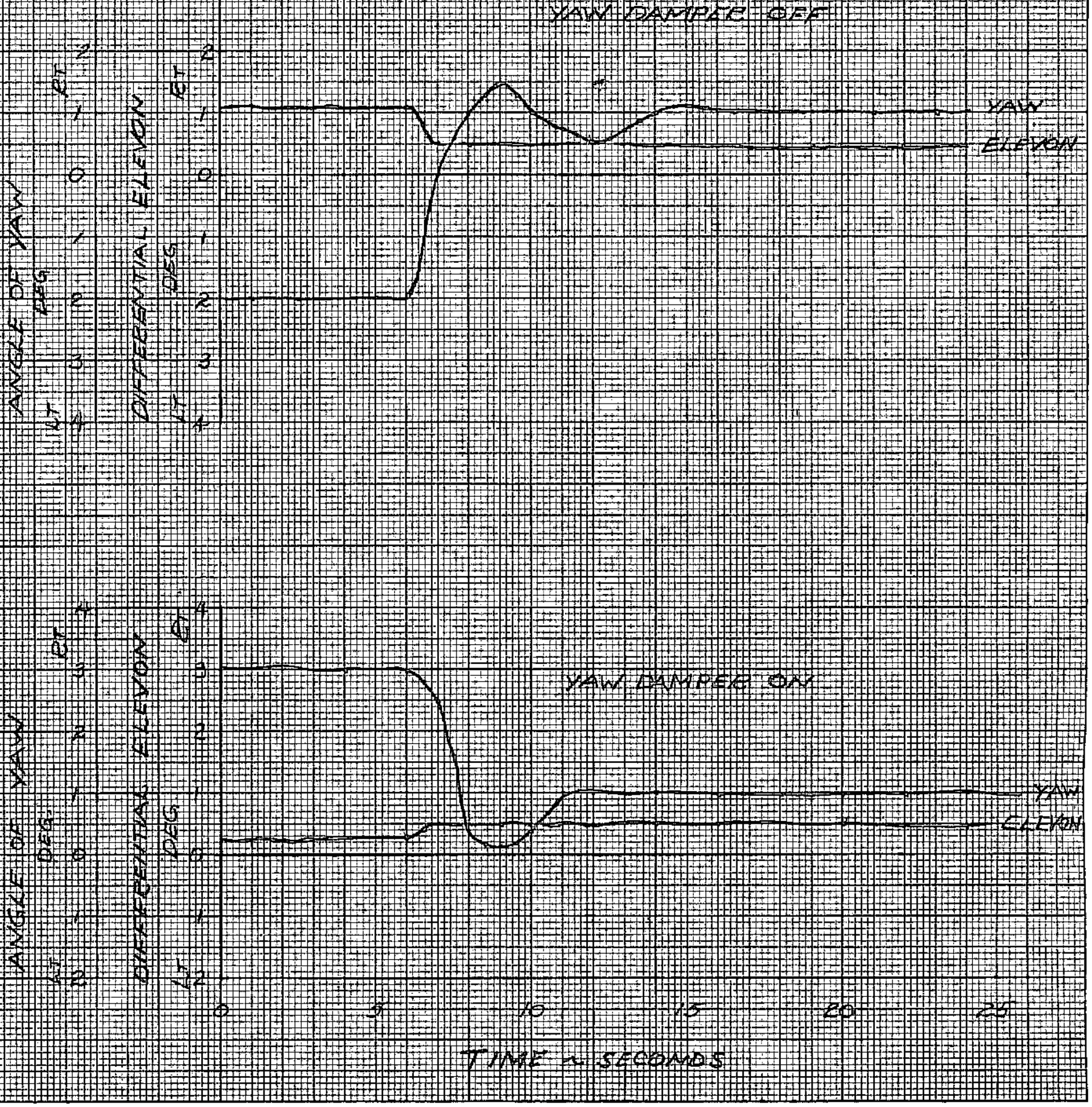


FIGURE 31
DYNAMIC LATERAL DIRECTIONAL
TF102A USAF S/N 54-1351
ALT 15000 FT
MACH NO. 0.6
WT 24000 LB
C.G. 23.5 %MAC
CRUISE CONFIGURATION
YAW DAMPER ON

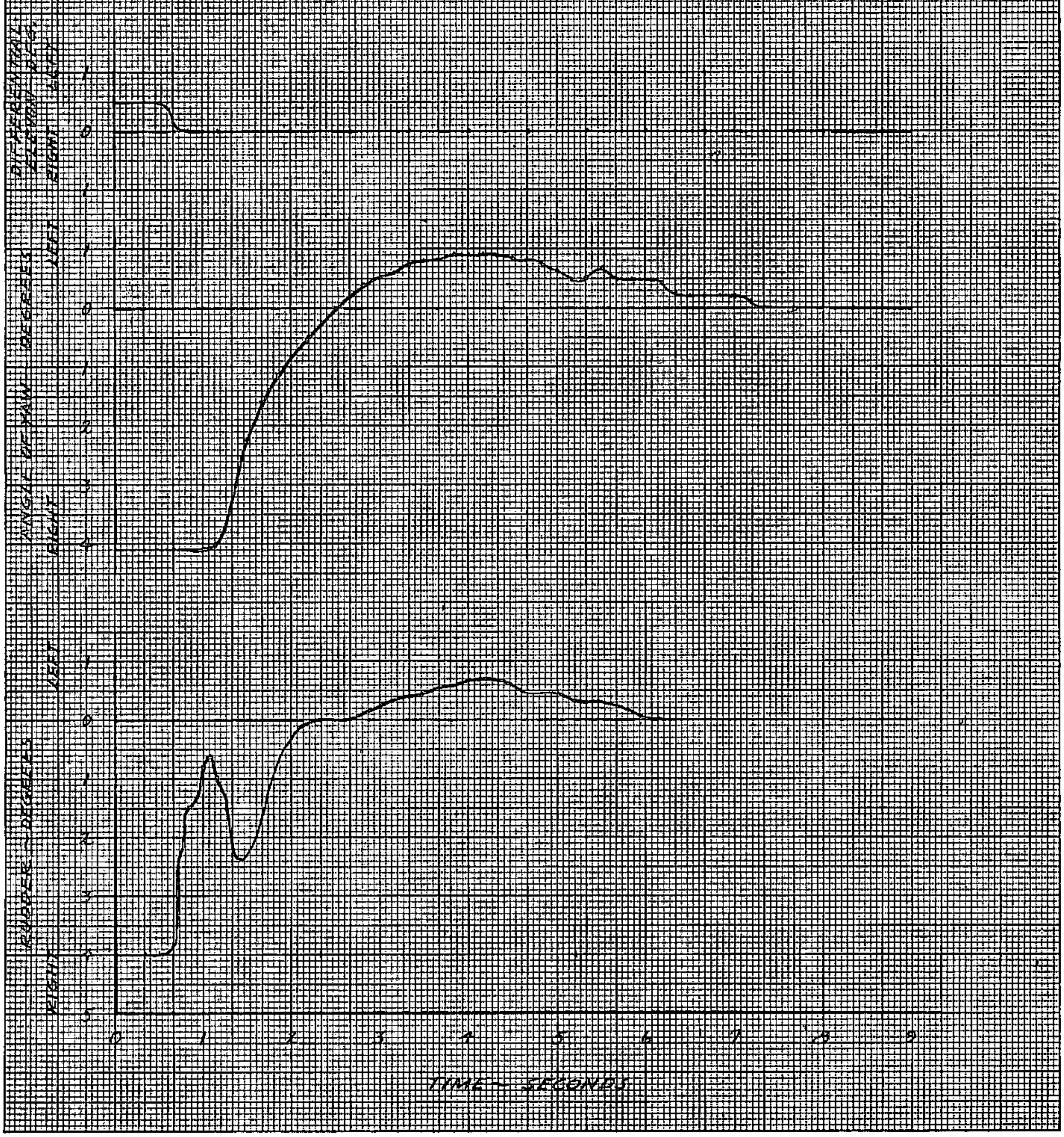


FIGURE 32
DYNAMIC LATERAL-DIRECTIONAL STABILITY
TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
ALT. 35000 FT
MACH NO. 0.7
WT. 25,700 LBS
C.G. 23.5 % MAC

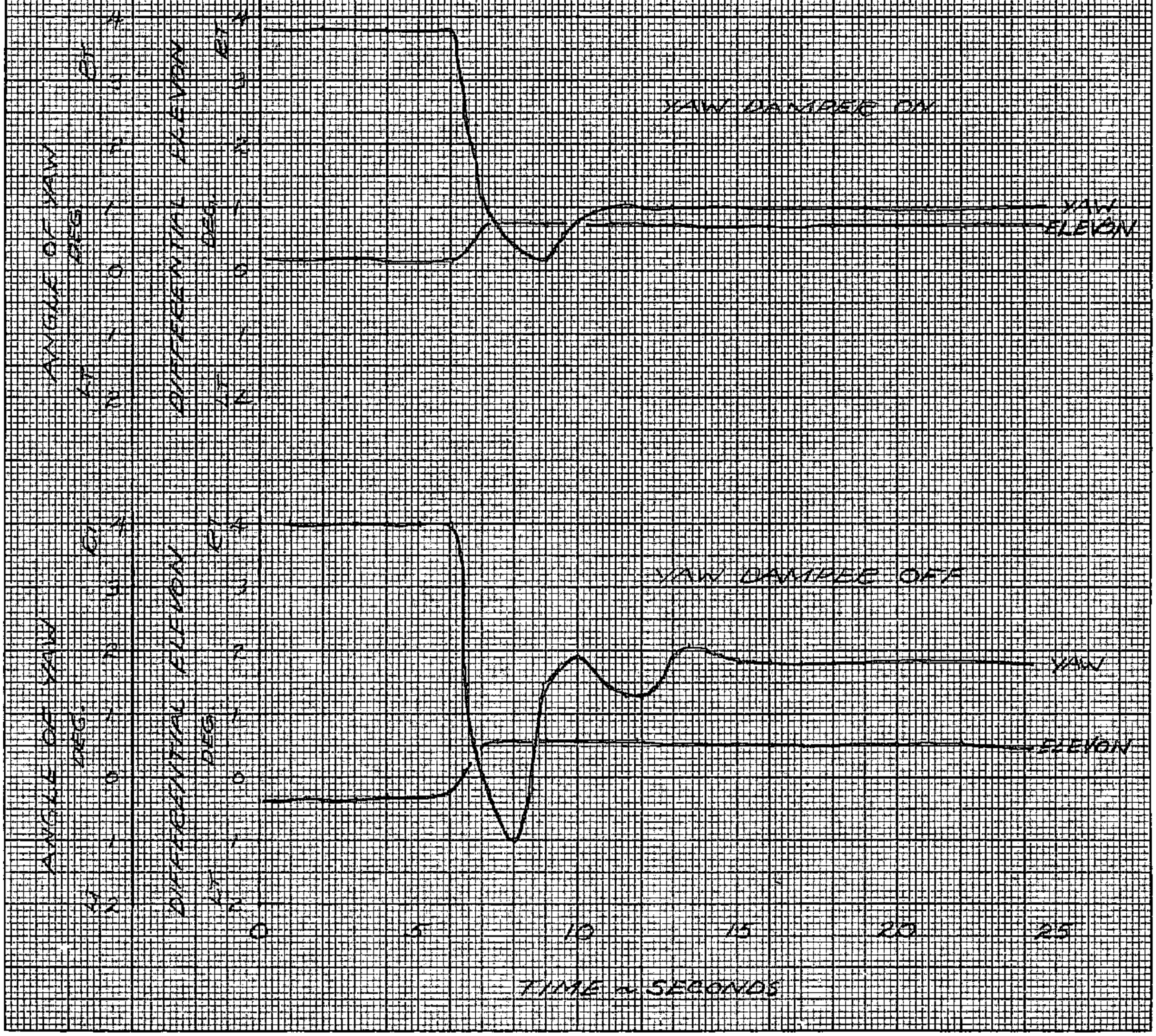


FIGURE 35

DYNAMIC LATERAL-DIRECTIONAL STABILITY

TF-102A USAF S/N 54-1351

CRUISE CONFIGURATION

ALT. 35000 FT

MACH NUMBER 0.8

WT. 26200 LBS

C.G. 23.3 % MAC

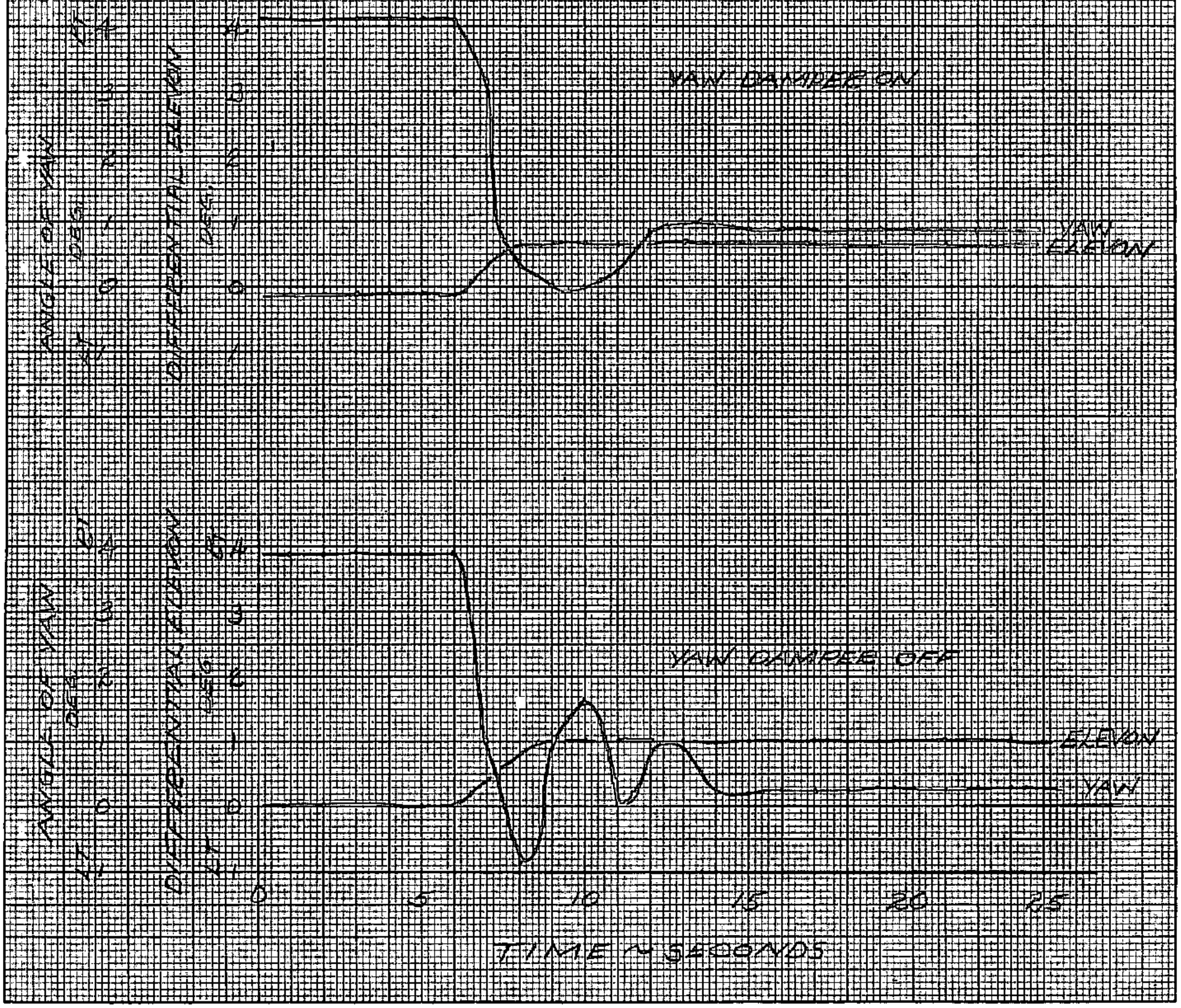


FIGURE 34
DYNAMIC LATERAL-DIRECTIONAL STABILITY
TF-102A USAF S/N 54-1351
POWER APPROACH CONFIGURATION
ALT. 10,000 FT
VIC 180 KTS
WT. 25000 LBS
C.G. 23.5 % MAC

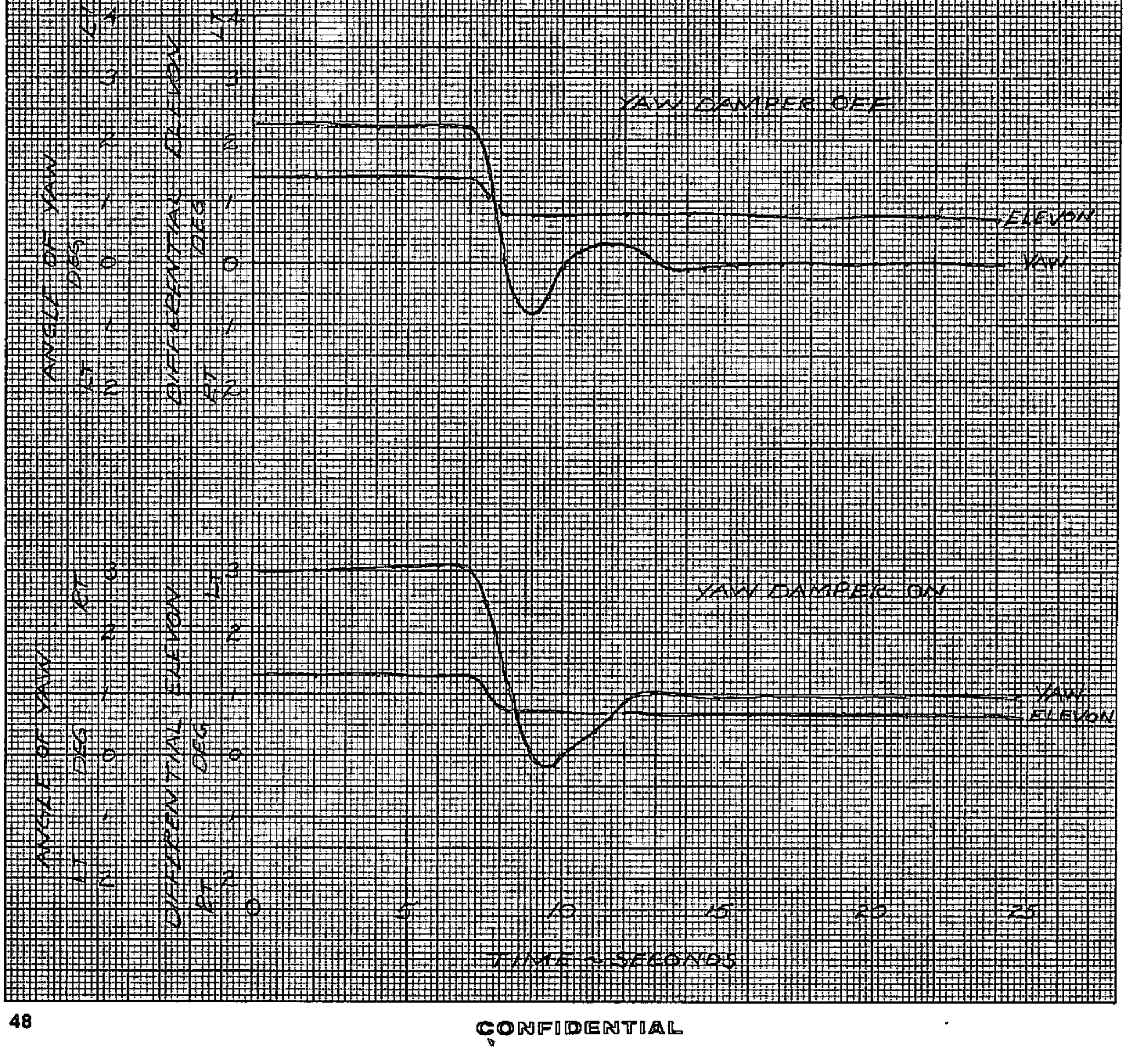


FIGURE 33
DYNAMIC LATERAL DIRECTIONAL
TF-102A USAF S/N 54-1351
ALT. 14700 FT
VIC. 182 KTS
WT. 25000 LBS
C.G. 23.5 %MAC
POWER APPROACH CONFIGURATION
YAW DAMPER ON

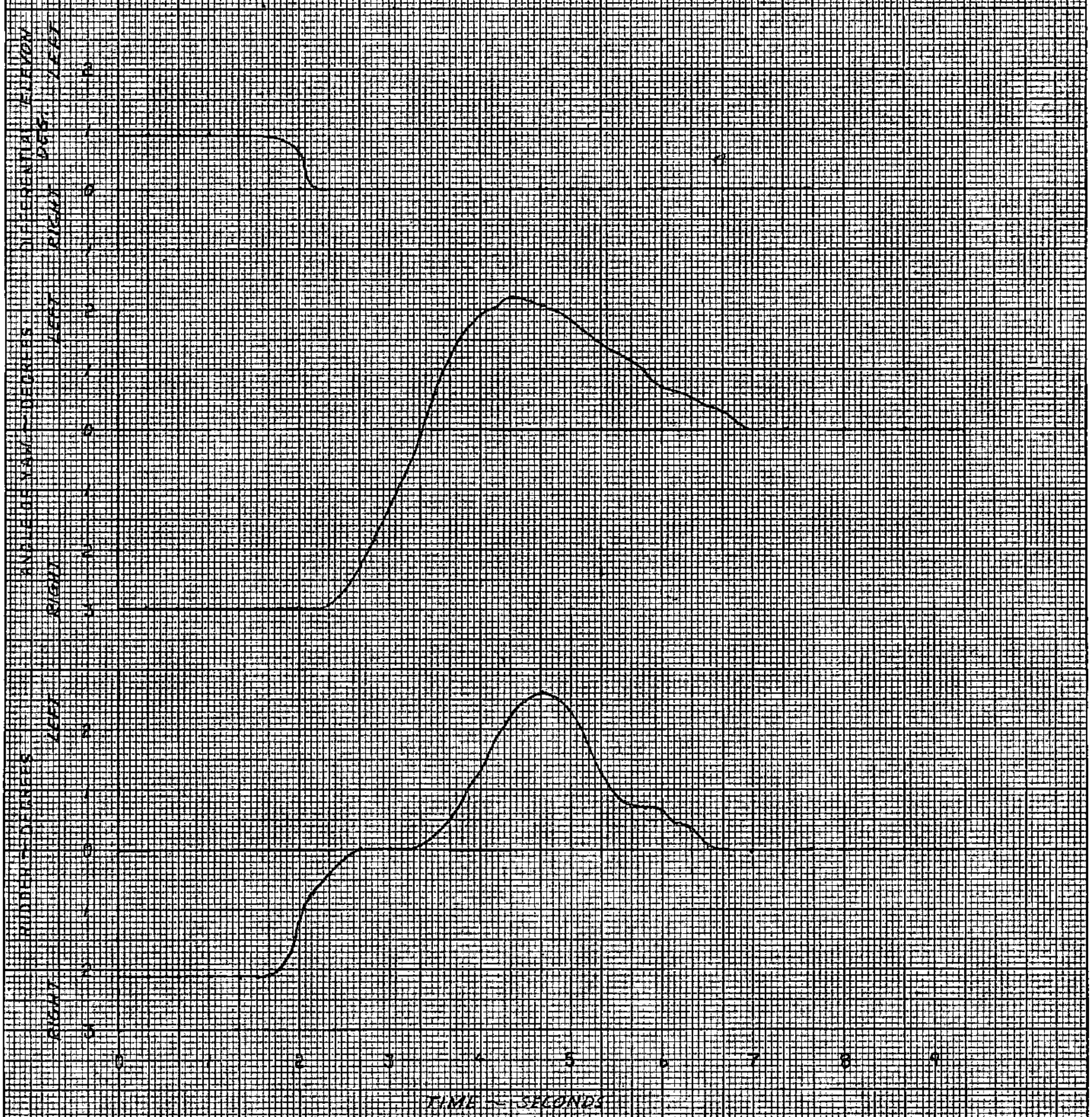


FIGURE 36

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
TRIM CONDITIONS:
WT. 24,850 LBS. MACH NO. .82
ALT. 34,750 FT. C.G. 23.8 % MAC
YAW DAMPER & TURN COORDINATOR OPERATING

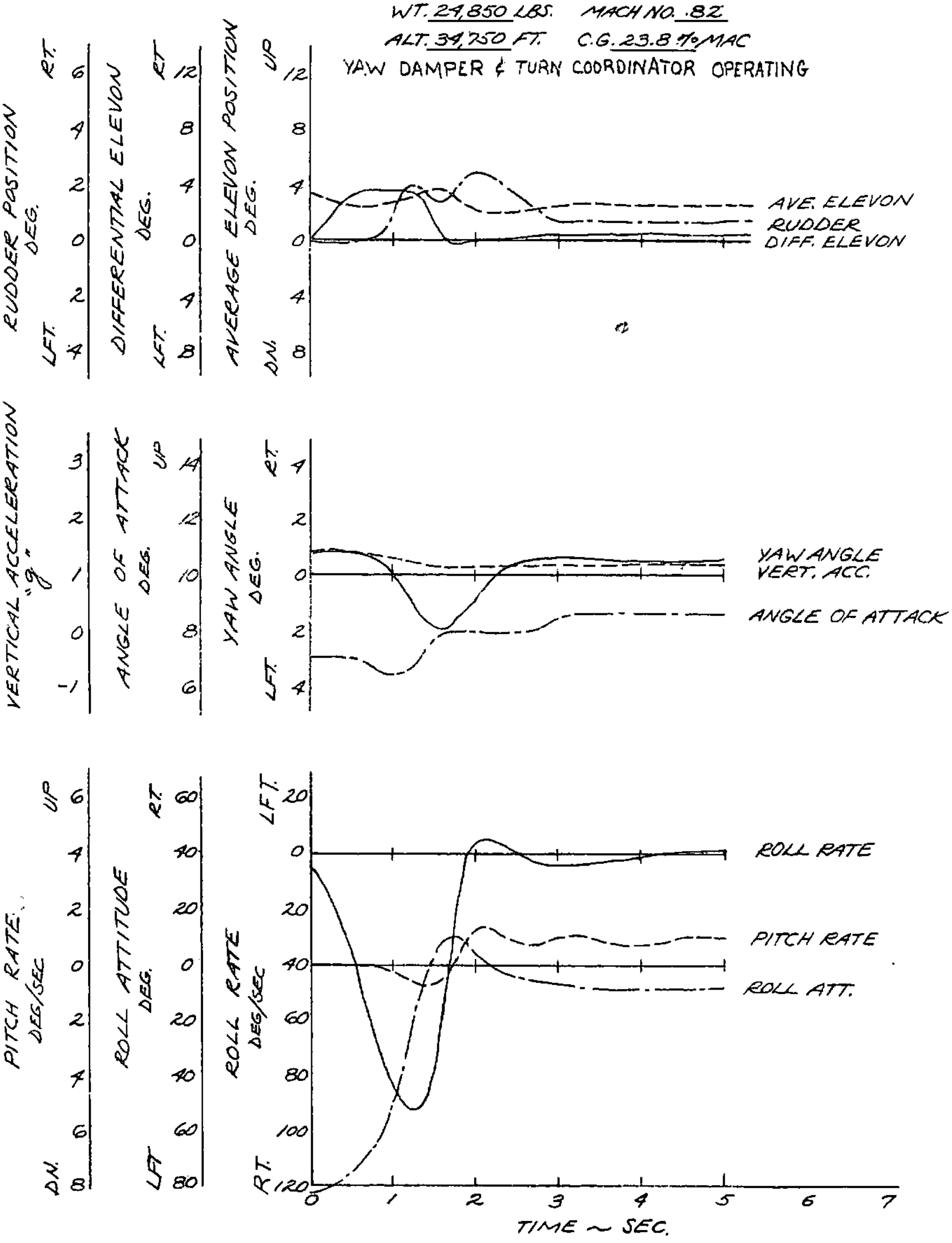


FIGURE 37
ROLL TIME HISTORIES
TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
TRIM CONDITIONS:
WT. 24,850 LBS. MACH NO. .82
ALT. 35,100 FT. C.G. 23.8% MAC
YAW DAMPER & TURN COORDINATOR OPERATING

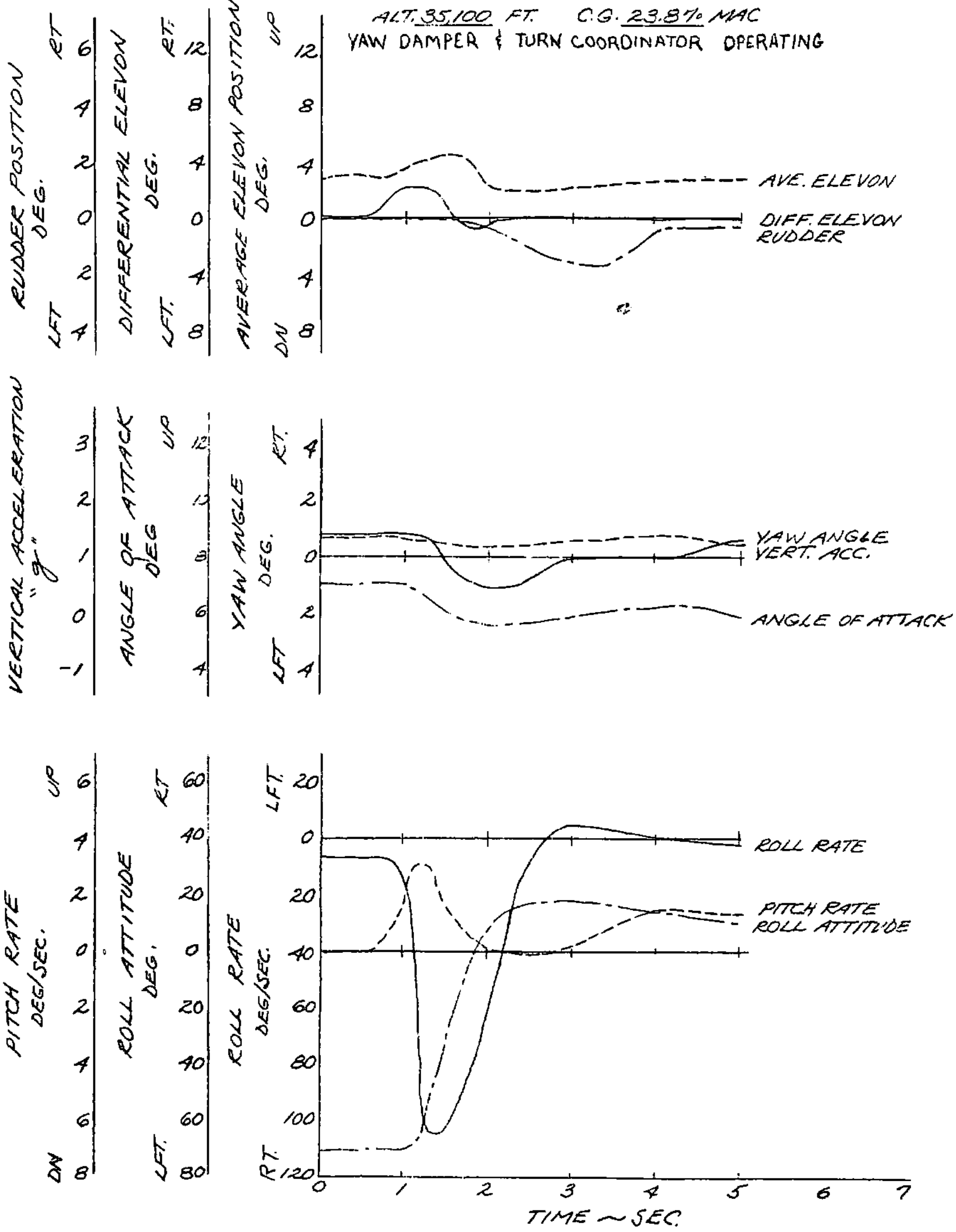


FIGURE 38
ROLL TIME HISTORIES
TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
TRIM CONDITIONS:
WT. 24,850 LBS. MACH NO. .83
ALT. 35,400 FT. C.G. 23.8 % MAC

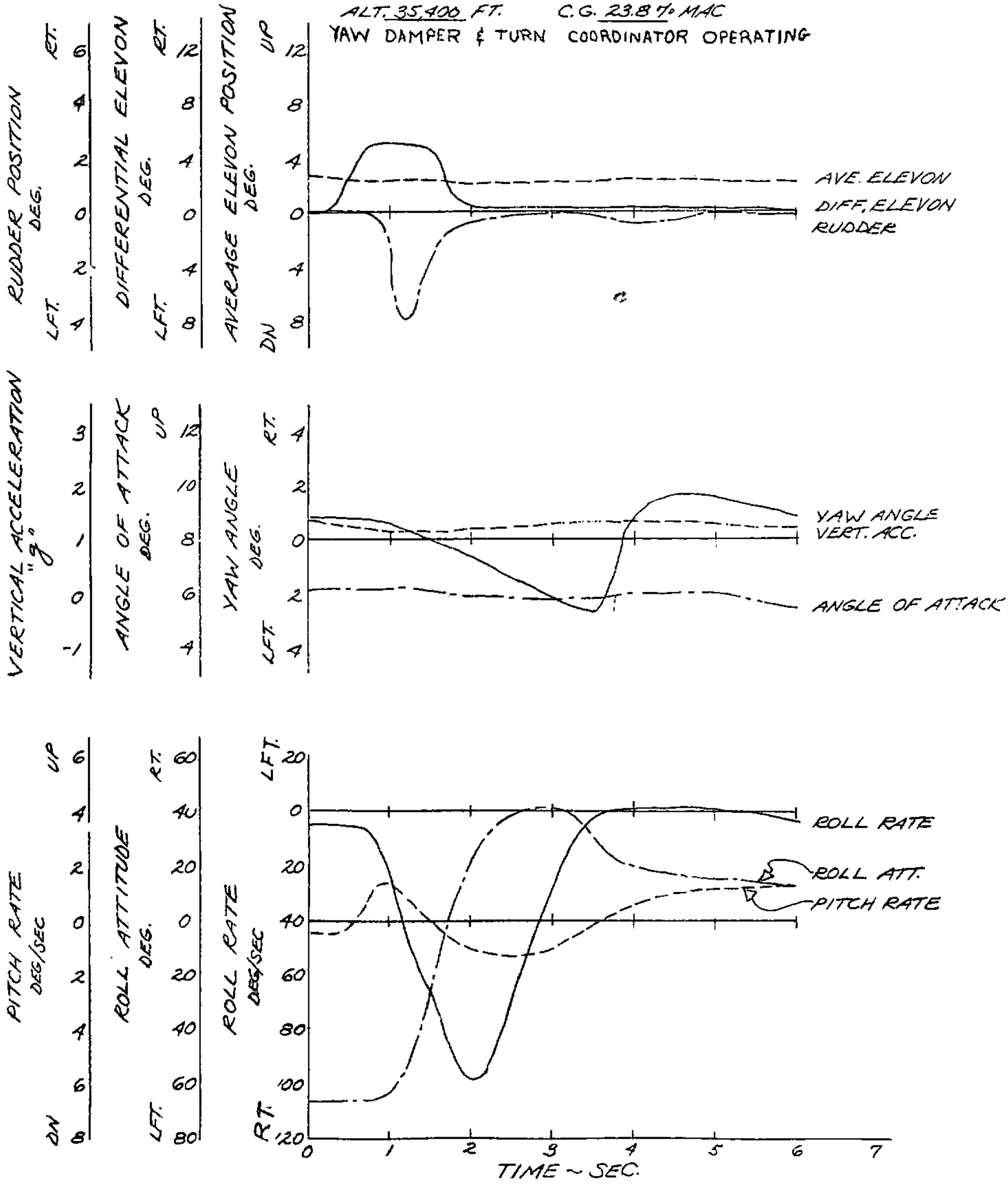


FIGURE 39

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
TRIM CONDITIONS:
WT. 24400 LBS. MACH NO. .77
ALT. 35,250 FT. C.G. 23.8% MAC
YAW DAMPER & TURN COORDINATOR OPERATING

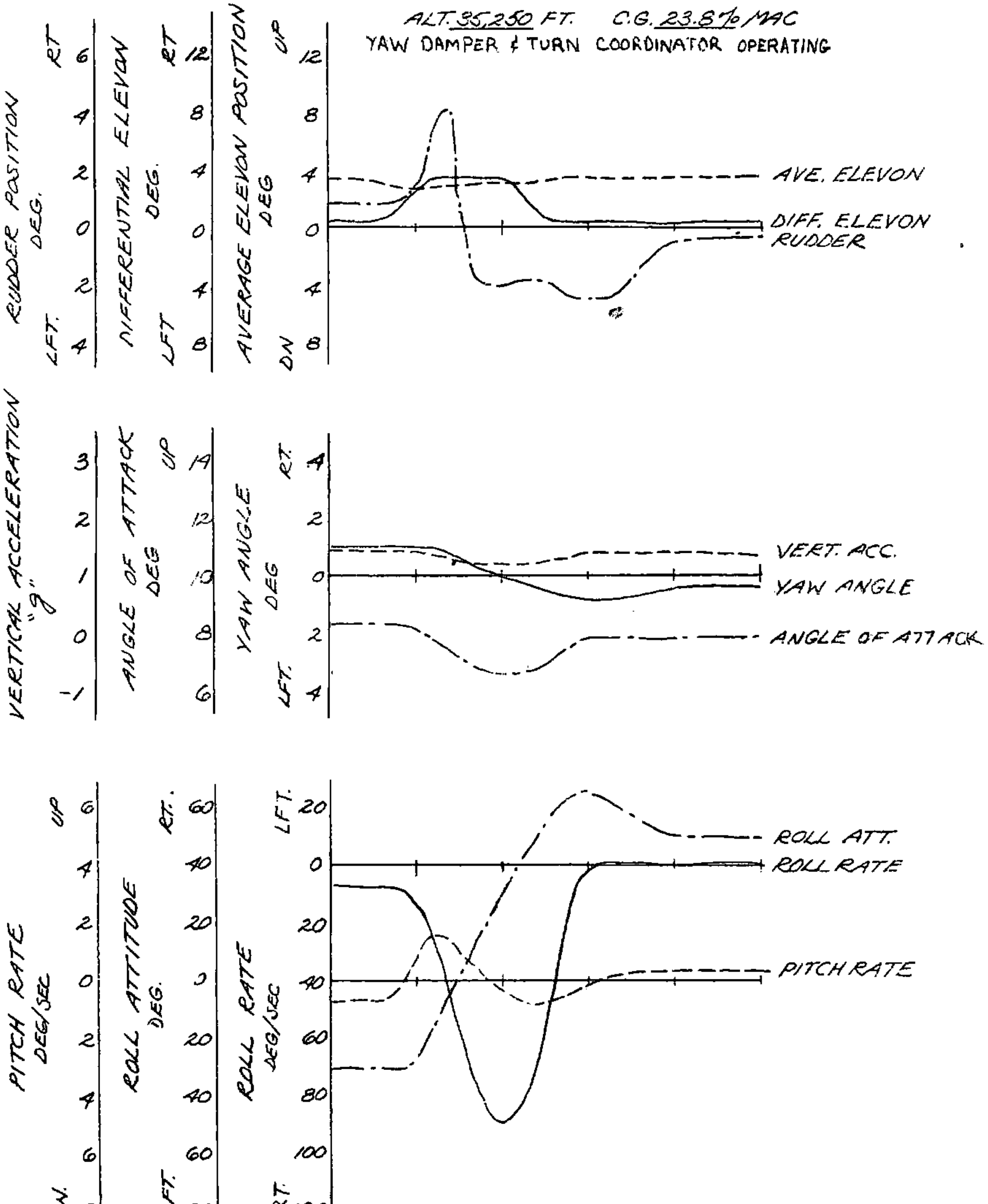


FIGURE 40

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
TRIM CONDITIONS:
WT. 24,400 LBS. MACH NO. .75
ALT. 35,050 FT. C.G. 23.8% MAC

YAW DAMPER & TURN COORDINATOR OPERATING

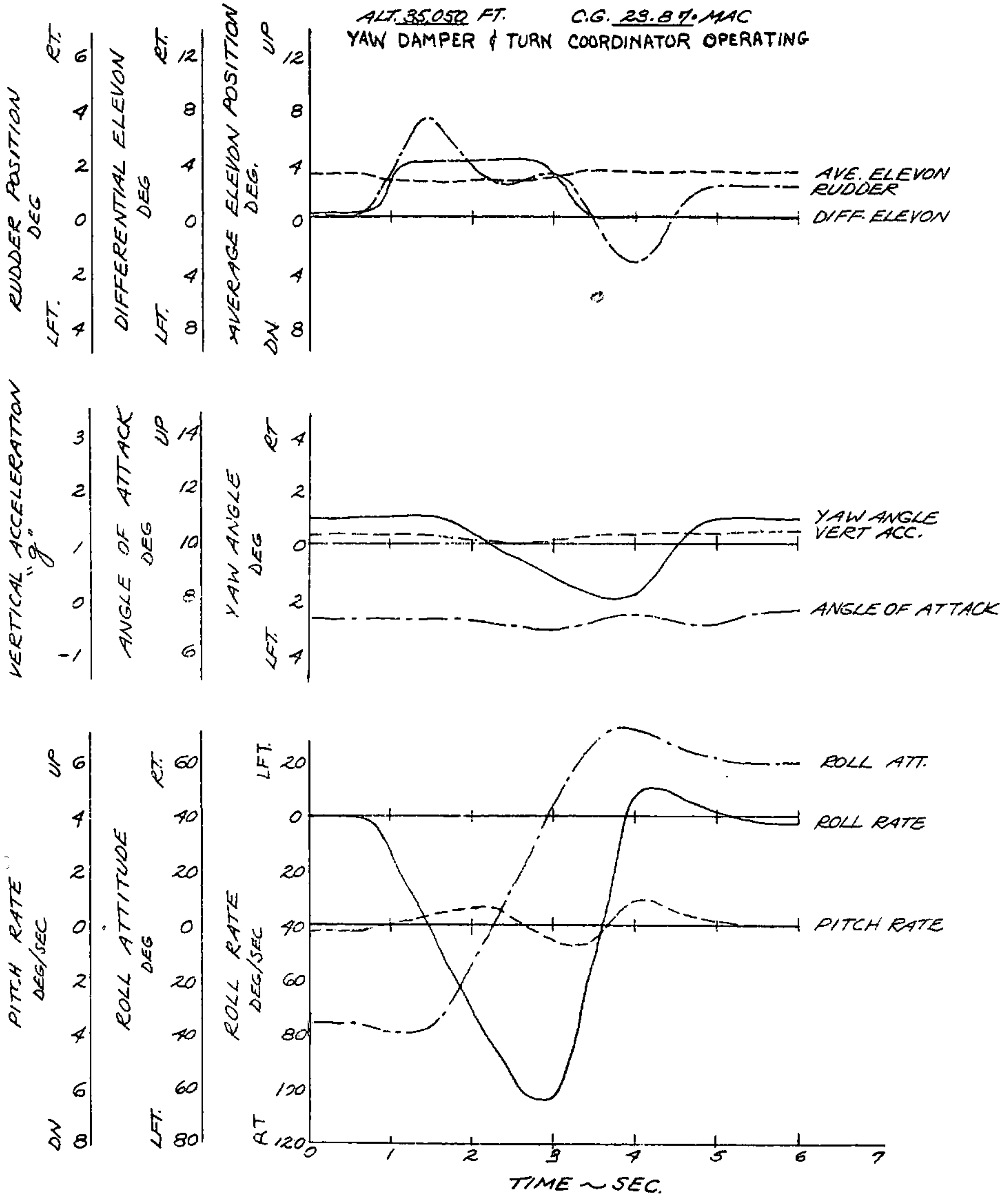


FIGURE 41
ROLL TIME HISTORIES
TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
TRIM CONDITIONS:
WT. 24,400 LBS. MACH NO. .72
ALT. 33,450 FT. C.G. 23.8% MAC
YAW DAMPER & TURN COORDINATOR OPERATING

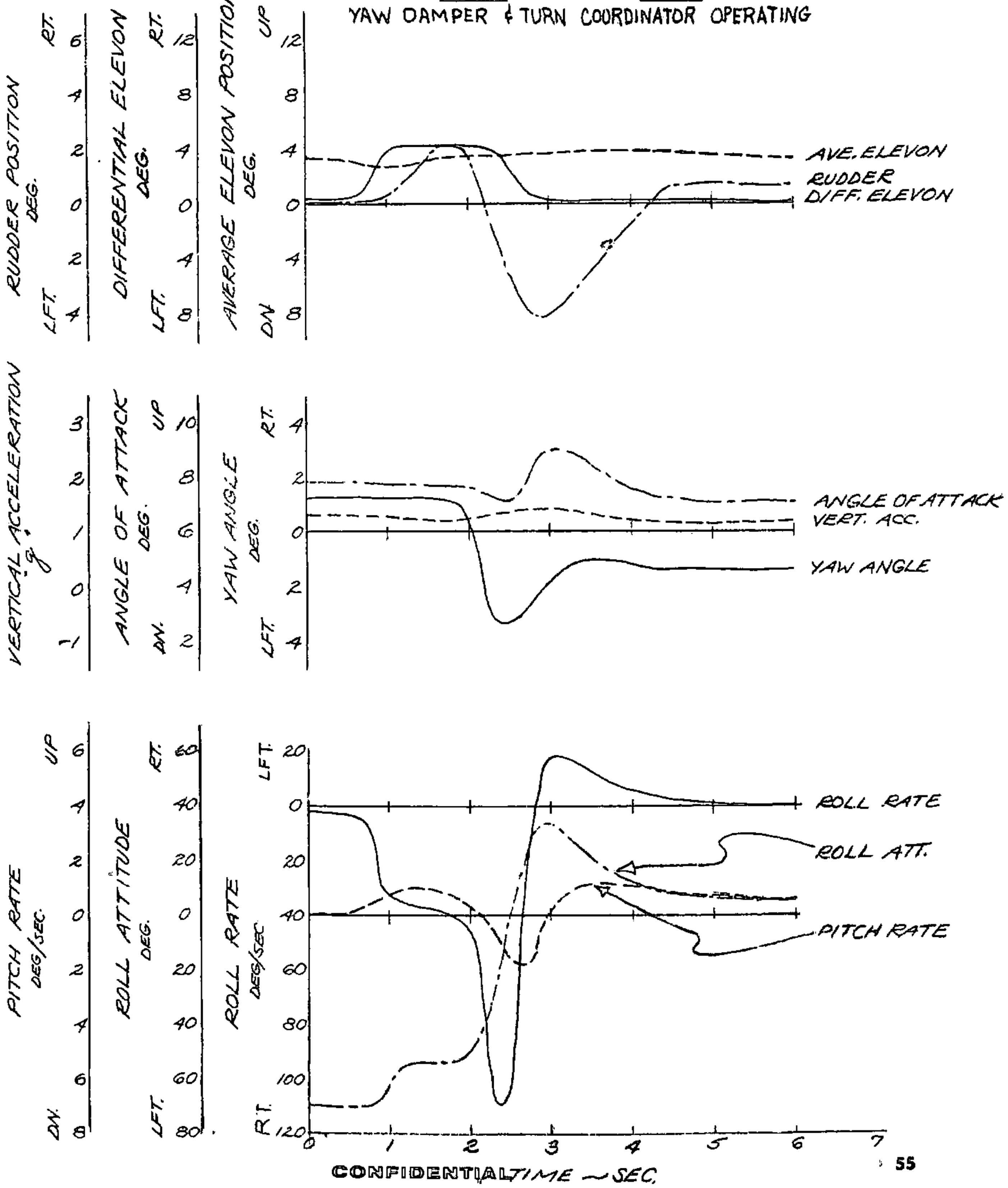


FIGURE 42

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351

CRUISE CONFIGURATION

TRIM CONDITIONS:

WT. 24,400 LBS. MACH NO. .73

ALT. 33,500 FT. C.G. 23.8% MAC

YAW DAMPER & TURN COORDINATOR OPERATING

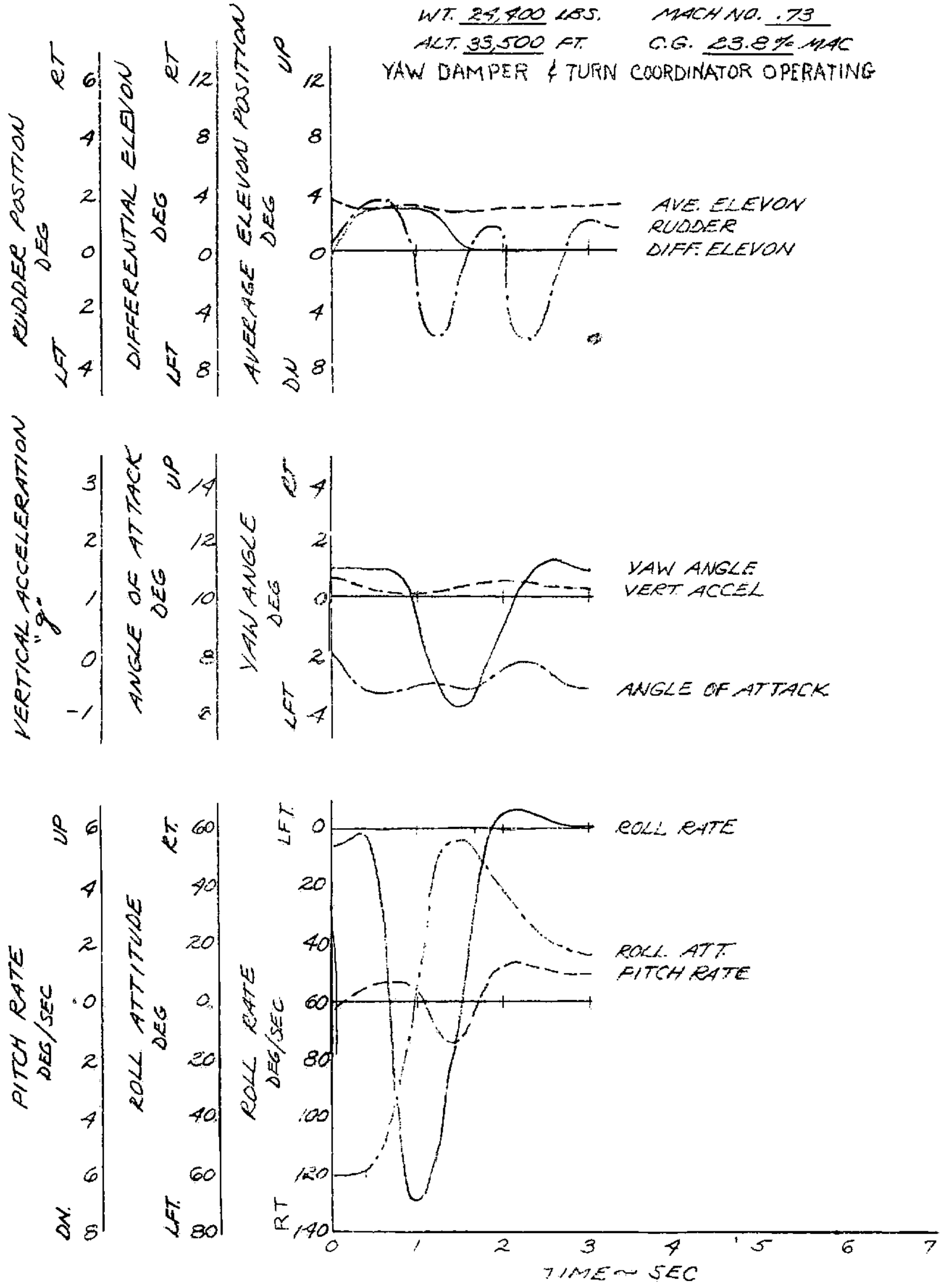


FIGURE 43

ROLL TIME HISTORIES

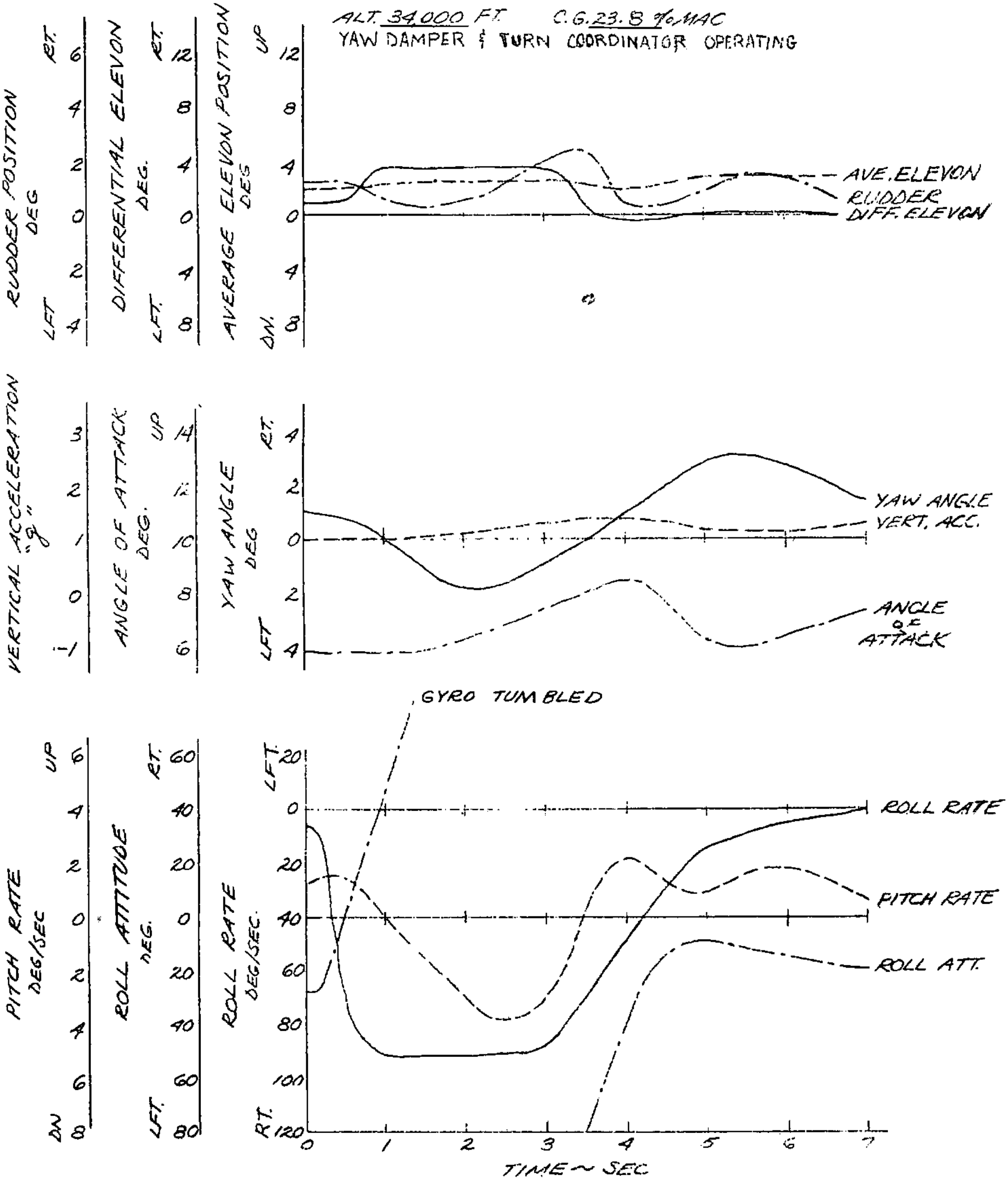
TF-102 A USAF S/N 54-1351

CRUISE CONFIGURATION

TRIM CONDITIONS:

WT. 24,200 LBS. MACH NO. .73

ALT. 34,000 FT. C.G. 23.8 % MAC



YAW DAMPER & TURN COORDINATOR OPERATING

FIGURE 44

ROLL TIME HISTORIES

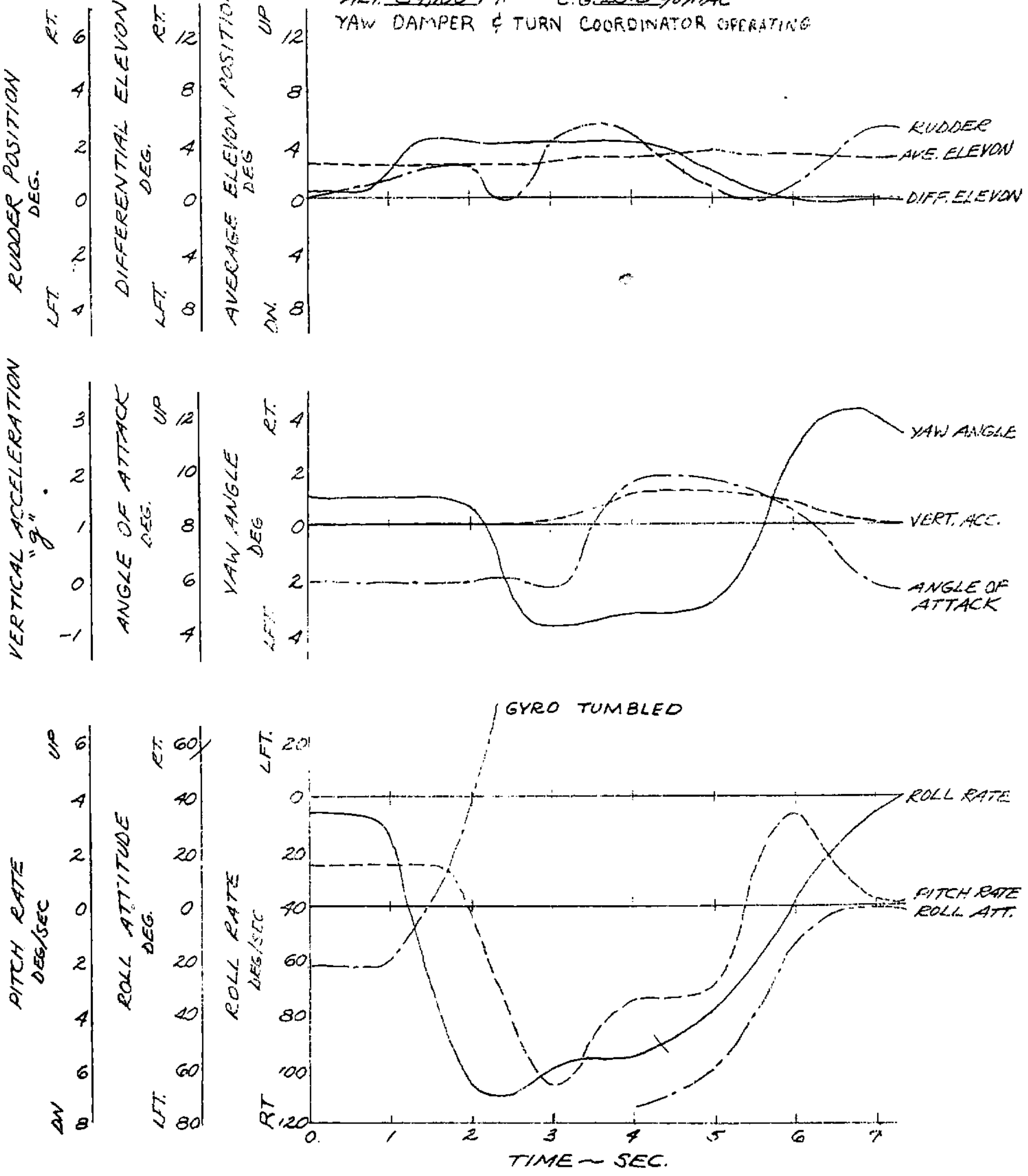
TF-102 A USAF S/N 54-1351

CRUISE CONFIGURATION

TRIM CONDITIONS:

WT. 24200 LBS. MACH NO. .79

ALT. 34,100 FT. C.G. 23.8 % MAC


YAW DAMPER & TURN COORDINATOR OPERATING

GYRO TUMBLED

FIGURE 45

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351
CRUISE CONFIGURATION
TRIM CONDITIONS:
WT. 23,850 LBS. MACH NO. .91
ALT. 34,250 FT. C.G. 23.7% MAC
YAW DAMPER & TURN COORDINATOR OPERATING

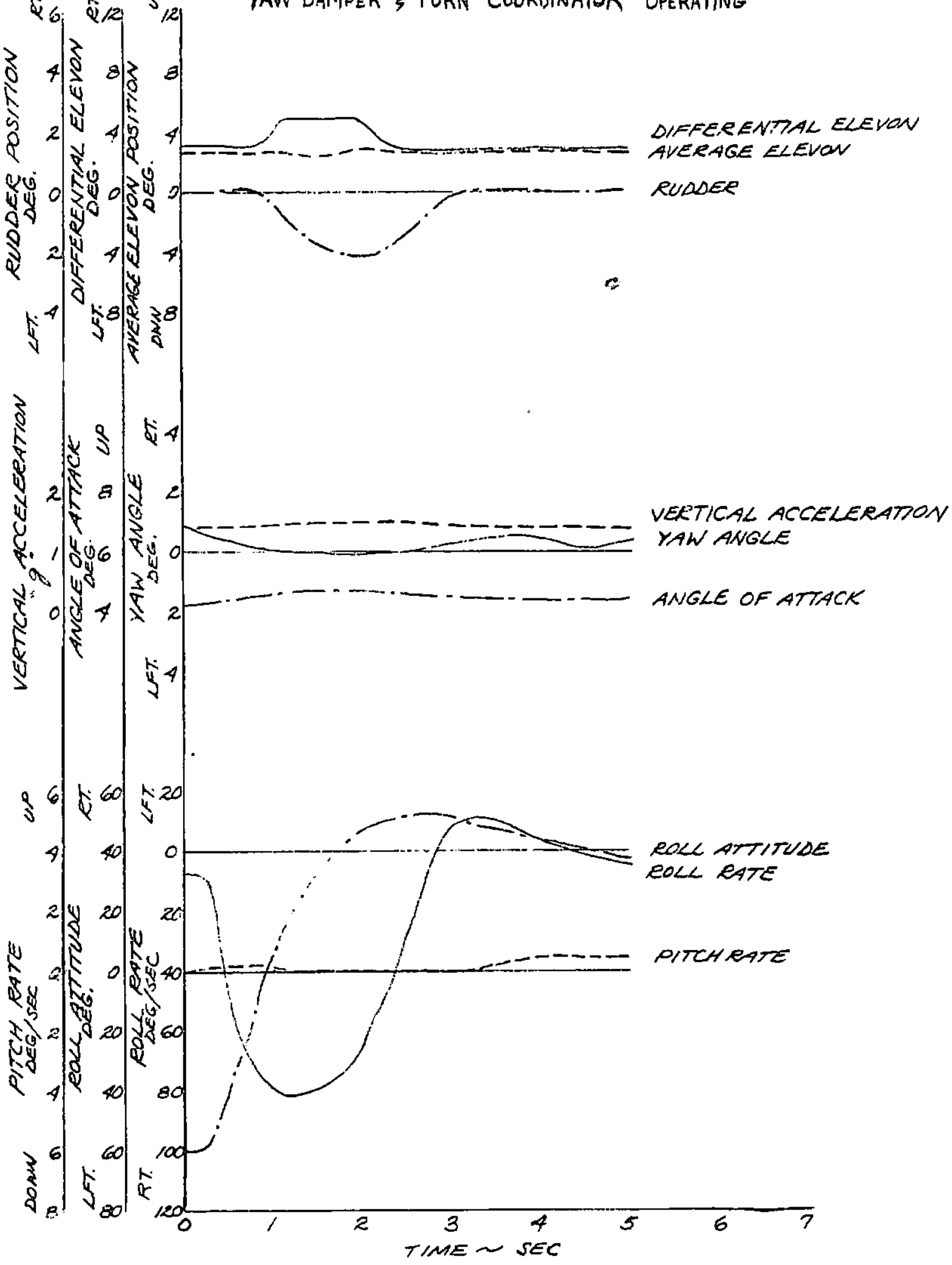


FIGURE NO. 46

ROLL TIME HISTORIES

TF-102A USAF NO. 54-1351

CRUISE CONFIGURATION

TRIM CONDITIONS:

WT. 23,850 MACH NO. .91

ALT. 34,600 C.G. 23.8% MAC

YAW DAMPER ¢ TURN COORDINATOR OPERATING

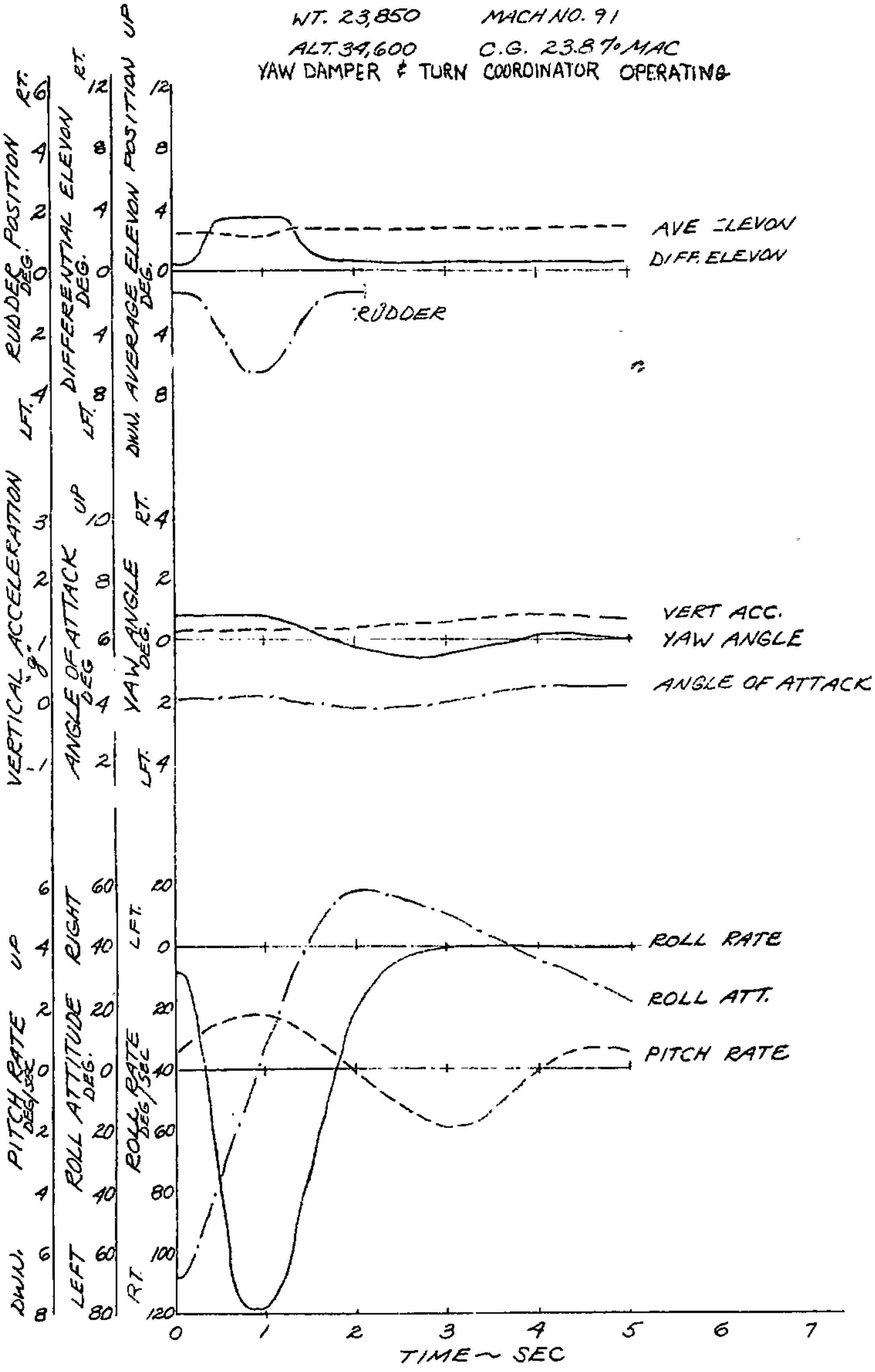


FIGURE 47

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351

CRUISE CONFIGURATION

TRIM CONDITIONS:

WT. 23,800 LBS. MACH NO. .91

ALT. 35,200 FT. C.G. 29.8% MAC

YAW DAMPER & TURN COORDINATOR OPERATING

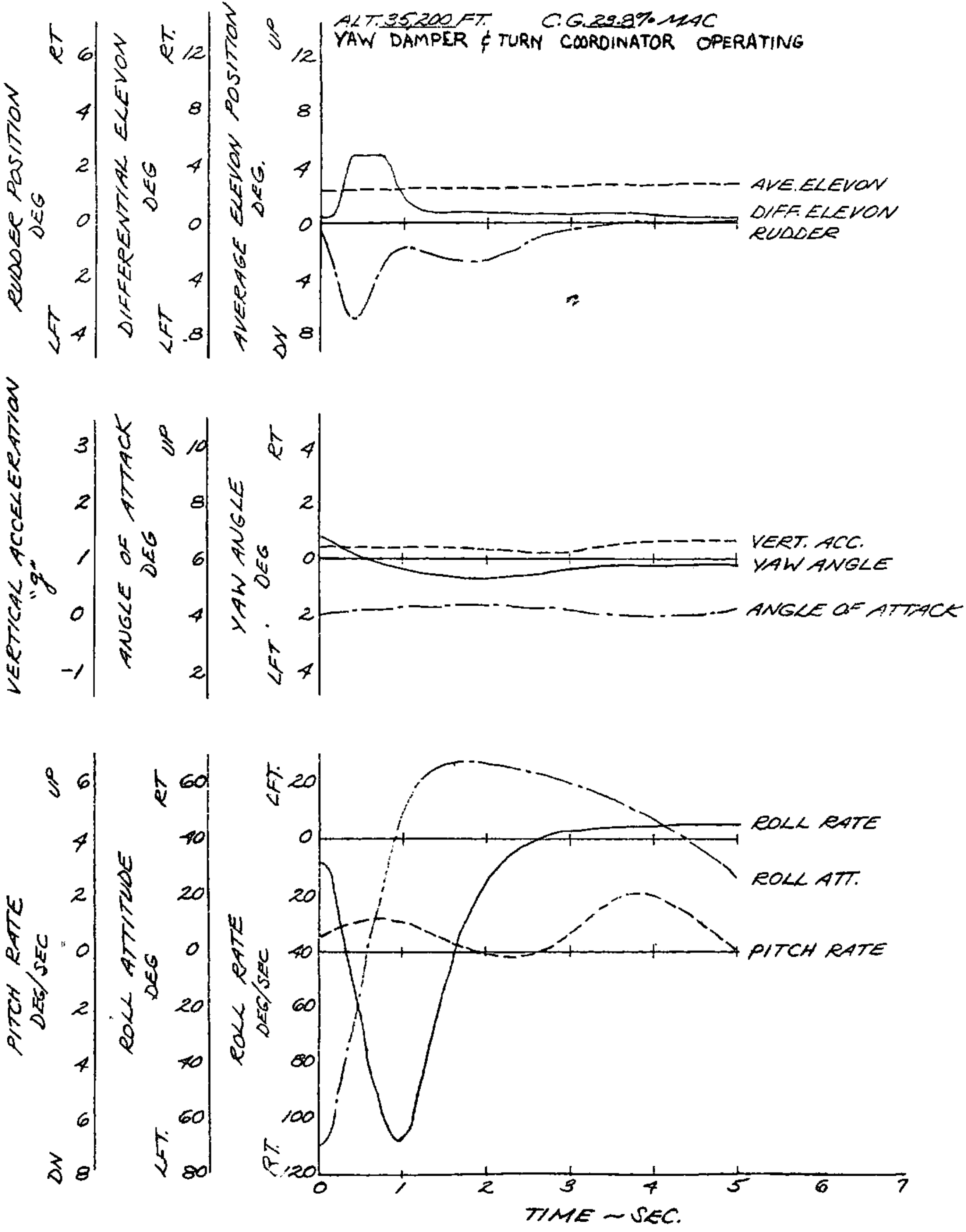


FIGURE 48

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351

P.A CONFIGURATION

TRIM CONDITIONS:

WT. 26,450 LBS. MACH NO. .37

ALT. 13,000 FT. C.G. 23.8% MAC

YAW DAMPER & TURN COORDINATOR OPERATING

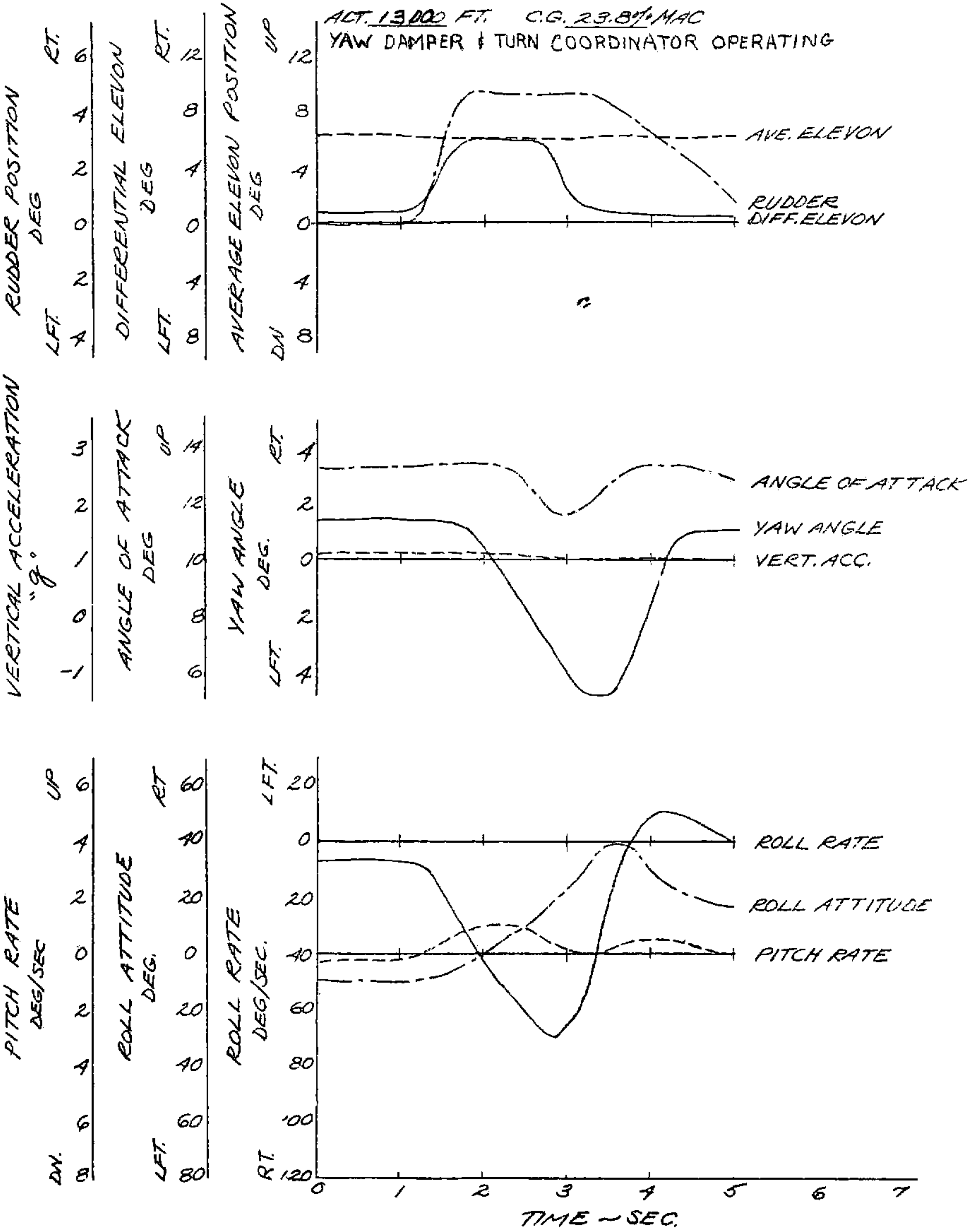


FIGURE 49

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351

P.A. CONFIGURATION

TRIM CONDITIONS:

WT. 26,450 LBS. MACH NO. .85

ALT. 13,400 FT. C.G. 23.4% MAC

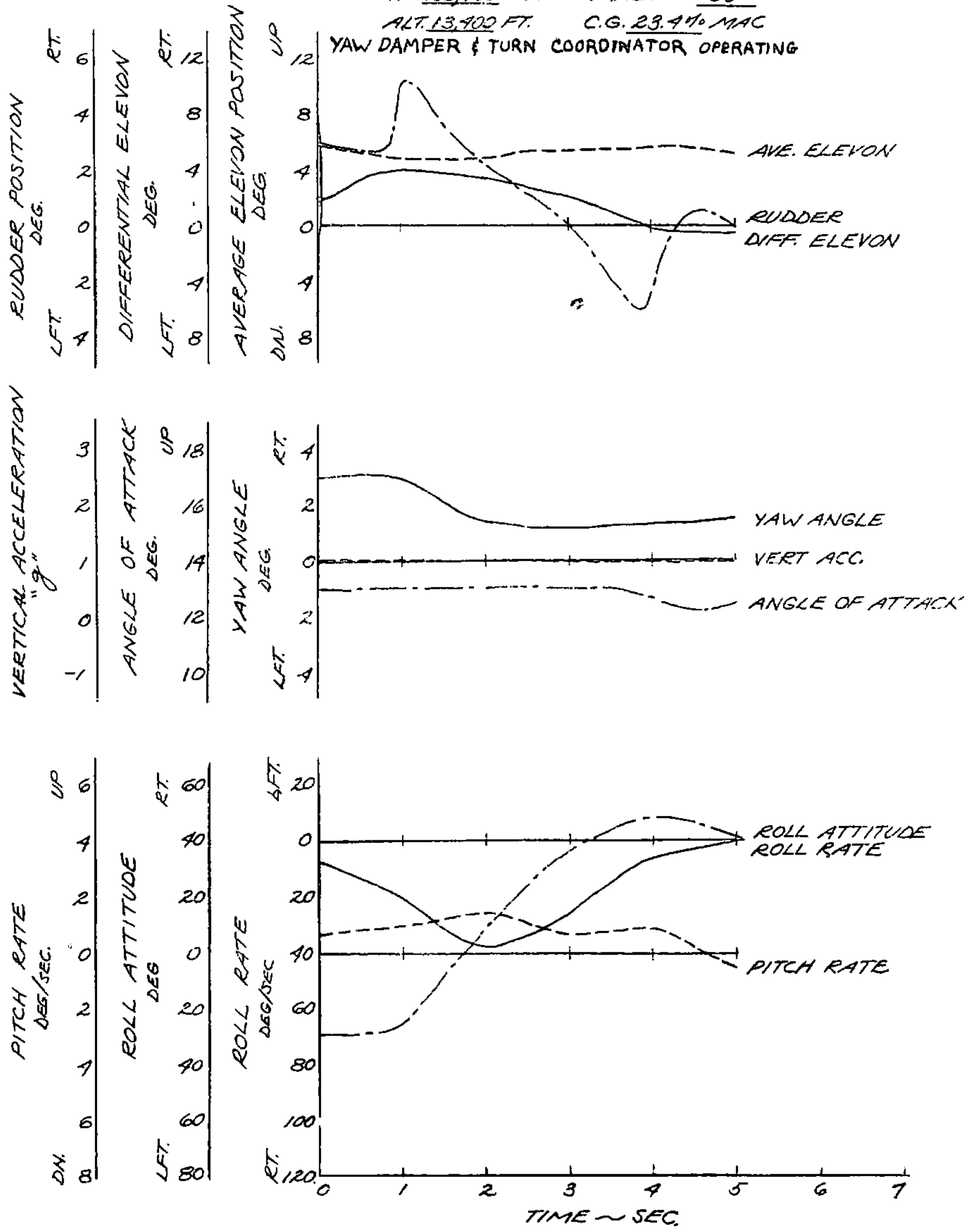


FIGURE 50

ROLL TIME HISTORIES

TF 102A USAF S/N 54-1351
P.A. CONFIGURATION
TRIM CONDITIONS:
WT. 26,450 LBS. MACH NO. .85
ALT. 11,960 FT. C.G. 23.4% MAC
YAW DAMPER & TURN COORDINATOR OPERATING

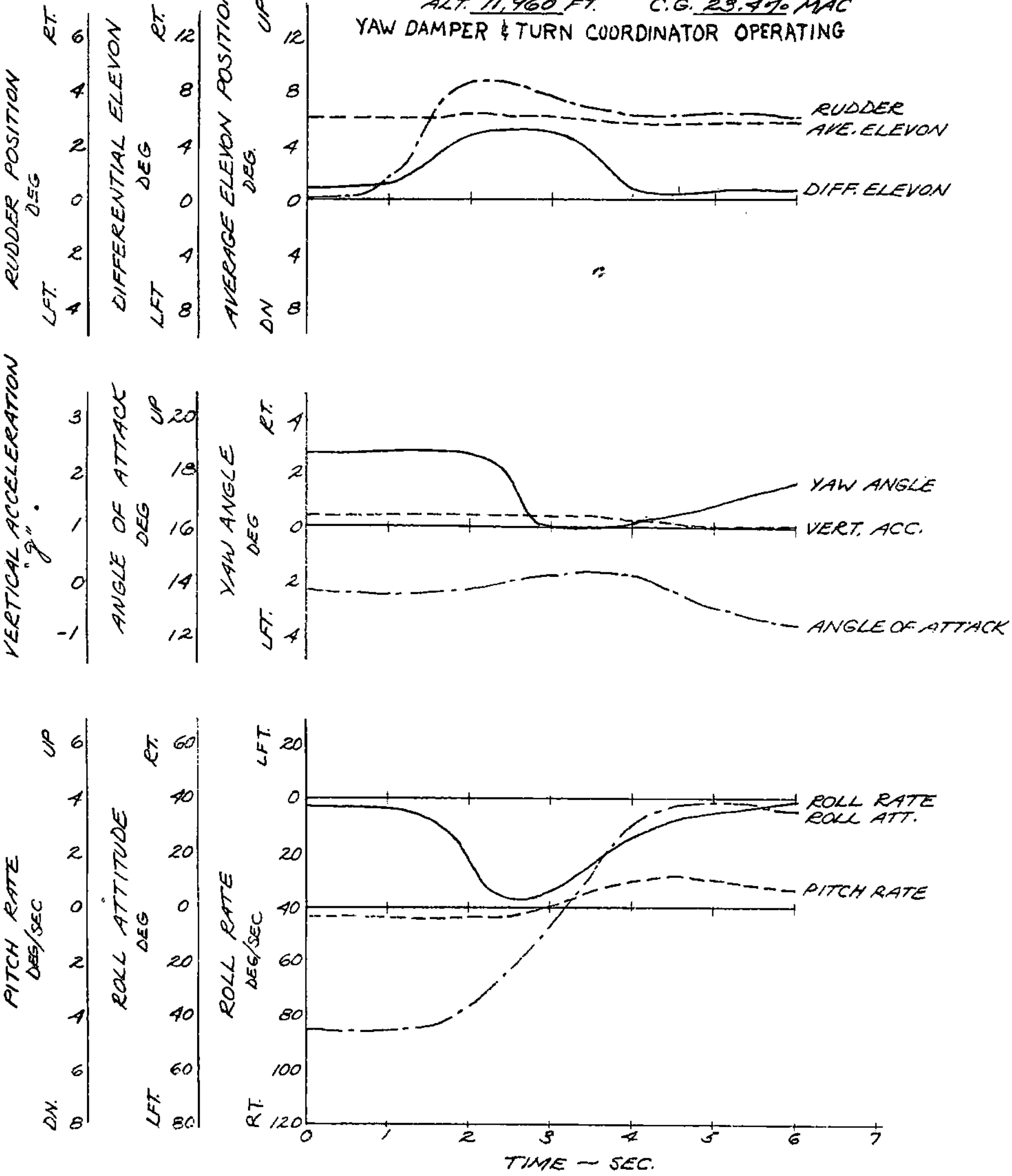


FIGURE 51

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351
P.A. CONFIGURATION
TRIM CONDITIONS:
WT. 25,450 LBS. MACH NO. .35
ALT. 14,200 FT. C.G. 23.4 % MAC
YAW DAMPER & TURN COORDINATOR OPERATING

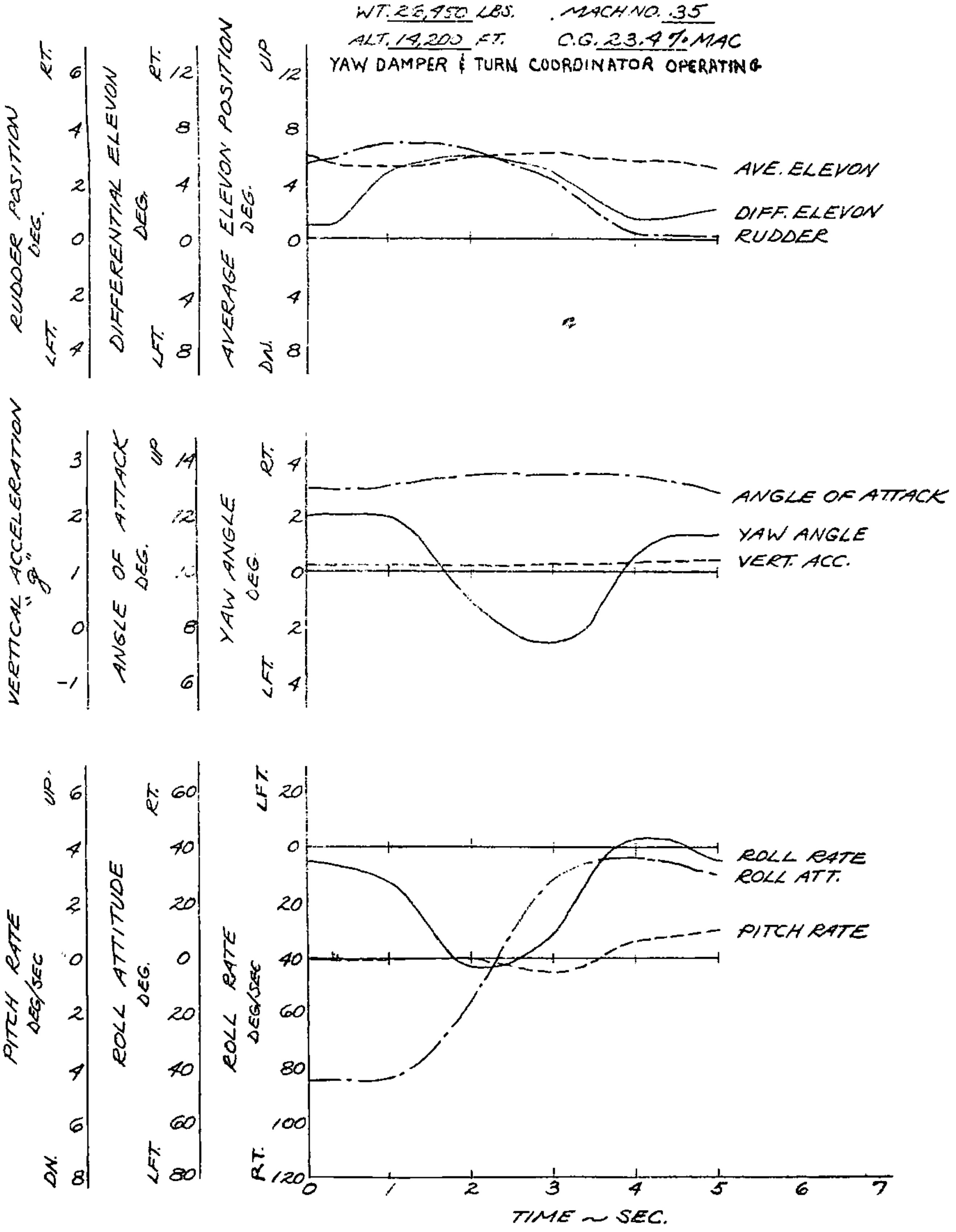


FIGURE 52

ROLL TIME HISTORIES

TF-102A USAF S/N 54-1351
P.A. CONFIGURATION
TRIM CONDITIONS:
WT. 26,450 LBS. MACH NO. .35
ALT. 13,200 FT. C.G. 23.4% MAC
YAW DAMPER & TURN COORDINATOR OPERATING

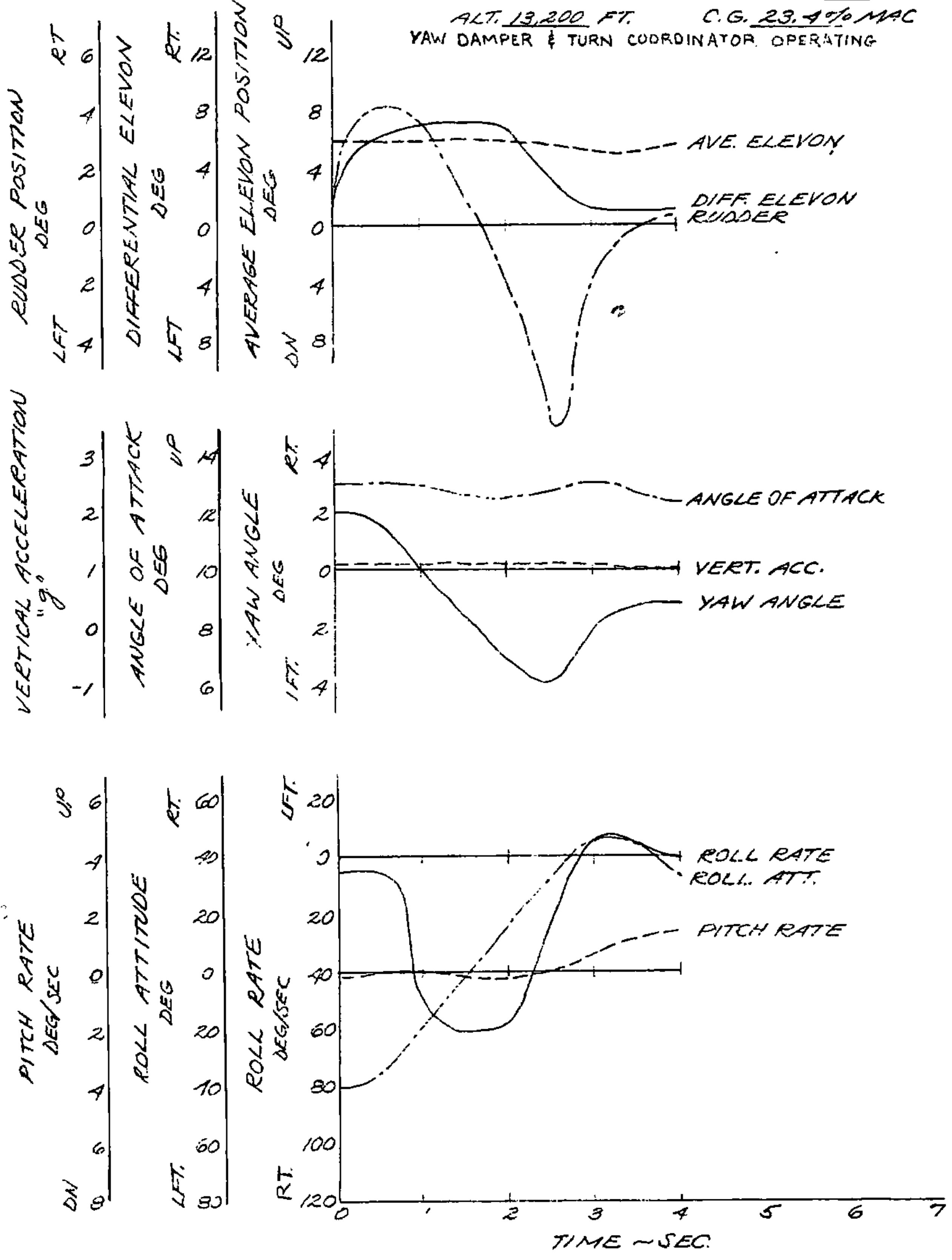


FIGURE 53

ROLL POTENTIAL

TF-102A USAF S/N 54-1351

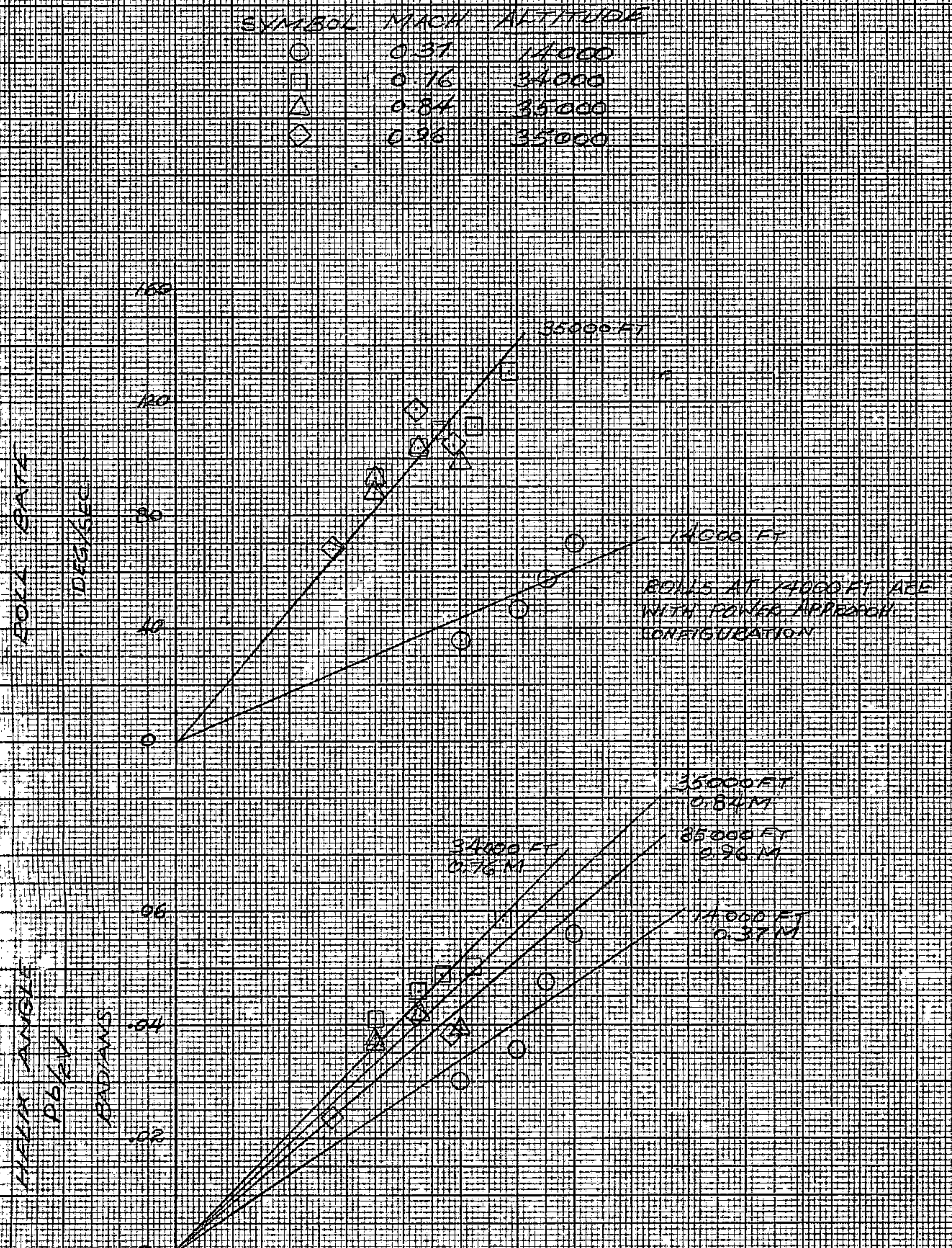
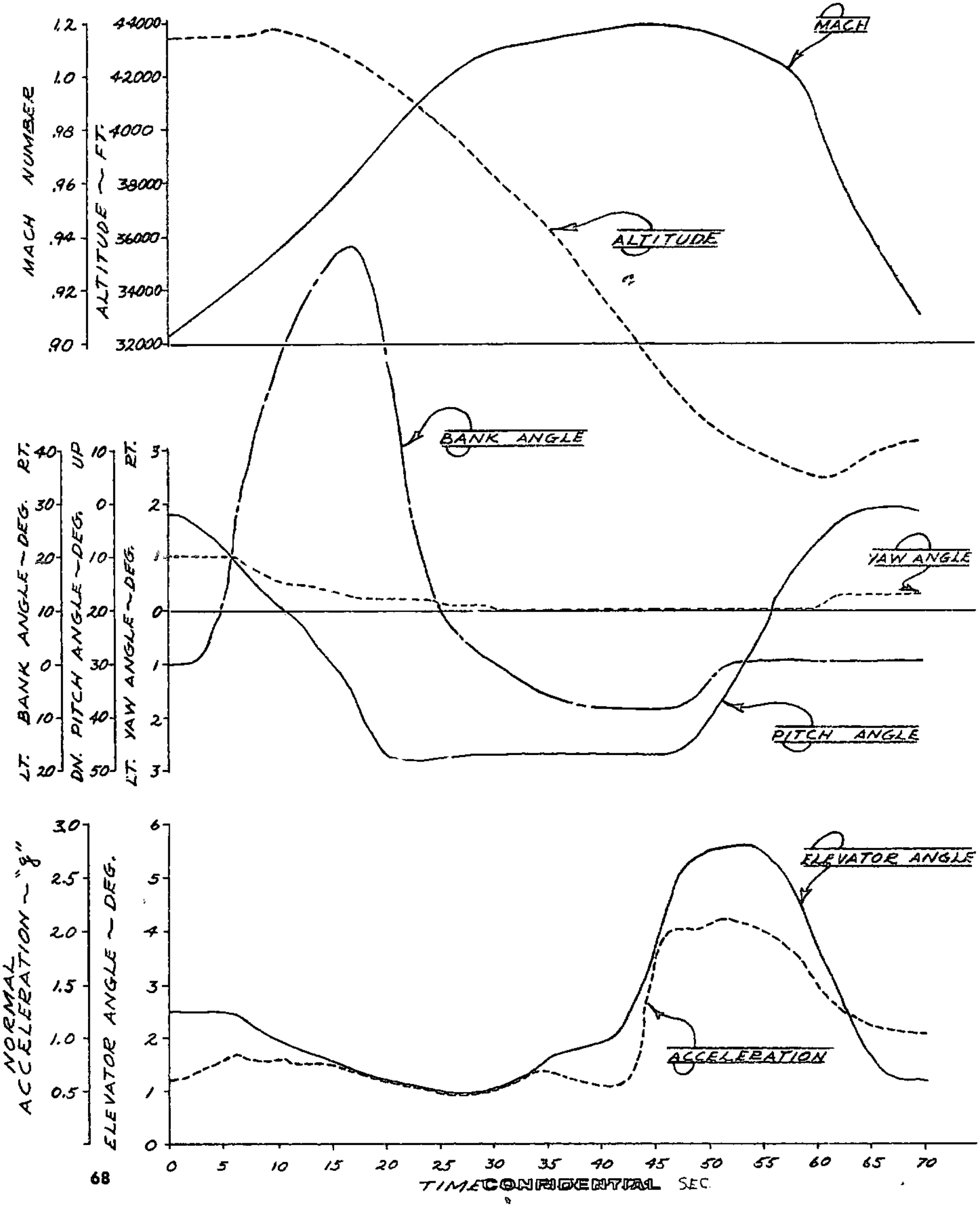


FIGURE 54
DIVE CHARACTERISTICS
TF-102A USAF S/N 54-1351

TRIM CONDITIONS:
WT 24,625 LBS C.G. 23.8 % MAC
MACH NO. .904 ALT. 43400 FT.



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APPENDIX II

general aircraft information

☐ aircraft dimensions

Wing	
area	695.05 sq ft.
span	38.16 ft.
MAC	23′ 9.1″
aspect ratio	2.189
airfoil section	NACA 0004-65 (Modified)
incidence	0°
dihedral	0°
leading edge sweepback	60°
Vertical Tail	
area	68.33 sq ft.
aspect ratio	1.10
airfoil	NACA 0004-65 (Modified)
leading edge sweepback	60°
Fuselage	
length (nose boom)	63.425 ft.
Elevons	
area	68 sq ft.
span (each)	12′10.75″
deflection	34° up; 19° down
Rudder	
area	7.43 sq ft.
deflection	25° right or left

☐ flight limitations

Engine	
maximum power	1 minute on ground 15 minutes in flight
military power	5 minutes on ground 30 minutes in flight
exhaust gas temperature limit	630°C below 35,000 ft. 660°C above 35,000 ft.
maximum high speed rotor rpm	102%
maximum low speed rotor rpm	106%
Airspeed	510 kts.
clean	1.35 Mach number
gear extended	240 kts.
drag chute	160 kts.
ram air turbine	345 kts.

Load Factors		
normal	+4.7g	−1.8g
asymmetrical	+3.1g	0.0g

Roll Limitations

Rolls limited to positive quadrant, and to one-half aileron within the positive quadrants.

Miscellaneous

Armament bay doors closed in flight.

Pilot assist may be used between 25,000 feet and 35,000 feet only.

☐ flight control system

Longitudinal and lateral control is accomplished by movement of the elevons in parallel motion as elevators, differentially as ailerons or in any combination of the two motions. Conventional stick movements are employed for elevon control. These control surfaces are actuated by irreversible full hydraulic power control systems which include artificial feel, gyro-stablized yaw and pitch damper systems and trimming devices. Pilot control motion mechanically meters hydraulic pressure to the control surface actuating cylinders. The control surfaces move only while the controls are being moved. When pilot control motion stops, a mechanical follow-up mechanism shuts off control valve.

A conventional rudder affords directional control. Actuation of the rudder surface is by an irreversible full hydraulic power system, including a gyro stabilized yaw damper system and a trimming device. The hydraulic power system functions in the same manner as described in the preceding paragraph.

Elevator feel force is made by a variable linkage. A pneumatic cylinder, modulated by dynamic pressure, provides a variation with airspeed. A trim servo automatic trimming device provides positive control in the unstable transonic speed range. Rudder feel is obtained from the combined action of an overload control spring assembly and a piston type feel cylinder which is loaded by amplified dynamic pressure. This system limits the lateral acceleration to a safe maximum yet provides adequate travel for maneuverability. Aileron feel is made directly proportional to control displacement by a simple spring device.

Trim is accomplished electrically by displacing the control surface positions and in turn repositioning the neutral point of the cockpit controls.

fuel system

All fuel is carried in right and left wing tank systems. Each system consists of three integral tanks. Except for a common, single-point refuel-defuel fitting the two wing systems are independent of each other and feed fuel to the engine through a common supply line either simultaneously or separately. Tank pressurization forces fuel from the aft outboard tank to the forward tank and then to the aft inboard tank where the booster pump is located. Total fuel capacity is 1110 gallons, 1084 gallons of which is usable fuel.

power plant

A Pratt and Whitney J57-P-23 turbo-jet engine with afterburner (flap-type nozzle, tandem cylinder, forged flap) is installed in this aircraft. The engine is rated at 16,000 pounds sea-level static thrust with afterburning and 9800 pounds non-afterburning.

weight and balance

The aircraft was ballasted to an engine start weight of 28,300 pounds including full fuel and oil, engineer and pilot. The center of gravity at engine start is 26.1 percent of the mean aerodynamic cord. Weight and center of gravity with aircraft empty of fuel are 21,200 pounds and 19.7 percent MAC.

test instrumentation

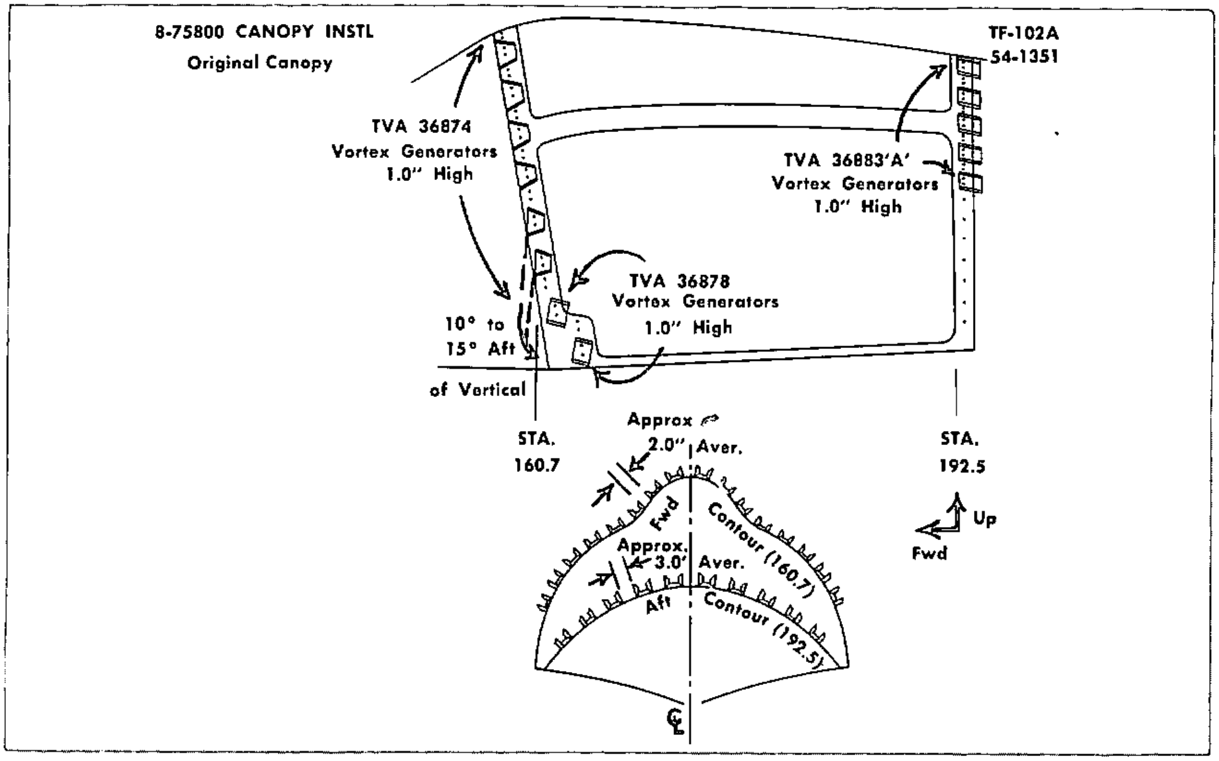
The following test instrumentation was used during the program:

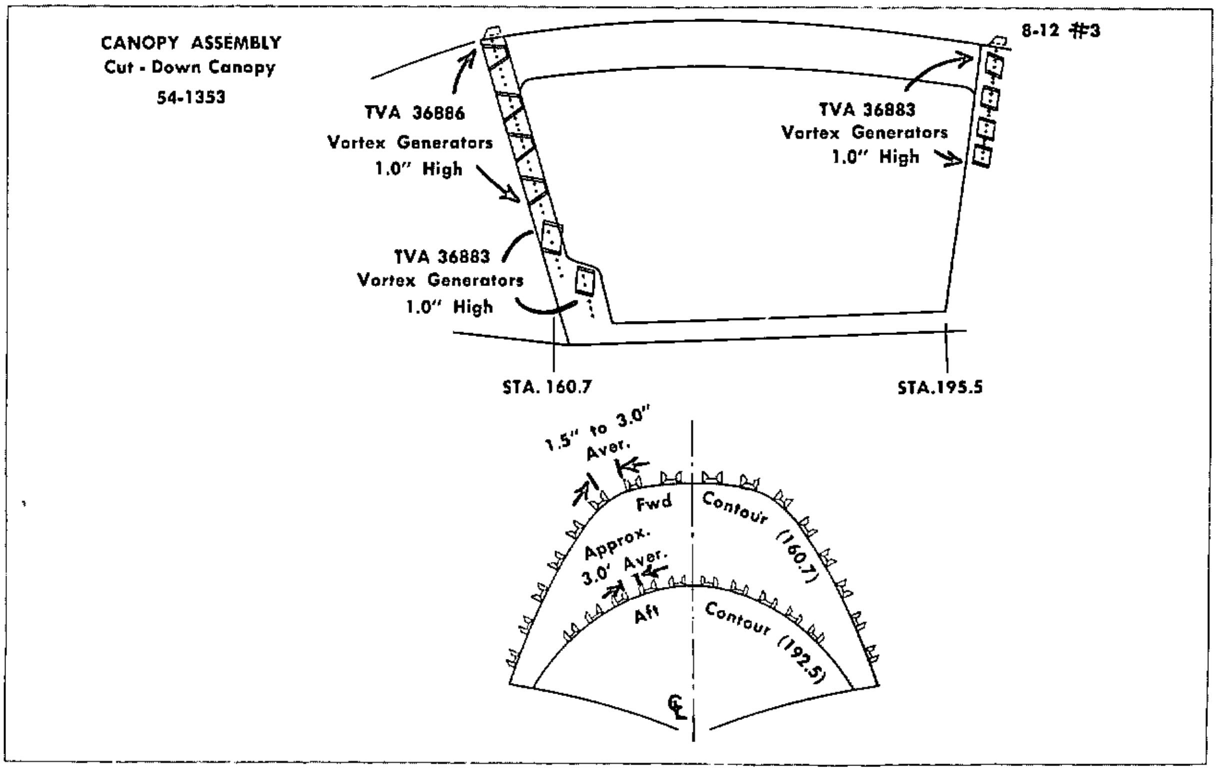
INSTRUCTOR'S PANEL
- AIRSPEED INDICATOR
- ALTIMETER
- RATE OF ROLL
- RATE OF CLIMB
- ATTITUDE
- DIRECTIONAL HEADING

STUDENT'S PANEL
- MACH INDICATOR
- AIRSPEED
- ALTIMETER
- NORMAL ACCELERATION
- RATE OF CLIMB
- RPM
- FUEL FLOW
- FUEL QUANTITY
- P_{t_7}/P_{t_0}
- OUTSIDE AIR TEMPERATURE
- DUAL TEMPERATURE INDICATOR
- YAW ANGLE
- ELEVON POSITION INDICATORS
- RUDDER POSITION INDICATOR

PHOTO PANEL (35mm PHOTO RECORDER)
- AIRSPEED(BOOM) INDICATOR
- ALTIMETER (BOOM)
- OUTSIDE AIR TEMPERATURE INDICATOR
- TIME (8-DAY CLOCK)
- TACHOMETER (LOW PRESSURE COMPRESSOR)
- TACHOMETER (HIGH PRESSURE COMPRESSOR)
- RATE OF ROLL INDICATOR
- ENGINE FUEL COUNTER
- ANGLE OF YAW
- BLEED VALVE AND AFTERBURNER INDICATOR
- VERTICAL ACCELERATION INDICATOR
- LEFT AND RIGHT HAND ELEVON INDICATOR
- ROLL ATTITUDE
- ALTIMETER INSTRUMENT STATIC INDICATOR
- MANIFOLD PRESSURE (P_{t_7})
- DUAL TEMPERATURE INDICATOR
- ANGLE OF ATTACK INDICATOR

OSCILLOGRAPH
- PITCH RATE
- ROLL RATE
- YAW RATE
- YAW ANGLE
- LEFT HAND ELEVON POSITION
- RIGHT HAND ELEVON POSITION
- RUDDER POSITION
- NORMAL ACCELERATION


8-75800 CANOPY INSTL
Original Canopy
TF-102A
54-1351
TVA 36874
Vortex Generators
1.0" High
TVA 36883'A'
Vortex Generators
1.0" High
TVA 36878
Vortex Generators
1.0" High
10° to
15° Aft
of Vertical
STA.
160.7
STA.
192.5
Approx
2.0" Aver.
Fwd
Contour (160.7)
Approx.
3.0' Aver.
Aft
Contour (192.5)
Up
Fwd
℄


CANOPY ASSEMBLY
Cut - Down Canopy
54-1353
8-12 #3
TVA 36886
Vortex Generators
1.0" High
TVA 36883
Vortex Generators
1.0" High
TVA 36883
Vortex Generators
1.0" High
STA. 160.7
STA.195.5
1.5" to 3.0"
Aver.
Fwd
Contour (160.7)
Approx.
3.0' Aver.
Aft
Contour (192.5)
℄

JOINT MESSAGEFORM "FTFEE-3-56-E. Para 1. Four qualitative flights have been conducted on the TF-102 to evaluate the effect of flow regenerators on the leading edge of the canopy. Initial buffet was encountered at 0.82 to 0.83 indicated Mach number, building up in intensity to approximately 0.88 IMN. The buffet disappeared at 0.88 to 0.89 IMN and was non exisent up to speeds where the duct buzz became apparent in the region of 1.2 IMN. In accelerated turns at 35,000 feet, buffet commenced at approximately 1.4 "G" at 0.7 IMN, 1.25 "G" at 0.79 IMN and 1.6 "G" at 0.93 IMN. Intensity of the buffet increased with an increase in dynamic pressure and very rapidly with increases in load factor. Para 2. It is concluded that the buffet problem is created by flow separation around the canopy and is reduced by the flow regenerators so that it is no longer a safety of flight item. The buffet is objectionable and must be elminated. Para 3. It is recommended that this aircraft not be accepted for general use in the present configuration. It may be advantageous, however, to accept a few articles for critical development areas."

JOINT MESSAGEFORM "FTFEE-5-58-E Attn: RDZ-LSFG. Subj: Evaluation of TF-102A Serial Number 54-1351 and 54-1353. /1/ Evaluation of the TF-102A aircraft has been completed in ten flights of 1351 and eleven flights of 1353 by a joint AFFTC - ADC evaluation team. Airplane 1351 had the orginal canopy with a double bank of 1 inch vortex generators installed according to Convairs TVA 36883 and ECP 36886. Airplane 1353 had the first cut down canopy with the same vortex generators. Neither aircraft had a radar or a glare shield. Airplane 1353 weighted 28,560 pounds with a CG of 26.8 percent MAC while 1351 weighed 28,560 pounds at 26.4 percent MAC at take-off. /2/ Comparative performance was obtained by flying the aircraft together in climbs and top speeds with military and with maximum power. Buffet was determined qualitatively in level flght and with maneuvering load factors. The climb performance with maximum power was comparable one with the other. Time to climb from 10,000 to 50,000 feet was 7 minutes and 40 seconds for each aircraft with the same test conditions. Top level flight speeds with maximum power were comparable, .92 indicated Mach number (.98 true Mach number) at 35,000 feet. The flap type nozzle with single actuators tended to remain open when coming out of afterburner. The eyelids would indicate closed after the power setting was reduced, but nozzle pressure ratios at military power settings indicated that the eyelids were not fully closed. Aircraft 1351 indicated less thrust at military power than 1353, and level flight speeds at these power settings indicated that 1351 was 10 to 20 knots slower than 1353. /3/ Buffet in 1351 was found to be annoying but not unacceptable. Buffet was encountered in unaccelerated conditions at 0.82 indicated Mach number increasing to maximum intensity at .85 indicated Mach number, and decreasing in intensity until it disappeared at approximately 0.9 IMN. Buffet in the cut down canopy of 1353 was barely noticeable at 0.88 to 0.9 IMN during unaccelerated flight and non-existent at other speeds. /4/ Buffet in accelerated flight was disconcerting in 1351. At 0.8 to 0.85 indicated Mach number and 42,500 feet in 1351, buffet encountered at 1.2 g was such that radar tracking may be affected. At .75 indicated Mach number and 25,000 feet buffet is very noticeable at 2.0 g and objectionable at 3.5 g. Buffet is encountered with the cut down canopy of 1353 at 0.5 to 1.0 higher load factor than 1351. At 0.8 idicated Mach number and 47,500 feet buffet starts at 1.4 g. At .7 to .75 indicated Mach number and 25,000 feet buffet starts at approximately 2.5 g. /5/ Visibility in 1351 is definitely better than that encountered in 1353 with the cut down canopy. This deficiency in 1353 is predominantly noticed in the form of a blind spot over the nose of the ship during maximum power climbs. Information received from Convair TF-102A project office indicated a 4 month delay in delivery of the TF-102 if the cut down canopy was required. This modification would require in the neighborhood of $60,000 per airplane. /6/ It is concluded that (A) a negligible difference in performance is incurred with either canopy. (B) The cut down canopy with flow regenerators provides reduced buffet in accelerated and unaccelerated conditions, and restricts the buffet to a smaller Mach number range and to a higher load factor. These factors psychologically may provide better training. (C) The cut down canopy does reduce the visibility. (D) The old canopy with flow regenerators is an acceptable configuration with the buffet an annoyance factor and not a physical limitation. (E) A delay in delivery of approximately 4 months and an additional cost of $60,000 per airplane make the old canopy airplane more desirable. /7/ It is recommended that (A) the old canopy with dual bank flow regenators be accepted to permit earlier delivery of the TF-102A airplane. (B) Close quality control be maintained to insure that no aircraft be delivered with buffet worse than encountered in 1351 during these tests. (C) A continuing effort be made by Convair to completely eliminate the buffet in 54-1351."

APPENDIX III

test data corrected for instrument error

TAKE-OFFS 74
CHECK CLIMBS (MAXIMUM POWER) 76
TURN CAPABILITIES 77
LEVEL FLIGHT (NON-AFTERBURNING) 77
LEVEL FLIGHT (AFTERBURNING) 78
THRUST RUN 79
AIRSPEED CALIBRATION 80
LANDINGS 80

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.47 IN. HG t_a = 16 °C
GROSS WEIGHT 28300 LBS. WIND 8 MPH AT 78 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	4204.16		32.5	35.26		49	4259.60	33.49
1	4196.90		33	90.17		49.5	4401.07	35.93
2	4160.32		33.5	201.58		50	4525.44	37.58
3	4154.41		34	316.10		50.5	4665.35	39.15
4	4122.28		34.5	445.13		51	4791.80	40.98
5	4082.38		35	561.21		51.5	4931.21	42.80
6	4039.37		35.5	692.32		52	5055.04	44.72
7	3924.59		36	802.43		52.5	5179.93	47.35
8	3909.92		36.5	929.45		53	5320.36	50.53
9	3836.25		37	1064.90		53.5	5446.86	53.84
10	3753.84		37.25	1125.53	T.O.	54	5584.64	56.09
11	3665.23		37.5	1186.16	-.52	54.5	5711.60	61.23
12	3562.81		38	1321.93	-.29	55	5854.11	64.99
13	3464.17		38.5	1443.71	-.14	55.5	5978.47	68.91
14	3357.94		39	1582.08	.00154	56	6104.90	71.92
15	3240.34		39.5	1724.88	1.54	56.5	6246.97	74.09
16	3114.90		40	1844.22	2.82	57	6373.96	75.39
17	2992.26		40.5	1966.57	4.51	57.5	65[illegible]	77.78
18	2841.81		41	2104.41	6.16	58	6645.92	79.72
19	2696.71		41.5	2228.26	8.69	58.5	6789.96	81.49
20	2540.22		42	2364.59	10.41	59	6913.31	83.66
21	2342.26		42.5	2488.91	12.02	59.5	7056.33	85.26
22	2173.95		43	2626.29	13.39	60	7193.81	87.61
23	1998.16		43.5	2751.12	14.99	60.5	7327.35	90.02
24	1815.25		44	2874.96	15.57	61	7450.16	91.80
25	1626.63		44.5	3015.92	16.58	61.5	7591.63	94.44
26	1432.92		45	3112.02	18.56	62	7721.18	96.18
27	1233.32		46	3465.72	21.50	62.5	7862.83	97.90
28	1025.55		46.5	3606.15	23.15	63	7996.86	99.13
29	827.05		47	3730.00	25.62	63.5	8206.22	101.15
30	608.40		47.5	3870.95	27.95	64	8334.73	102.29
31	396.58		48	3994.29	29.66	64.5	8481.90	103.38
32	156.50		48.5	4135.25	32.03	65	8611.45	105.07

* 45.5 3341.87 19.69

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.47 IN. HG T_a = 16 °C
GROSS WEIGHT 28300 LBS. WIND 8 MPH AT 78 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
65.5	8762.24	105.54						
66	8891.28	106.59						
66.5	9024.45	107.46						
67	9175.25	109.95						
67.5	9312.57	111.03						
68	9459.22	111.53						
68.5	9598.10	113.41						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAX. P_a = 27.53 IN. HG t_a = 6 °C
GROSS WEIGHT 28300LBS. WIND 9 MPH AT 183 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	4289.74		26.5	9.32	30.14			
1	4276.80		27	127.43	35.51			
2	4256.08		27.5	220.89	41.88			
3	4217.75		28	457.88	52.31			
4	4165.95		29	715.82	69.52			
5	4104.83		30	1008.99	92.40			
6	4028.69		31	1326.50	119.67			
7	3944.27		32	1639.86	146.60			
8	3846.37		33	1949.60	176.14			
9	3737.08							
10.5	3544.40							
11	3476.62							
12	3327.38							
13	3173.54							
14	3003.65							
15	2816.62							
16	2628.65							
17	2421.46							
18	2217.90							
19	1995.34							
20	1804.57							
20.5	1630.54							
21	1511.93							
21.5	1386.58							
22	1260.20							
22.5	1118.79	2.81						
23	990.34	5.08						
23.5	859.81	7.58						
24	729.29	9.66						
24.5	580.12	13.45						
25	393.65	15.98						
25.5	303.01	19.35						
26	165.23	24.20						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAX. P_a = 27.445 IN. HG t_a = 6 °C
GROSS WEIGHT 28300 LBS. WIND 16 MPH AT 818 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	Distance Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	4318.68		23	624.95	41.55			
1	4303.65		23.5	490.74	49.80			
2	4224.11		24	338.38	60.55			
3	4232.14		24.5	203.65	71.03			
4	4128.25		25	62.20	82.29			
5	4106.22							
6	3998.43							
7	3888.57							
8	3229.75							
9	3656.42							
10	3572.98							
11	3356.90							
12	3186.73							
13	3006.08							
14	2821.60							
15	2614.32							
15.5	2511.20							
15.75	2459.90							
16	2394.08							
16.5	2286.82							
17	2176.96							
17.25	2121.51	T.O.						
17.5	2019.48	1.14						
18	1920.45	2.22						
18.5	1812.85	3.29						
19	1684.15	5.25						
19.5	1562.89	6.12						
20	1424.53	8.41						
20.5	1302.24	11.86						
21	1171.24	15.44						
21.5	1032.25	21.00						
22	903.22	22.04						
22.5	772.64	32.52						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.415 IN. HG t_a = 13.3 °C
GROSS WEIGHT 28300 LBS. WIND 12 MPH AT 24 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
1	4315.05		31	421.30		47	3650.20	44.20
2	4301.06		31.5	309.37		47.5	3777.65	46.86
3	4280.95		32	179.30		48	3901.64	48.46
4	4249.24		32.5	64.78		48.5	4050.77	51.33
5	4211.93		33	52.34		49	4180.94	52.90
6	4162.18		33.5	193.96		49.5	4304.35	54.87
7	4104.66		34	301.59		50	4452.56	56.98
8	4042.48		34.5	436.32				
9	3966.82		35	555.96				
10	3884.95		35.25	615.60	T.O.			
11	3792.71		35.5	675.21	.25			
12	3692.88		36	810.46	1.95			
13	3589.57		36.5	929.65	3.03			
14	3472.98		37	1051.95	5.39			
15	3355.35		37.5	1187.20	6.69			
16	3232.17		38	1308.97	8.57			
17	3082.25		38.5	1429.71	10.49			
18	2933.01		39	1552.55	12.33			
19	2789.94		39.5	1688.30	15.16			
20	2626.24		40	1810.59	17.69			
21	2456.27		40.5	1947.40	20.45			
22	2278.53		41	2069.69	22.65			
23	2105.45		41.5	2202.53	25.46			
24	1919.23		42	2328.74	28.30			
25	1716.80		42.5	2453.16	30.67			
26	1525.06		43	2592.04	32.42			
27	1314.16		43.5	2712.96	34.08			
28	1097.55		44	2842.33	36.12			
29	873.69		44.5	2983.80	37.17			
29.5	766.42		45	3097.72	39.16			
30	656.04		45.5	3235.12	40.04			
30.5	546.70		[illegible]	3356.90	41.40			
30.75	477.28		[illegible]	3494.11	42.45			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 2753 IN. HG t_a = 25.0 °C
GROSS WEIGHT 28300 WIND CALM MPH AT CALM DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	10103.88		33	5543.70		55	520.37	47.91
1	10066.28		34	5287.19		56	819.27	52.42
2	10059.23		35	5029.13		57	1139.00	65.87
3	[illegible]		35.5	4904.76				
4	10042.10		36	4781.95				
5	10013.39		36.5	4643.07				
6	9952.57		37	4512.15				
7	9916.28		37.5	4378.27				
8	9544.95		37.75	4331.12				
9	9743.41		38	4253.39	.17			
10	9671.17		38.5	4127.98	.30			
11	9549.56		39	4004.65	0			
12	9468.55		39.5	3858.04	.5			
13	9350.92		40	3732.08	.02			
14	9218.62		40.5	3604.60	.82			
15	9099.59		41	3459.50	.80			
16	8964.86		41.5	3331.51	1.60			
17	8816.65		42	3201.96	1.72			
18	8661.19		42.5	3056.34	2.56			
19	8505.22		43	2926.28	3.32			
20	8335.25		43.5	2792.24	4.38			
21	8155.54		44	2649.56	6.18			
22	7972.51		44.5	2512.42	7.51			
23	7780.10		45	2370.77	9.48			
24	7591.63		46	2105.96	12.76			
25	7385.39		47	1825.10	16.10			
26	7157.22		48	1540.09	18.90			
27	6962.30		49	1269.59	21.39			
28	6737.64		50	980.43	24.20			
29	6512.40		51	686.10	28.33			
30	6229.03		52	402.64	32.59			
31	6033.92		53	100.53	37.11			
32	5793.63		54	208.32	43.69			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM P_a = 27.51 IN. HG t_a = 5.6 °C
GROSS WEIGHT 28300 WIND 8.5 MPH AT 12.0 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	10219.42		26.5	5508.98	24.00			
1	10198.18		27	5359.27	28.90			
2	10188.65		27.5	5187.18	34.04			
3	10118.32		28	5033.79	40.48			
4	10056.19		29.5	4876.78	47.89			
5	9983.64		29	4702.15	57.08			
6	9893.99		29.5	4543.58	66.96			
7	9790.87		30	4361.17	78.00			
8	9685.16							
9	9568.56							
10	9415.69							
11	9260.75							
12	9092.00							
13	8931.18							
14	8741.00							
15	8543.05							
16	8335.77							
17	8125.38							
18	7899.04							
19	7641.38							
20	7386.42							
21	7131.47							
21.5	6982.93							
22	6854.75							
22.5	6703.44							
23	6568.19							
23.25	6497.71							
23.5	6439.83	2.00						
24	6220.22	5.74						
24.5	6128.75	9.59						
25	5969.15	12.04						
25.5	5824.57	16.19						
26	5656.67	20.26						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.53 IN. HG t_a = 13.2 °C
GROSS WEIGHT 28300 WIND 12 MPH AT 60 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	4303.13		30.5	564.32		46.5	3492.67	36.09
1	4292.95		31	456.53		47	3618.59	38.43
2	4285.00		31.5	332.17		47.5	3745.03	41.19
3	4263.25		32	218.68		48	3889.61	43.93
4	4233.69		32.5	109.16		48.5	4016.05	45.95
5	4194.31		33	24.87		49	4162.18	47.39
6	4146.12		33.5	138.88		49.5	4291.21	49.62
7	4087.56		34	271.02		50	4421.28	52.37
8	4026.41		34.5	389.17		50.5	4567.91	54.50
9	3952.31		35	508.35		51	4698.00	56.96
10	3871.99		35.5	641.53		51.5	4859.69	60.15
11	3780.29		35.75	701.12		52	4990.37	62.59
12	3686.99		36	761.25	.01	52.5	5116.19	65.28
13	3581.26		36.5	896.49	.43	53	5266.98	68.44
14	3466.24		37	1017.24	1.24	53.5	5403.37	71.50
15	3347.91		37.5	1139.00	1.75	54	5552.71	75.41
16	3230.61		38	1273.24	3.02	54.5	5698.13	78.50
17	3082.77		38.5	1395.51	4.16	55	5835.49	81.26
18	2932.68		39	1533.87	5.59	55.5	5992.46	85.98
19	2784.29		39.5	1655.13	7.22	56	6136.52	88.66
20	2624.68		40	1777.94	8.57	56.5	6295.09	92.38
21	2467.67		40.5	1915.79	10.35	57	6432.08	96.59
22	2292.52		41	2035.60	11.78			
23	2110.63		41.5	2179.03	13.30			
24	1924.59		42	2301.33	15.02			
25	1741.67		42.5	2425.69	17.28			
26	1541.65		43	2567.68	19.73			
27	1333.85		43.5	2693.60	20.90			
28	1120.87		44	2835.52	23.48			
28.5	1018.26		44.5	2958.40	25.54			
29	900.63		45	3089.33	27.61			
29.5	794.92		45.5	3226.31	30.65			
30	655.67		46	3374.52	33.19			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM P_a = 27.48 IN. HG t_a = 8.25 °C
GROSS WEIGHT 28300 WIND 12 MPH AT 96 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	4281.46		24.5	868.07	12.18			
1	4271.62		25	690.96	15.76			
2	4262.29		25.5	609.64	18.74			
3	4242.61		26	469.27	22.04			
4	4208.94		26.5	318.03	26.02			
5	4165.95		27	186.47	31.27			
6	4108.98		27.5	44.54	36.33			
7	4041.12		28	103.07	40.67			
8	3949.45		28.5	258.98	46.54			
9	3854.14		29	406.60	52.03			
10	3751.07		29.5	558.36	57.55			
11	3624.17		30	709.61	63.65			
12	3488.98		30.5	863.96	70.78			
13	3342.91		31	1032.29	78.81			
14	3183.38		31.5	1195.45	86.59			
15	3012.97							
16	2835.83							
17	2640.56							
17.5	2532.49							
18	2432.52							
18.5	2332.37							
19	2213.21							
19.5	2106.54							
20	1990.52							
20.5	1880.19							
21	1759.51							
21.25	1692.69							
21.5	1634.16	1.40						
23	1571.93	3.48						
22.5	1396.35	5.94						
23	1264.34	5.62						
23.5	1123.97	8.29						
24	1000.18	10.16						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: CHECK CLIMBS (MAXIMUM POWER)
W_{ES} 28504 LBS.

TIME Min:Sec	IAS Knots	ALTITUDE Feet	Pt7 In.HG	N1 RPM	N2 RPM	F.A.T. DEG. C	Tt7 DEG. C	(TOTAL) F.U.Eng Lbs.	F.U.A/B Lbs
8:23:13	405.5	6875	67.18	5826	9488	+49	530	1106.95	
:23:21	397.5	8265	64.48	5845	9488	47	540	1172.45	
:23:29	394.5	9595	62.68	5855	9488	44	540	1237.95	
:23:37	396.0	10750	60.50	5865	9488	43	550	1302.45	
:23:45	396.0	12040	57.25	5875	9488	43	550	1362.10	
:23:53	392.0	13425	56.90	5895	9488	40	550	1421.35	
:24:01	384.0	15085	54.90	5913	9488	37	550	1473.25	
:24:09	374.0	16300	52.22	5933	9488	35	570	1532.20	
:24:17	368.0	17675	49.92	5943	9488	30	590	1595.10	
:24:25	361.5	19020	48.12	5967	9475	27	600	1630.95	
:24:31	354.5	20090	46.58	5991	9448	23	600	1670.25	
:24:37	349.5	21125	45.34	5991	9438	20	600	1703.0	
:24:43	343.0	22115	43.80	5991	9438	18	600	1735.75	
:24:48	337.5	23145	42.74	6015	9438	15	600	1775.05	
:24:54	332.0	24130	41.40	6020	9428	11	600	1801.25	
:25:00	327.0	25070	40.24	6015	9397	10	600	1834.0	
:25:06	320.5	25975	39.02	6035	9387	6	600	1866.75	
:25:14	316.0	27120	37.56	6035	9387	4	600	1906.05	
:25:22	312.0	28165	36.14	6040	9378	1	600	1945.35	
:25:30	307	29185	35.14	6040	9378	-3	600	1984.65	
:25:36	303.5	29965	34.16	6040	9368	-5	600	2010.85	
:25:46	297.5	31200	32.69	6045	9354	-7	600	2050.15	
:25:53	296.5	32175	31.88	6045	9349	-8	600	2089.45	
:26:02	286.5	33240	29.86	6055	9345	-10	600	2122.20	
:26:07	280.0	34010	29.39	6060	9335	-12	600	2141.85	
:26:15	270.0	35080	28.09	6064	9312	-15	600	2174.60	
:26:23	265.5	35950	27.08	6064	9302	-17	600	2207.35	
:26:33	263.5	36835	26.24	6068	9292	-20	600	2240.10	
:26:43	262.0	37745	25.42	6068	9292	-21	600	2279.40	
:26:52	252.0	38965	24.00	6074	9283	-21	600	2312.15	
:27:02	248.0	39825	23.22	6084	9279	-23	600	2344.90	
:27:12	244.0	40815	22.31	6090	9264	-25	600	2377.65	
:27:22	236.0	41790	21.35	6100	9245	-29	600	2410.40	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: CHECK CLIMBS (MAXIMUM POWER)
W_{ES} 28504 LBS.

TIME MIN: SEC.	IAS Knots	ALTITUDE Feet	P_{t7} In. HG	N1 RPM	N_2 RPM	F.A.T. DEG. C	T_{t7} DEG. C	F.U. Eng (TOTAL)	F.U.A/B LBS.
8:27:34	226.0	42705	20.34	6110	9245	-30	610	2493.15	
:27:49	222.5	43665	19.43	6110	9221	-31	620	2489.0	
:28:07	214.5	44620	18.43	6120	9216	-33	620	2541.40	
:28:35	214.0	45600	17.95	6120	9199	-35	630	2613.95	
:28:56	206.0	46550	17.10	6115	9179	-37	635	2622.40	
:29:22	198.0	47555	16.15	6115	9165	-39	635	2724.15	
:29:51	196.5	47905	15.95	6105	9150	-39	635	2803.40	
:30:18	196.5	48050	15.85	6109	9146	-39	640	2848.90	
:30:49	198.0	48200	15.75	6109	9151	-39	640	2927.95	
:31:08	201.0	48275	15.75	6100	9142	-39	640	2986.90	
:31:24	214.5	42910	15.55	6065	9142	-38	635	3026.10	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: CHECK CLIMBS (MILITARY POWER)
WEI 28504 LBS.

TIME Min:Sec	IAS Knots	ALTITUDE Feet	Pt7 In. HG	N1 RPM	N2 RPM	F.A.T. Deg. C	Tt7 DEG. C	F.U.Eng. Lbs.
29:43	304.1	7740	59.78	5923	9488	+32	540	1146.25
29:46	300.2	7990	54.38	6933	9488	32	540	1152.80
30:00	294.6	8960	56.22	5944	9478	28	575	1179.0
30:25	305.1	9930	55.27	5944	9478	26	595	1224.85
30:47	309	11000	54.39	5944	9478	27	695	1270.70
31:04	306.1	12005	53.17	5944	9478	25	600	1303.95
31:21	302.2	12980	51.52	5944	9478	23	600	1329.65
31:47	301.7	13980	50.33	5944	9478	21	600	1368.95
31:04	300.1	15005	49.12	5966	9468	20	600	1408.25
32:24	298.2	16015	48.89	5966	9468	17	600	1441.0
32:43	295.6	17005	46.55	5966	9462	17	600	1467.20
33:03	293.5	18040	45.19	5966	9436	14	600	1506.50
33:26	292.1	19020	44.05	5982	9436	12	600	1539.25
33:49	290.5	20000	42.93	5991	9426	9	600	1572.10
34:06	288	21020	41.72	5956	9396	7	600	1611.30
34:33	284	22025	40.45	5986	9286	5	600	1649.05
34:57	284	23025	39.47	5986	9386	2	600	1676.80
35:24	279.5	24035	38.23	5991	9386	0	600	1716.10
35:49	279	25035	37.24	5991	9362	-5	600	1748.75
36:14	276.5	26025	36.23	6000	9392	-8	600	1791.60
36:42	277.5	27010	35.15	6000	9352	-8	600	1825.90
37:13	266.5	27985	33.94	6000	9342	-11	600	1860.30
37:48	263.5	29005	32.74	5992	9312	-14	600	1906.05
[illegible]	261.5	29980	31.60	5992	9292	-16	600	1945.35
38:56	261	30700	31.06	5992	9292	-18	600	1959.65
39:18	260	31450	30.35	5992	9292	-19	600	2017.40
40:00	257.3	32255	29.62	5992	9284	-21	600	2056.70
40:27	252.4	33100	28.72	5992	9254	-22	600	2089.95
41:04	249.7	33800	28.13	5992	9244	-24	600	2128.75
41:44	244.6	34830	27.11	5992	9223	-26	600	2168.05
42:24	240.5	35520	26.18	5992	9200	-28	600	2207.35
43:02	238	36125	25.54	5992	9200	-28	600	2246.65
43:42	240	36460	25.39	5992	9200	-28	600	2285.15
44:06	239.5	36775	25.08	5992	9100	-28	600	2305.60

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TURN CAPABILITIES
(~~DIRECTIONAL HEADING INSTRUMENT INOPERATIVE~~)

IAS Knots	ALTITUDE Feet	Nz "G"	WEIGHT Lbs.	Pt7 In. HG	F.A.T. °C	FLIGHT NO.
290.5	36185	1.9	249837	27.67		3
291.5	36090	1.9		27.57		
293.5	36055	1.5		27.47		
291.5	36300	1.5		27.15		
284.5	36250	1.7		27.15		
284.5	36055	1.8		27.40		
280.5	36240	1.4		27.15		
285.5	36155	1.7		27.25		
285	36010	1.7		27.50		
255.2	36300	0.90		25.90		
264.5	36320	1.5		26.25		
259.5	36580	1.6		25.90		
255.7	36940	1.7		25.47		
256.3	36755	1.8		25.52		
254.2	36480	1.6		25.62		
251.4	36475	1.7		25.57		
251.4	36475	1.7		25.52		
255.3	36135	1.9		25.95		
266.5	35390	1.4		27.10		
270	35595	1.6		27.00		
274	35520	1.8		27.10		
270.5	35920	1.7		27.00		
266.5	36200	1.7		26.55		
266.5	36185	1.6		26.50		
265.5	36040	1.9		26.50		
258	36420	1.5		25.90		
261	36200	1.6		25.72		
268	36555	1.6		26.05		
266.5	36490	1.7		26.30		
262.5	37720	1.5		25.95		
218	37905	1.5		23.07		
214	38005	1.4		22.90		
212.5	39140	1.4	↓	22.69		↓

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: TURN CAPABILITIES
~~(DIRECTIONAL HEADING INSTRUMENT INOPERATIVE~~

IAS Knots	ALTITUDE Feet	Nz "G"	WEIGHT Lbs.	Pt7 In. HG	F.A.T. °C	FLIGHT NO.
212	38200	1.35	249837	22.58	-33	3
214.5	38200	1.4		22.63	-33	
219	38155	1.4		22.90	-33	
321	30095	1.4		32.12	-9	
320.5	32000	1.4		31.92	-7	
325	31035	1.3		32.17	-7	
326.5	29975	1.4		32.47	-5	
321.5	33310	1.2		31.97	-5	
319	32225	1.3		31.37	-7	
216	38625	1.1	↓	22.86	-5	↓

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: LEVEL FLIGHT (NON-AFTERBURNING)
WES 28355 LBS WS 27597 LBS WD

IAS Knots	ALTITUDE Feet	F.A.T. °C	Pt6 In. Hg	N2 RPM	N1 RPM	δB Avg. Deg.	Wf Engine Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
437.4	9325	43.0	66.00	9490	5892	0.75	7566	CLOSED	27258
430.5	9520	42.0	60.60	9365	5720	0.75	6670		26905
388.5	10175	35.0	53.81	9200	5543	1.0	5922		26292
367.5	11040	36.0	49.82	9066	5493	0.50	5238		25691
345.5	11490	28.0	44.92	8892	5313	0.65	4392		25397
321.0	12000	26.0	41.21	8756	5200	0.50	3810		25214
301.2	12185	23.0	39.30	8698	5136	0.55	3504		25051
279.0	12585	20.0	36.95	8564	5043	0.50	3120		24842
258.3	12710	19.0	35.63	8548	5019	0.50	3186		24685
234.0	13085	14.0	31.26	8293	4746	0.50	2382		24376
212.0	13180	10.0	31.80	8312	4816	0.35	2718		24261
181.0	13605	9.0	31.75	8360	4867	0.50		↓	24071

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS

TEST: LEVEL FLIGHT (NON-AFTERBURNING)
WES 28355 LBS WS 26552 LBS WD

IAS Knots	ALTITUDE feet	F.A.T. °C	Pt7 In. Hg	N2 RPM	N1 RPM	√E Avg. Deg.	Wf Engine Lbs/Hr	BLEED-VALVE	WEIGHT Lbs.
358	23590	17	42.86	9408	5971	0.75	5557	CLOSED	26540
358	23630	17	42.81	9408	5971	0.75	5277		26955
358	23630	17	42.81	9408	5971	0.75	4681		26442
321.5	24075	14	34.52	8928	5538	0.65	3720		26043
296.2	25065	3	31.23	8779	5418	0.75	3345		25730
282.0	25645	-3	30.00	8737	5418	0.50	3094		25528
232	25845	-7	26.66	8559	5239	0.55	2380		25208
242.7	25965	-9	26.71	8559	5250	0.55	2971		25120
229.5	26470	-11	24.40	8412	5126	0.50	2360		24953
205.7	26495	-14	24.25	8412	5126	0.65	2472		24679
356.5	23600	+17	42.6	9406	5971	0.75	4600	↓	26611

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS

TEST: LEVEL FLIGHT (NON-AFTERBURNING)
WES 28355 LBS WS 26076 LBS WD

IAS Knots	ALTITUDE Feet	F.A.T. °C	Pt7 In. Hg	N2 RPM	N1 RPM	√E Avg Deg.	Wf Engine Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
292.0	34180	-17	29.92	9245	5895	0.85	3927	CLOSED	25723
[illegible]	34450	-19	27.42	9050	5746	0.70	3221		25478
266.5	34700	-21	25.70	8935	5695	0.50	3199.9		25201
248.5	35320	-24	24.05	8845	5670	0.60	2636.6		25025
229.5	35610	-30	22.36	8705	5560	0.75	2547.7		25861
203.5	35270	-35	20.59	8560	5445	0.75	2340		24666
184.0	35675	-38	20.39	8595	5505	0.75	2672.9	↓	[illegible]

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS

TEST: LEVEL FLIGHT (NON-AFTERBURNING) MILITARY
WES 23300 WS WD

IAS Knots	ALTITUDE Feet	F.A.T. °C	Pt7 Ing Hg.	N2 RPM	N1 RPM	√E Avg. Deg.	Wf Engine Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
316	42080	-39	16.90	9090	6300	5.20			25255
354	25060	+10	41.10	9400	6050				
276.5	35145	-20	22.70	9220	6300	5.15			
241	39940	-25	18.85	9200	6330	5.10			↓

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS

TEST: LEVEL FLIGHT (AFTERBURNING) MAXIMUM
WES 23300 WS WD

IAS Knots	ALTITUDE Feet	F.A.T. °C	Pt7 In. Hg.	N2 RPM	N1 RPM	√E Avg. Deg.	Wf Engine Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
266.5	42150	-25	22.80	9150	5080	5.15			24555
354	25250	+10	43.55	9450	6050				
316.5	35020	-9	21.10	9270	6000	5.125			
293	40050	-14	24.95	9210	6000	5.15			↓

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: THRUST RUN
DATE 7 JUNE 1956 Pa 27.5375 to 23

Fg Lbs.	Pt7 In. Hg	N2 RPM	WF Total Lbs/Hr.	Tt7 DegreesC
440	29.05	5530	904.12	250
13155	63.44	9380		520
8495	60.54	9430	7569.5	580
565	29.58	6020	831.95	255
6590	53.08	9005	5566.7	420
4040	42.93	8455	3272.3	305
2410	36.14	7950	2268.5	260
800	30.46	6500	742.38	230
500	29.36	5680	912.1	240
8270	60.04	9415	7613.9	512
12790	60.89	9455		590
1110	31.38	6955	1192.2	258
2270	35.70	7955	1433.52	250
11650	58.78	9245		490
7620	57.52	9280	6767.48	500
680	29.86	5755	775.76	270
7175	55.38	9155	5992.189	450

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: THRUST RUN
DATE 7 JUNE 1956 Pa 27.5375 to 23

FG Lbs	Pt7 In. Hg	N2 RPM	WF Total Lbs/Hr.	Tt7 DegreesC
12210	59.29	9365	-	550
570	29.57	5965	854.75	273
2530	37.10	8035	11754.0	257

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
Test: Thrust Run
DATE 25 JUNE 1956 Pa 27.505 to 28.86

Fg Lbs.	Pt7 In. Hg	N2 RPM	WF Total Lbs/Hr.	Tt7 DegreesC
440	29.04	5520	803.87	260
12830	61.15	9410	-	490
8300	61.25	9460	7013.34	530
575	29.50	6020	729.87	282
6220	52.20	9010	5110.64	380
3760	41.30	8460	3105.0	310
2250	35.80	7990	2350.8	260
760	30.15	6355	940.32	240
490	29.20	5650	759.31	250
7800	58.40	9370	7010.0	480
11630	58.40	9320	—	504
1080	31.55	7015	1198.90	255
1450	35.10	7945	1532.70	255
11310	57.80	9275	—	490
7240	55.55	9275	6370.61	490
710	29.85	6230	646.50	270
6730	54.55	9215	5782.97	455

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: THRUST RUN
DATE: 25 JUNE 1956 Pa 27.505 to 28.86

FG lbs	Pt7 In. Hg	N2 RPM	WF Total lbs/Hr	Tt7 DegreesC
12520	59.25	9490	16244.0	570
600	29.45	5945	712.30	272

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: AIRSPEED CALIBRATION - PACER
(PACER AIRCRAFT F-100A, USAF No. 53-1565

PACER V_c	PACER H_c	PACER M	IAS Knots	ALTITUDE Feet
423.1	178756		410.5	16985
408.9	17734		397.5	16825
359.1	17864		349.5	17170
305.2	176528		298	17185
257.3	176883		251	17360
204.9	17700		200.5	17505
313.4	32345		304	31425
299.7	320838		290.5	31265
279.3	320314		270.5	31360
258.8	318371		250.5	31360
236.7	31858.6		230	31400
215.7	348722		206.5	31525

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: LANDING
Pa = 27.47 ta = 38.50 GROSS WEIGHT 23000 LBS.
WIND 9 MPH AT 48 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	5597.13	93.53	16.5	2235.51		45	283.46	
.5	5664.99	62.96	17	2149.49		46	284.49	
1	5736.03	33.17	17.25	2096.54				
1.5	5592.41	62.12	17.5	2035.74				
2	5461.63	51.98	18	1979.34				
2.5	5335.39	43.94	18.5	1894.54				
3	5193.40	34.30	19	1804.37				
3.5	5066.01	27.07	19.5	1729.73				
4	4945.74		20	1650.99				
4.5	4806.31		21	1496.56				
5	4686.09		22	1349.52				
5.5	4558.39		23	1204.92				
6	4434.24		24	1078.37				
6.5	4319.20		25	949.83				
7	4262.20		26	836.15				
7.5	4205.19		27	719.76				
8	4080.31		28	613.05				
8.5	3969.93		29	511.46				
9	3847.64		30	423.37				
9.5	3633.10		31	333.72				
10	3514.95		32	250.29				
10.5	3412.35		33	172.04				
11	3328.73		34	104.16				
11.5	3207.66		35	39.32				
12	3095.21		36	20.73				
12.5	2997.27		37	-72.07				
13	2890.52		38	-119.19				
13.5	2795.17		39	-159.61				
14	2703.45		40	-194.33				
14.5	2596.25		4	-222.3				
15	2509.64		42	-241.18				
15.5	2418.29		43	-265.84				
16	2384.60		44	-277.24				

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: LANDING
Pa = 27.53 ta = 36.0 GROSS WEIGHT 23000 LBS
WIND 14.97 MPH AT 81.6 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	5943.07	74.30	36	406.08				
5	5773.70	66.28	38	357.91				
1	5640.80	55.33	40	335.64				
1.5	5514.27	46.28						
2	5394.71	37.29						
2.5	5238.65	29.87						
3	5110.19	23.52						
3.5	4984.85	18.38						
4	4861.57	15.39						
4.5	4724.31	10.43						
5	4608.29	7.33						
5.5	4487.61	4.26						
6	4367.44	.84						
6.2	4335.33							
6.5	4240.02							
7	4146.27							
8	3877.65							
9	3676.48							
10	3455.31							
11	3255.90							
12	3065.29							
14	2690.80							
15	2393.49							
16	2352.57							
18	2036.42							
20	1753.29							
22	1491.72							
24	1258.12							
26	1051.98							
28	871.73							
30	718.41							
32	586.33							
34	482.74							

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: LANDING
Pa = 27.445 ta = 23.0 GROSS WEIGHT 23000 LBS
WIND 13.40 MPH AT 38.4 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	7650.63	105.01	16.5	2805.55		36	$\overline{925.49}$	
.5	6896.72	91.81	17	2803.78		37	$\overline{963.33}$	
1	6761.47	79.93	17.5	2702.93		38	$\overline{1113.61}$	
1.5	6611.20	66.87	18	2588.93		39	$\overline{1251.45}$	
2	6476.98	55.39	18.5	2490.47		40	$\overline{1376.34}$	
2.5	6327.74	43.40	19	2390.09		41	$\overline{1506.93}$	
3	6197.15	34.13	19.5	2280.60		42	$\overline{1630.36}$	
3.5	6063.98	25.60	20	2171.79		43	$\overline{1751.58}$	
4	5912.33	16.96	20.5	2273.94		44	$\overline{1895.06}$	
4.5	5788.29	10.59	21	1965.01		45	$\overline{2006.47}$	
5	5645.79	5.61	21.6	1869.67				
5.5	5516.83	2.29	22	1773.80		47	$\overline{2212.39}$	
6	5379.95		22.5	1618.86		48	$\overline{2310.14}$	
6.5	5258.18		23	1472.21		49	$\overline{2402.89}$	
7	5135.36		23.5	1368.05		50	2536.59	
7.5	5001.67		24	1268.55				
8	4883.00		24.5	1178.91		52	$\overline{2689.98}$	
8.5	4748.78		25	1088.22		53	$\overline{2760.45}$	
9	4632.19	(Nwd)	25.5	989.26		54	$\overline{2826.78}$	
9.5	4502.64		26	902.70		55	$\overline{2889.18}$	
10	4386.56		26.5	801.14		56	$\overline{2948.04}$	
10.5	4271.52		27	715.12		57	$\overline{2997.79}$	
11	4145.08		27.5	617.18		58	$\overline{3049.09}$	
11.5	4031.60		28	530.64		59	$\overline{3097.80}$	
12	3927.75		28.5	446.69		60	3142.88	
12.5	3817.40		29	351.34		61	3194.20	
13	3675.54		29.5	268.43		62	$\overline{3234.09}$	
13.5	3569.88		30	186.55		63	$\overline{3268.81}$	
14	3450.18		31	2.07				
14.5	3344.46		32	$\overline{157.01}$				
15	3241.86		33	$\overline{321.28}$				
15.5	3123.19		34	$\overline{473.12}$				
16	3021.11		35	$\overline{629.61}$				

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: LANDING -
Pa = 27.64 ta = 29.5 GROSS WEIGHT 23000 LBS
WIND 4.49 MPH AT 60 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	11492.12	106.42	28	6010.60					
.5	11363.61	95.57	30	5857.73					
1	11213.85	84.96	32	5722.48					
1.5	11080.67	75.61	34	5602.22					
2	10928.12	65.56	36	5527.69					
2.5	10779.81	58.22	38	5498.10					
3	10660.74	51.66	40	5482.56					
3.5	10530.34	45.03	42	5474.26					
4	10386.78	36.56							
4.5	10252.07	30.17							
5	10124.59	24.63							
5.5	9978.41	16.64							
6	9849.95	11.73							
6.5	9725.06	8.39							
7	9579.45	3.87							
7.5	9456.19	.57							
7.6	9414.66								
8	9339.02								
8.5	9209.45								
9	9091.82								
10	8851.37								
11	8626.99								
12	8402.61								
13	8190.67								
14	7983.91								
15	7795.80								
15.25	7740.87								
16	7608.21								
18	7158.95								
20	[illegible]								
22	6654.75								
24	6413.24								
26	6193.[illegible]								

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: LANDINGS -
Pa = 27.50 ta = 35.5 GROSS WEIGHT 23000 LBS
WIND 2.30 MPH AT 85 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	11209.70	95.92	17	6754.22					
.5	11048.54	83.93	18	6581.66					
1	10889.97	73.40	19	6410.65					
1.5	10719.49	62.16	20	6253.64					
2	10565.06	52.74	22	5952.58					
2.5	10415.82	44.77	24	5697.61					
3	10248.94	36.07	26	5474.12					
3.5	10104.90	27.90	28	5281.49					
4	9947.89	22.16	30	5116.71					
4.5	9804.86	17.29	32	4981.97					
5	9669.09	14.29	34	4864.34					
5.5	9512.60	11.16	36	4776.25					
6	9372.87	7.88	38	4702.67					
6.5	9233.81	6.39	40	4649.81					
7	9102.18	5.38	42	4617.16					
7.5	8924.71	4.33	44	4599.03					
8	8832.20	3.83		[illegible]					
8.5	8709.72	3.62							
9	8578.28	3.41							
9.5	8443.55	1.66							
10	8321.77	.63							
10.5	8190.15	−.14							
10.62	8160.10								
11	8073.04								
11.5	7956.44								
12	7827.41								
12.5	7715.48								
13	7591.11								
13.5	7480.22								
14	7373.99								
14.5	7259.98								
15	7157.39								
16	6952.2								

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: LANDING -
Pa = 27.53 ta = 29.0 GROSS WEIGHT 23000 LBS
WIND 13.48 MPH AT 60 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	6102.32	44.25	16.5	2282.15		43	1489.93		
.5	5972.70	34.37	17	2190.95		44	1576.88		
1.5	5700.72	22.36	17.5	2088.35		45	1663.94		
2	5571.17	16.51	18	1999.22		46	1746.32		
2.5	5446.81	11.22	18.5	1902.05		47	1825.62		
3	5313.10	8.43	19	1808.52		49	1969.16		
3.5	5198.27	4.65	19.5	1722.50		50	2036.52		
4	5066.49	1.88	20	1636.49		51	2107.52		
4.5	4931.19	−.08	20.5	1541.65		52	2159.3		
4.87	4842.58		21	1456.66		53	2219.45		
5	4819.60		21.5	1373.23		54	2267.69		
5.5	4697.48		22	1292.39		55	2315.22		
6	4562.93		22.5	1199.11		56	2359.36		
6.5	4454.97		23	1106.36		57	2403.41		
7	4328.52		24	947.79		58	2443.21		
7.5	4216.08		25	781.96		59	2479.03		
8	4104.66		26	621.32		60	2504.46		
8.5	3980.81		27	462.75		61	2531.41		
9	3872.51		28	320.25		62	2554.73		
9.5	3763.69		29	171.01		63	2574.38		
10	3645.46		30	23.84		64	2589.45		
10.5	3532.23		31	118.67		65	2602.42		
11	3430.48		32	248.22		65.5	2605.51		
11.5	3316.93		33	382.95					
12	3210.77		34	511.46					
12.5	3094.17		35	637.90					
13	2994.68		36	757.39					
13.5	2898.81		37	870.06					
14	2784.81		38	982.51					
14.5	2686.87		39	1093.40					
15	2577.01		40	1193.41					
15.5	2482.18		41	1292.05					
16	2387.35		42	1394.98					

TEST DATA CORRECTED FOR INSTRUMENT ERROR
TF-102A, USAF NO. 54-1351
PHASE II FLIGHT TESTS
TEST: LANDINGS -
Pa = 27.15 ta = 32.0 GROSS WEIGHT 23000 LBS
WIND 4.72 MPH AT 80 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	5757.13	71.68	19	1910.75					
1	5426.60	55.82	19.5	1847.05					
2	5261.49	39.87	20	1783.34					
3	4996.24	26.09	20.5	1725.32					
4	4759.02	17.78	21	1667.83					
5	4518.66	12.54	21.5	1607.75					
5.5	4408.36	9.91	22	1544.56					
6	4290.78	8.59	23	1444.07					
6.5	4175.28	5.22	24	1342.03					
7	4072.72	4.37	25	1243.10					
7.5	3947.89	2.52	26	1150.39					
8.05	3931.35	(0)	27	1068.07					
8.5	3736.05		28	989.30					
9	3631.92		29	912.38					
9.5	3528.86		30	852.56					
10	3437.70		31	790.92					
10.5	3324.22		32	732.40					
11	3224.82		33	679.05					
11.5	3125.89		34	634.50					
12	3025.92		35	592.99					
12.5	2934.76		36	549.56					
13	2846.19		37	512.78					
13.5	2756.07		38	481.70					
14	2671.12		39	457.26					
14.5	2580.48		40	440.78					
15	2495.53		41	428.35					
15.5	2420.95		42	421.10					
16	2343.25		43	419.03					
16.5	2266.59								
17	2190.97								
17.5	2124.15								
18	2052.68								
18.5	1976.02								

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The TF-102A is acceptable as a transitional trainer for delta-wing aircraft. The airspeed-altitude performance envelope is significantly inferior to performance characteristics of the F-102A. This performance degradation combined with airframe buffet encountered in the .8 to .9 Mach number region will tend to restrict the utility of the aricraft as a weapon system proficiency trainer.

Contractual guarantees for take-off, maximum power climb performance, and service ceiling are met or exceeded, but the aircraft is deficient with respect to military power climb performance and maximum speed.

In general, handling characteristics are acceptable.

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March 1957
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FUNCTIONAL DEVELOPMENT

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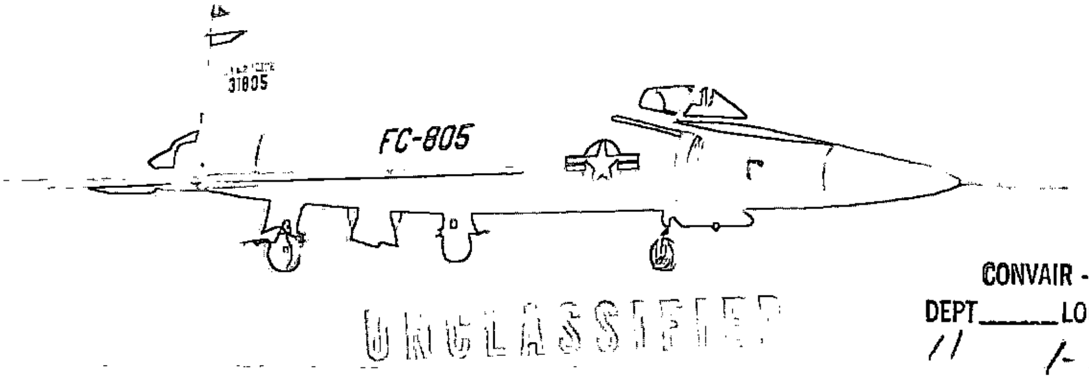

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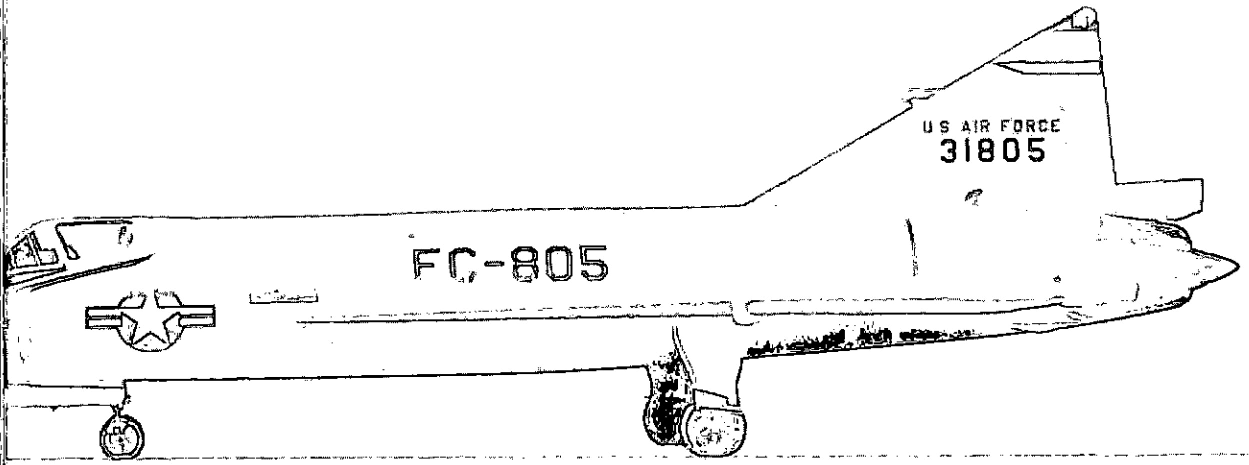

FUNCTIONAL DEVELOPMENT

AFFTC-TR-56-34 March 1957

ASTIA Document No. AD 112421

ROBERT L. STEPHENS, Major, USAF, Project Officer

RICHARD D. COUSINS, Captain, USAF, Project Engineer

ROBERT M. WHITE, Captain, USAF, Project Pilot

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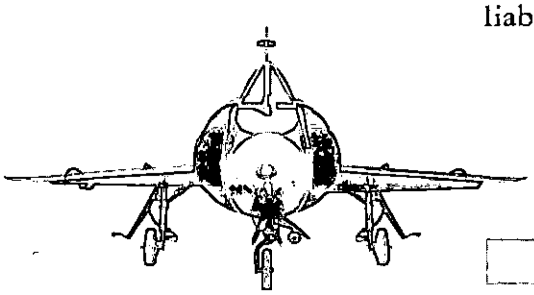
ABSTRACT

THE F-102A AIRCRAFT has the potential to furnish the Air Defense Command with an excellent weapon system. The handling characteristics of the aircraft are exceptional, but demonstrated deficiencies such as engine compressor surge, afterburner blowout and engine over-temperature at high altitude, and low kill probability of the armament system limit the operational capability of the aircraft.

Accessibility for maintenance of the aircraft and its sub-systems is an improvement over the F-86D interceptor aircraft. Major functional deficiencies which limit utilization include an insufficiently refined fire control system, an unsatisfactory canopy operating mechanism and unreliable cabin pressurization and unreliable windscreen de-fogging systems.



This report has been reviewed and approved / **2 February 1957**

HA Hanes
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Colonel, USAF
Director Flight Test

J.S. Holtoner
J. S. HOLTONER
Brigadier General, USAF
Commander

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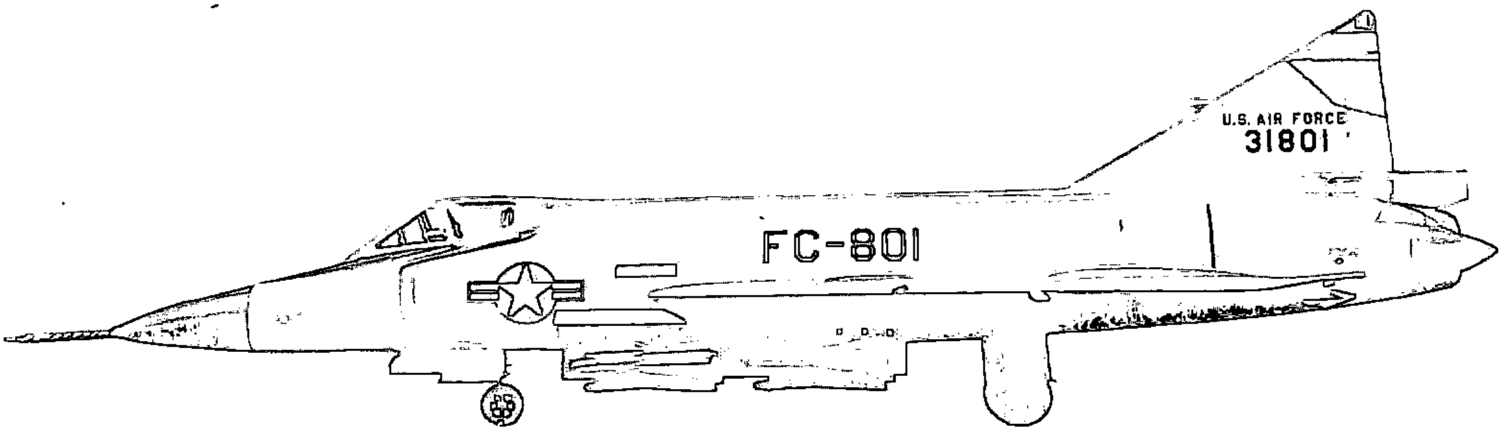
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INTRODUCTION

The F-102 Phase VI Functional Development tests were conducted to evaluate and determine the serviceability of the aircraft and its complete associated equipment from a functional and maintenance viewpoint, and to obtain a qualitative analysis of the operational characteristics of the aircraft.

These tests, directed by Air Research and Development Command Test Directive No. 5593-FI, were accomplished at Edwards AFB between 4 January 1956 and 28 August 1956. Runway overrun barrier engagement tests were performed in conjunction with the Phase VI tests and were accomplished between 16 August and 23 November 1956.

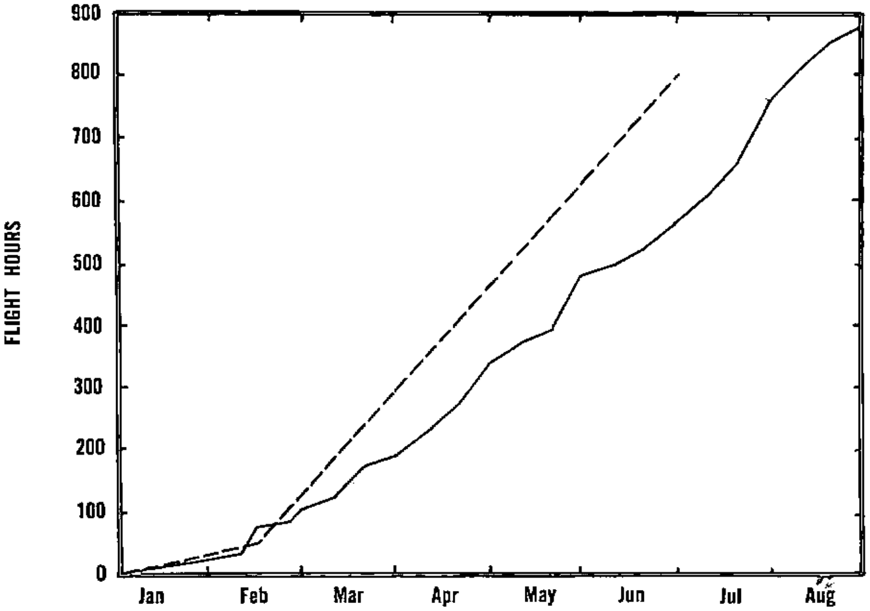
This report includes a resume of the test program, manhour and parts consumption data, utilization rates, functional analysis of the various systems, maintenance problem areas, operational comments by pilots who flew the aircraft, and Unsatisfactory Reports. In some cases solution to problem areas and deficiencies have already been obtained and in other cases action towards solution has been initiated.

Personnel support was drawn largely from the Air Defense Command, the ultimate using command. The Air Training Command provided support on a smaller scale during the first portion of the program. Participation by the Air Defense Command afforded its personnel an opportunity to familiarize themselves with the aircraft, and to become qualified as instructors for their own squadron training program. A card system of inspection, similar to the inspection procedure utilized in the Air Defense Command, was utilized during this test. This card system was greatly refined as a result of the aircraft inspections performed during the test and will be available for use by Air Defense squadrons.

The F-102A aircraft is a single place, supersonic, delta wing, high altitude, all weather interceptor. The airplane with its MG-10 fire control system and varied armament comprises a complete weapons system. The MG-10 system tested consisted of the MG-3 radar only. The automatic flight control system and data link were not installed. The aircraft is powered by a Pratt and Whitney J57-P23 engine which is rated at 10,200 pounds sea level static thrust at Military Power and 16,000 pounds at maximum power with afterburning. Armament consists of 6 GAR (Falcon) missiles carried internally in two missile bays located in the lower center section of the fuselage. Twenty-four 2.75-inch FFAR rockets are carried inside the missile bay doors. Outstanding physical characteristics of this aircraft are its delta wing and vertical fin, pointed radome nose, cheek-mounted air inlet ducts, and waisted (area rule) fuselage.

CHART NO. 1

FLIGHT PROGRESS



FLIGHT SUMMARY

230	Familiarization
170	Profile
235	Radar & Systems
50	Missile Firing
75	FTF
38	Night
18	Formation
10	Pressure Suit
5	Aerobatics
7	Fuel Control
45	Ferry & Air Shows

TEST PROGRAM

The Phase VI test program was conducted with five test aircraft. A total of 865 hours were flown in 701 flights. Dates of arrival and accumulated test time are shown in the charts depicted in Table No. 1 and Chart No. 1.

The program was conducted in five basic phases: (1) aircraft familiarization and pilot transition; (2) systems evaluation and shakedown flights; (3) simulated combat profile missions, (4) air-to-air live missile firing, and (5) runway barrier engagement tests. The fourth phase was conducted at Holloman Air Force Base, New Mexico. A detailed breakdown of the flight time expended and the results of the missile firings at HADC is discussed in a latter portion of this report. A discussion of the runway barrier tests is also located in a latter portion of this report. A breakdown of the total flight time expended on all Phase VI missions flown during the test program is shown in Chart No. 2, "Flight Summary Chart."

The unusually large number of pilot check-outs required during this program somewhat restricted the orderly progress of data collection and mission scheduling. Pilots qualified during the program are as follows:

58	USAF
5	RAF
2	RCAF
5	Contractor
—	
70	Total

Table No. 1 Aircraft Participation

Aircraft Serial No.	Date of Arrival	Total A/C Flt. Time	Total A/C Ph VI Time	Total No. Ph VI Flts.
58-1800	6 Jan 56	218:30	212:45	167
53-1801	4 Jan 56	164:35	154:40	138
53-1803	3 Feb 56	100:50	92:20	78
53-1805	3 Feb 56	253:15	237:25	188
53-1808	9 Feb 56	173:25	167:50	130
	Total	910:35	865:00	701

Of the five aircraft participating in the test program, four contained pre-production MG-10 fire control systems. Only one, S/N 53-1808, contained the production version of the system. The four containing pre-production systems were supported supply-wise by the system contractor, Hughes Aircraft Company, through a special contract written for that purpose. Some equipment was missing from each aircraft and not all systems were operational. Certain restrictions to aircraft maneuvering were imposed throughout the program; however, as the program progressed, some of these restrictions were lifted or made less stringent as a result of contractor modifications and flight tests.

Serious difficulties experienced in the firing of 2.00-inch rockets during other Air Force projects caused curtailment of the firing of rockets during the early portion of the Phase VI program. Before revisions to the rocket system could be accomplished, a decision was made by higher headquarters to change the rocket firing capability of the F-120A aircraft to 2.75-inch rockets. Accordingly, a message from the F-102A Weapons System Project Office restricted the aircraft from firing either the 2.00-inch or the 2.75-inch rockets until such time as a contractor redesign proposal could be approved and retrofit action accomplished for the 2.75-inch rockets. Since no retrofit action occurred in time to be functionally tested during the Phase VI program, such testing will be required at some future date.

MAINTENANCE AND INSPECTION

☐ aircraft availability and maintenance status

The F-102A aircraft were maintained on an 18-hour day, 6-day work week schedule. Disregarding inactive time during which no maintenance or flying was performed and considering in-commission time as being that time the aircraft were being flown or were ready to fly, the average aircraft was out-of-commission 16.9 hours for every hour of in-commission time. A breakdown of the elapsed time (hours) for the accomplishment of various types of maintenance and the manhours expended by flight line and maintenance shop personnel is tabulated in the following table. The average in-commission rate for the test aircraft was 8.2 percent which is computed by dividing the total in-commission time by the total effective available (in-commission plus out-of-commission) time.

Table No. 2

Aircraft Availability and Maintenance Status

Aircraft	Aircraft Serial Numbers					
	53-1800	53-1801	53-1803	53-1805	53-1808	Total
Flight Test Hours	212:45	154:40	92:20	237:25	167:50	865:00
In-Commission Hours	314	232	214	309	231	1300
Out-of-Commission Hours						
Scheduled Maint.	1380	1170	1228	1462	1012	6252
Unscheduled Maint.	1508	1492	1082	1601	957	6640
A.O.C.P	370	163	296	44	636	1510
Modification, E.O., T.O.C.	37	55	57	35	67	251
Total	3295	2880	2663	3142	2672	14,653
In-Commission Rate	8.7%	7.5%	7.5%	8.9%	7.9%	8.2%
A.O.C.P. Rate	10.1%	5.4%	10.0%	1.4%	19.2%	9.4%

⊡ Phase VI manhour data

A detailed breakdown of cumulative direct manhours expended on various systems and on scheduled inspections is as follows:

Table No. 3

Aircraft Serial Numbers

	53-1800	-1801	-1803	-1805	-1809	Total
Scheduled						
Servicing	488	327	291	391	287	1,784
Pre-flight	1,085	1,345	902	927	756	5,015
Post-flight	483	470	453	766	475	2,647
Periodic	3,045	2,163	2,850	3,418	2,199	13,675
Total	5,101	4,305	4,496	5,502	3,717	23.121
Unscheduled						
Airframe	1,037	661	911	683	876	4,168
Utilities	487	94	223	164	123	1,091
Power Plant	1,779	1,656	788	2,571	1,317	8,111
Landing Gear	470	185	99	314	369	1,437
Instrument	132	99	181	109	47	568
Radio & Fire Control	1,052	1,161	1,006	647	422	4,288
Armament	164	375	393	208	125	1,265
Electrical	904	328	642	635	712	3,221
Fuel System	79	184	192	112	155	722
Hydraulic	671	454	229	724	820	2,898
Total	6,775	5,197	4,664	6,167	4,966	27,769
Total Maintenance Man-hours	11,876	9,502	9,160	11,669	8,683	50,890

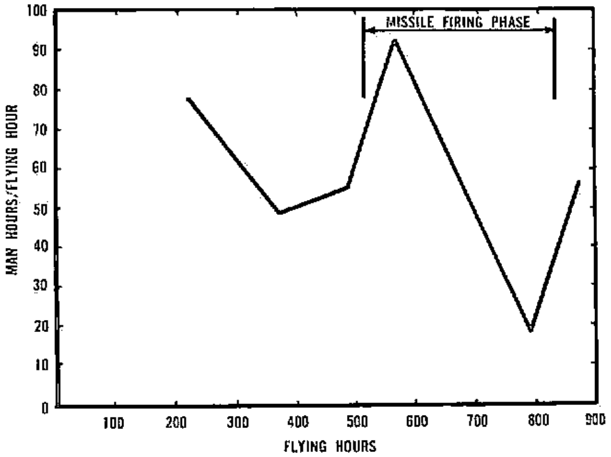
A total of 50,890 direct man-hours were expended during the program. With two 9-hour shifts conducted 6 days a week, the average total maintenance man-hours per flying hour ratio amounted to 58.6. The operating procedures of Air Defense Command facilities should have a marked influence on reducing this ratio. The average turnaround time, excluding servicing of the armament was 30 minutes.

The peak which exists in the manhour curve at approximately 560 hours flying time on the following chart is attributed to the large amount of maintenance required to place the armament systems and pre-production fire control systems on two aircraft in operational condition in order to perform the guided missile firing function at HADC satisfactorily.

CHART NO. 3

MAINTENANCE

MANHOURS FLYING HOUR



During the entire test program the MG-3 radar of the MG-10 fire control system was maintained in an in-commission status as much as practicable. The following tabulation shows the manhours of maintenance performed on the fire control system alone. During the test it was found that 11.6 manhours of maintenance time were required for each hour of MG-3 radar airborne operation. Approximately 70 percent of the ground operating time during the firing program at HADC was occupied with missile range gate harmonization. Without harmonization, it is estimated that the maintenance manhour requirement per airborne operating hour would drop considerably for an intensive firing program.

Table No. 4

Aircraft Number	Sched Maint Mnhrs	Unsched Maint Mnhrs	Ground Operating Time — Hrs	Airborne Operating Time — Hrs	Ratio of Maintenance Mnhrs per Airborne Operating Hour
53-1800	512.3	562.3	193.1	115.4	9.3
53-1801*	776.7	969.5	367.3	102.4	17.0
53-1803*	503.2	856.2	234.1	52.8	25.8
53-1805	331.2	420.0	184.9	119.3	6.3
53-1808	260.1	306.0	137.0	82.4	6.9
Total	2383.5	3114.0	1116.4	472.3	11.6

* These two aircraft were used for guided missile firing at Holloman Air Development Center.

☐ recurring maintenance

The following table lists a number of recurring items for which the aircraft were out-of-commission. Included with each item is the optimum number of personnel required to perform the work and the average elapsed time found necessary to perform the work.

Table No. 5

Type of Work	No. of Men	No. of Elapsed Hours
Pre-flight Inspection	3	1:20
Post-flight Inspection	3	1:10
Safety of Flight Inspection	2	0:30
Periodic Inspection	5	82.00
Engine Installation	6	10:00
Engine Removal	4	6:20
Engine Ground Run	3	0:45
Replace Main Wheel	2	1:00
Replace Nose Wheel	2	0:30
Remove, Inspect and Replace Brakes	2	1:45
Landing Gear Operation Check	3	1:00
Remove and Replace Elevon Actuator	2	1:45
Check Fuel, Pneumatic and Hydraulic Filter Screens	2	1:00
Service Oxygen System	1	0:15
Service Pneumatic System	1	0:30
Refuel	1	0:15
Fire Control System Periodic	3	24:00
Radar Pre-flight	2	0:30
Radar Post-flight, Radome Removal, Radome Installation	2	0:30
Missile Check-out (6 Missiles)	4	1:25
Range Gate Harmonization (6 Missiles)	3	2:00*

* With improved systems in ADC aircraft this figure will decrease considerably.

☐ maintenance personnel requirements

Every effort was made to duplicate the maintenance and inspection methods that will be utilized by the Air Defense Command. In this respect considerable assistance was obtained from the TDY support personnel. These personnel were rotated in three groups with a slight overlap for orientation and training purposes.

Personnel as organized in the following table will be capable of performing organizational maintenance while operating on a two-shift basis with 28 aircraft (25 F-102A and 3 TF-102A), averaging 25 flying hours per month per aircraft. This table represents suggested modifications from the published Organizational Table (O/T 1135, dated 30 April 1956) for one F-102A Squadron. Major differences indicated from Phase VI testing are that the Armament and Ground Powered Equipment fields will require personnel augmention.

TABLE No. 6

FUNCTION CODE-POSITION AND TITLE	AFSC	GRADE M/S	T/S	S/S	A/1	A/2	Total No.	Net Change
80010 Aircraft Maintenance								
APR A/C Jet 1 Engine Mechanic	43131C					(12)	12	— 3
A/C Jet Engine Mechanic	43151C			12	(10)		22	—12
A/C Maint. Jet 1 Eng Tech	43171C	1					1	
Crew Chief	43171C		28				28	
Flight Chief	43171C	(6)					6	— 2
Line Chief	43171C	1	1				2	
A/C Inspector	43171C	*4					4	+ 3
APR Clerk	70230						1	
Clerk	70250			1	2	1	3	
80110 A/C Engine Maintenance								
APR Jet Engine Mech	43230						4	— 1
Jet Engine Mech	43250			5	7	(4)	12	
Jet Engine Tech	43270	(1)	(2)				3	— 2
80210 Airframe Maintenance								
APR Airframe Repairman	53430					(3)	3	— 2
Metals Pro Spec	53250			1			1	
Airframe Repairman	53450			1	*2		3	+ 1
Airframe Repair Tech	53470		1				1	
80310 A/C Elec System Maintenance								
APR A/C Elec Repairman	42330					(2)	2	— 1
A/C Elec Repairman	42350			2	4		6	
A/C Elec Repairman Tech	42370		1				1	
80410 A/C Hydraulic Maintenance								
APR A/C Hyd. Repairman	42132					2	2	
A/C Hyd. Repairman	42152			1	2		3	
A/C Hyd. Maint Tech	42172		1				1	
80510 A/C Inst. Maintenance								
APR Inst Repairman	42230					(2)	2	— 1
Inst Repairman	42250			2	4		6	
Mech ACC/Eqp Repairman	42251			3			3	
Inst Repairman Tech	42270		1				1	
82010 Armament Sys Maintenance								
APR Gdnce Sys GAR-1 Mech	31130E					*5	5	+ 4
Gdnce Sys GAR-1 Mech	31150E			2	3		5	
Gdnce Sys GAR-1 Tech	31170E	*2	2				4	+ 1
APR Fire Control MG-10 Mech	32230H					2	2	
MG-10 Comp Con Mech	32250H			(2)	(3)		5	— 2
MG-10 Comp Con Tech	32270H	1	2				3	
APR MG-10 Radar Mech	32230G					*4	4	+ 2
82010 Armament Sys Maintenance								
MG-10 Radar Mech	32250G			*5	*20		25	+18
MG-10 Radar Tech	32270G	*1	3				4	+ 1
APR MG-10 Data Flow Mech	32230FT					8	8	
MG-10 Data Flow Mech	32250FT			12	17		29	
MG-10 Data Flow Tech	32270FT	6	8				19	
APR Munitions Spec	46130					2	2	
Munitions Spec	46150			*1	*3		4	+ 2
APR Weapons Mech	46230					13	13	
Weapons Mech	46250			(10)	(13)		23	— 4
Weapons Main Supervisor	46270	(2)	(3)				5	— 6
84610 ABN RAD Maintenance								
APR A/C RAD COMM EQP Repairman	30130A					1	1	
A/C RAD COMM EQP Repairman	30150A			*3	*4		7	+ 3
APR A/C RAD Nav EQP Repairman	30130C					1	1	
A/C RAD Nav EQP Repairman	30150C			*2	*3		5	+ 1
A/C RAD Maint Tech	30170	1	*4				5	+ 2
86410 Aero Photo Maintenance								
APR Photo Repairman	40230					*2	2	+ 2
Photo Repairman	40250				2		2	
92410 Gr Pwd EQP Maintenance								
APR Gr Pwd/Spt EQP Repairman	47230				*3	*5	8	+ 6
Gr Pwd/Spt EQP Repairman	47250						11	+ 5
Gr Pwd/Spt EQP Tech	47270	*1	*2	*5	*6		3	+ 2
98010 Clo EQP Maintenance								
APR Parachute Rigger	58130			1	1	*2	3	+ 2
Parachute Rigger	58150		1				2	
TOTAL		27	61	71	109	72	340	+17

Note: * Indicates where additions have been made.
( ) Indicates where deletions have been made.

☐ supply support

Supply support for the Phase VI test was provided by the F-102 Prime Depot, the San Antonio Air Materiel Area. The support was accomplished through a new plan of weapon system supply support. This concept, titled "Delta Cover," to identify the scheme with the F-102 weapons system, consisted essentially of a single control point at the AMA through which all requisitions for F-102 parts were channeled. The AMC had originally intended that this system be utilized for the test phase of each new weapon system. The plan has been expanded to include using command support as well as test phase support for each of the new weapon systems. "Delta Cover" was concerned with the F-102 weapon system alone.

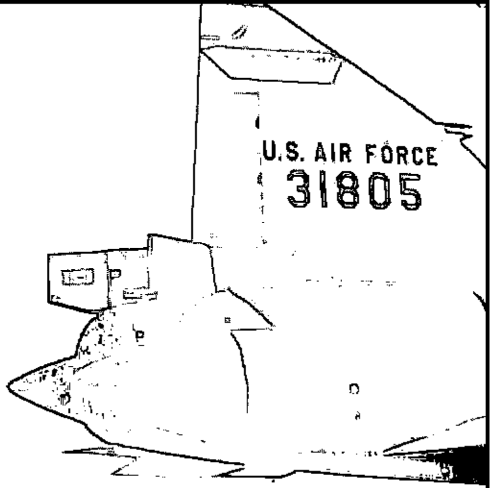
This system of support could be an acceptible method of controlling the tactical logistic support of a weapon system which has progressed beyond the testing and development stages and is in use in a Combat Ready status. However, this concept is quite unsatisfactory insofar as timely support of USAF phase testing is concerned.

In the case of the F-102A Phase VI test, the delays built into the embryo system became intolerable, particularly the procedural delays experienced on AOCP and EODP requisitions. In some instances, EODP requisitions required several months to be filled; in other instances, EODP requisitions were never filled. Because the concept was newly devised, early deficiencies in the "Delta Cover" support can be accounted for; however, after a period of several months, action was expected to become much more expeditious. That this did not occur is best illustrated by the continuous high Phase VI AOCP rate experienced during this test. Because of untimely, unnecessary, and intolerable delays in obtaining Amended Shipping Instruction Numbers from "Delta Cover," the contractor was not able to furnish parts which were immediately available at his installation. Decisions by the Depot to locally manufacture items to fill requisitions also induced delays into the program. In many instances, these parts were available at the contractor's installation, but no attempts were made to furnish Amended Shipping Instructions to the contractor, directing shipment of the required parts to this Center. One example of this nature consisted of a decision to fabricate an aft radar access door to replace one which had been lost in flight. Due to problems encountered in fabricating this assembly, the project was abandoned and a door was finally obtained from the contractor. A period of 17 days elapsed between the time the door was ordered and the time the door was received. A second example consisted of a "Delta Cover" decision to locally manufacture an armament bay fitting instead of requesting shipment from the contractor. The fitting involved reduced the missile firing capability of the two Phase VI aircraft at Holloman Air Force Base, New Mexico, by 25% at a time when range schedules for target drones were at a premium.

Supply support of ground support equipment was particularly inadequate. Of the items listed in Part IV (contractor furnished equipment) of the F-102A Phase VI Test Support Table, only 55.4 percent had been received at the termination of the test program. Supply support of the ground environment required to support the new weapon systems coming into the Air Force inventory for test and for operational assignments must be immediate and complete.

Experience and past records have proven that it is impossible to predict consumption accurately enough to initiate timely procurement and establish effective levels of supply within the Air Force to meet the early and constant demands of Phase testing. If initial supply support is not tailored to meet the demands imposed by drastically new aircraft production methods, very little progress can be expected. Production-type aircraft are presently undergoing as much development as experimental prototype aircraft have experienced in the past. This, of course, is done to expedite production scheduling. However, as many as 200 aircraft might be produced before an optimum configuration is achieved which will have incorporated all of the improvements generated during concurrent Air Force and Contractor Phase testing.

It is mandatory that the supply support delays associated with not only this program, but all Phase VI programs, be eliminated. Some method of effecting a realistic Air Force supply build-up capable of meeting the demands imposed by aircraft still essentially under development must be devised. Such a method must be designed to preclude in every case possible the unnecessary procurement and stockpiling of many items which are rendered obsolete through normal development by the time they are stockpiled. Contractor support of a weapon system for all accelerated test programs should be considered as a possible means of accomplishing effective and timely sup-



ply support. Consumption data obtained during the testing cycle would be factual and would contain an accurate indication of future tactical requirements.

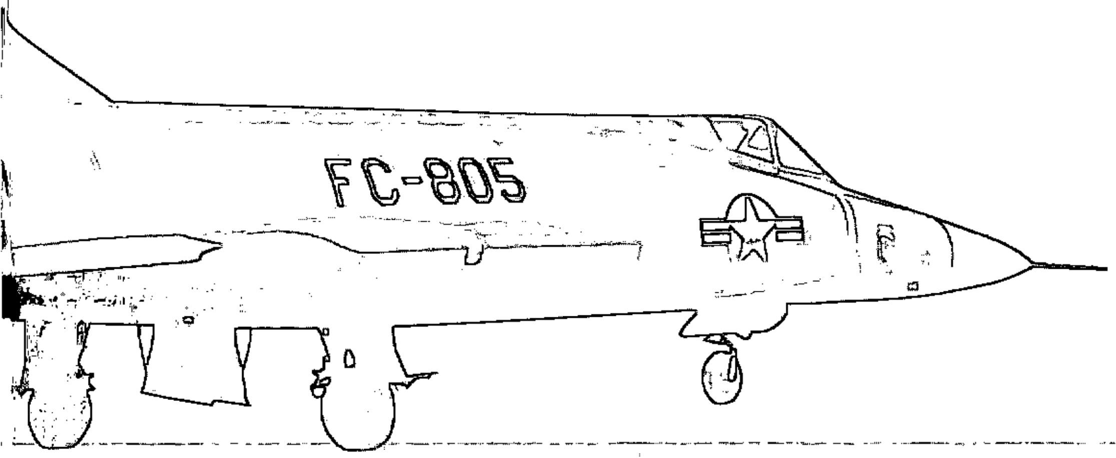
In the case of the F-102A Functional Development program, the "Delta Cover" support concept was not satisfactory. It is illogical to expect a weapon support system to attain full effectivity unless sufficient time is allotted for its build-up. Such a build-up cannot be based upon probabilities of requirements but must be established, where production lead time permits, on a part for part basis. If a part is to be redesigned as a result of in-service development, the time for quantity procurement of the part is after that development. The "Delta Cover" support of this Phase VI program would have been more satisfactory had the availability of parts at the contractor's site been utilized to a greater extent.

☐ handbooks

In general, the handbooks were very satisfactory. There were several instances where items were on the aircraft but not listed in the parts handbook, minor inspection requirements were not accurate in detail, and specifications for some replacement materials were not entirely complete.

Two major deficiencies were encountered in the handbooks. In Section III of the Flight Handbook dealing with engine failures and air start procedures, the pilot is not instructed to energize the emergency a-c generator after flameout. It has been established that a windmilling engine will not deliver a-c power directly through the a-c generator but will produce and maintain sufficient hydraulic pressure to drive the emergency a-c generator. In the event of a flameout condition, therefore, the pilot should switch from the main to the emergency a-c generator in order to maintain electrical power for the primary flight and engine instruments. These instruments include the directional and attitude indicators, fuel quantity, fuel flow and hydraulic pressure gages. Among the latter, fuel flow is important to air start procedure while hydraulic pressure is a critical consideration in dead stick landings.

The other deficiency concerns the check-out procedure which is required prior to loading missiles on the missile launchers and also during rigging of the launchers as outlined in Technical Order 1F-102A-12. The present procedure is to make adjustments and then, using the aircraft electrical and pneumatic systems, retract the launchers. This means that in the event the system is improperly rigged, serious damage can and has occurred to the snubber mount brackets, the unlocking cylinders, the launching actuator cylinders, and the stop bolts. This is an unsatisfactory method since the system should be operated very slowly and with the least amount of pressure required to actuate the launchers.



FUNCTIONAL ANALYSIS

AIRCRAFT SYSTEMS

This section of the report contains discussions of the various aircraft systems from a functional viewpoint. These discussions are arranged in the same sequence in which they appear in the Maintenance Handbook. The discussions are limited to major malfunctions and failure trends which affect the operation of a system. Detailed information on specific functional deficiencies can be obtained by reviewing the Unsatisfactory Reports as listed in Appendix I.

The testing program produced a total of 699 Malfunction Reports and 408 Unsatisfactory Reports. The UR/Flying Hour frequency trend stabilized at approximately 0.3 UR/Flying Hour, which compares favorably with contemporary fighters. With the exception of a few areas which will require modification and further test, the 865 hours of Phase VI flight time has demonstrated the F-102A systems to be functionally sound.

□ power plant

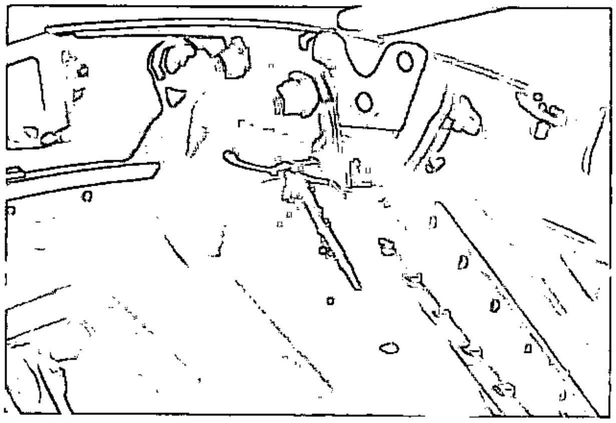
The overall performance of the Pratt and Whitney J57-P23 engine was excellent throughout the Phase VI test period. Durability of the engine was considered outstanding. Undesirable high altitude deceleration and low altitude acceleration compressor stalls were considered a major operational deficiency. Turbine blade stretch was a problem which will require additional contractor study. With the incorporation of certain modifications for compressor stall corrections, the J57-P23 engine will be extremely reliable both operationally and from a maintenance standpoint.

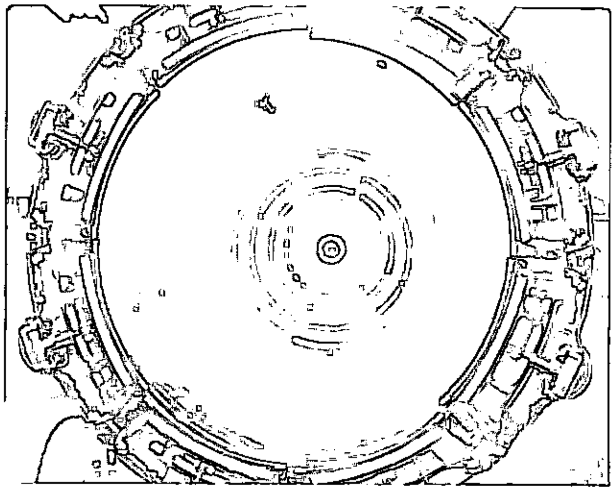
During the Phase VI Flight Test Program, eight engines were used, two of which suffered foreign object damage and were returned for overhaul. Total flight time accumulated on all J57-P23 Phase VI engines was 865 hours, with an additional 73 hours logged during static test cell runs. Two engines operated more than 150 hours flight time with the high time engine reaching 178:30 total flying hours.

□ *Combustion Chamber Weldment:* The only significant recurring discrepancy noted in the combustion section was cracking and buckling of combustion chambers. A total of 38 chambers were rejected during the 13 periodic inspections. This was an average rejection rate of approximately three chambers per periodic inspection. There are eight combustion chambers per engine. The limits used for rejecting combustion chambers were obtained from Technical Order

2J-57-22. Almost all of the chamber rejections were caused by buckling and cracking of the lip where the inner and outer weldment of the chamber are joined. This buckling and cracking was confined mainly to the No. 6 swirl cup area which is the first swirl cup counter-clockwise from the cluster fuel inlet. Insufficient cooling air in this area and resulting high temperatures stress the area, which causes cracking and buckling.

☐ *Afterburner Nozzle Segments:* Considerable difficulty was encountered with afterburner nozzle segment warpage. Twenty-two nozzle segments and one complete afterburner were replaced due to warpage of segments beyond limits. This problem proved to be serious since spare segments were in short supply. Also, no spare air seals were available which necessitated the difficult and time consuming job of exchanging seals each time a segment was changed. The segments were all fabricated from sheet metal and proved to be susceptible to heat warpage.





Two new "waffle" type afterburners incorporating forged steel flap segments and duel tandem actuators were received late in the program for testing. These forged segments represented a contractor modification, and through 270 total hours of operation no discrepancies were noted. The positive opening and closing action and the rapidity of operation of these segments will contribute materially to the capability of the aircraft.

☐ *First Stage Turbine Blade Stretch:* Experience with J57 series engines at this Center indicates that first stage nozzle guide vane bowing is directly associated with first stage turbine blade stretch. For this reason, whenever nozzle vane bowing in excess of the limits of 0.105 inches at the leading edge and 0.175 inches at the trailing edge is encountered, the first stage turbine blades are removed and stretch checked. Two such cases were encountered during this program on engines with 100:50 and 145:00 flight hours, respectively. The first case was noted when nine first stage nozzle vanes were found over limits. The first stage turbine blades were removed, vapor blasted, fluorescent penetrant inspected, and stretch checked. Results showed that two blades were cracked and that all blades were either stretched over limits or very close to the limit. One hundred thirteen blades and 31 nozzle vanes were replaced. The second was discovered when 7 first stage nozzle vanes were found bowed over limits. Disassembly and inspection of the first stage turbine assembly showed that 2 blades were cracked and that all 113 blades were stretched over limits. All blades and 22 first stage nozzle guide vanes were replaced.

In order to obtain more data on turbine blade stretch, a special test was run on a new engine. The test consisted of measuring and installing new first stage turbine blades in the engine. The blades were to be removed and measured each 50 hours for 150 hours of flight time. Unfortunately, the engine sustained compressor foreign object damage after 60 total engine hours and the test was not completed. Some useful data was obtained however. Results of the test showed the average stretch of all 113 blades to be 0.0109 inches during the 60 hours of engine operation. The blades averaged 0.0173 inches under the maximum replacement limit when installed. It is apparent from these average figures that the blades were rapidly approaching the replacement limit. Two blades had already stretched over the maximum limit and several others were close to this limit. Turbine blade stretch has been proven to be a serious problem and warrants further study.

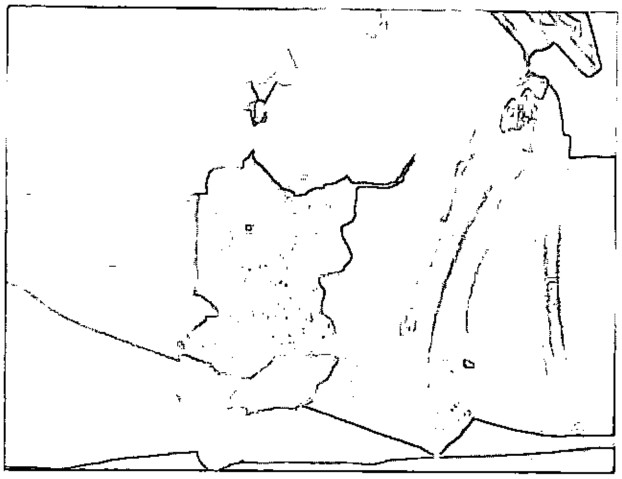
☐ *Compressor Stall:* During a number of flights, engine compressor stalls were encountered as the throttle was being retarded toward idle at altitudes above 35,000 feet. The throttle was retarded either to reduce airspeed for descent or to fully close the afterburner eyelids. The stalls occurred below 82 percent rpm, predominantly in the 79 to 80 percent range. In most instances, they could not be stopped by ceasing aft throttle movement or by trying to advance power, although these methods worked occasionally. In one case, attempts to recover failed and an exhaust temperature of 700 degrees C was reached, necessitating an engine change. In two additional instances, the aircraft were dived supersonically to approximately 30,000 feet where satisfactory recoveries were effected without encountering exhaust temperature overheat conditions.

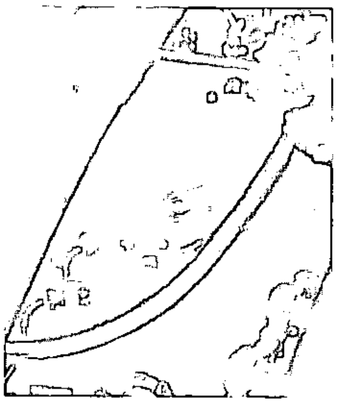
A fourth instance of this type of compressor stall occurred when a maximum power climb was terminated. At 50,000 feet, the throttle was moved from the full afterburner position to the military power position. Compressor stalls began immediately and continued, unaffected by various throttle adjustments, during a supersonic descent to lower altitude. The stalls continued and at approximately 20,000 feet, with the throttle in "IDLE" and the engine maintaining 79 to 80 percent rpm, the engine was shut down. Airspeed was maintained at 220 knots as engine rpm decreased. The stalls stopped at 30 percent rpm and at 27 percent rpm an air start was successful, although accompanied by some "chugging" until idle rpm was stabilized.

As a result of these difficulties, the Phase VI aircraft were restricted to a lower limit power setting of 83 percent rpm at altitudes above 30,000 feet. This engine deficiency seriously limited the usefulness of the F-102A weapon system in addition to preventing certain tests normally required in Phase VI testing. Many maneuvers in the Air Defense Command missions will require a flexibility in engine operation which is not provided in the present installation. A temporary solution to the engine deceleration stall problem was found by rescheduling the bleed valve governor on aircraft S/N 1805 to the schedule shown by the Pratt and Whitney Overhaul Manual for Pratt and Whitney part No. 272736. At all airspeeds and at altitudes from sea level to 56,000 feet for all rates of throttle movement, the pilots reported that they were unable to induce an engine deceleration stall.

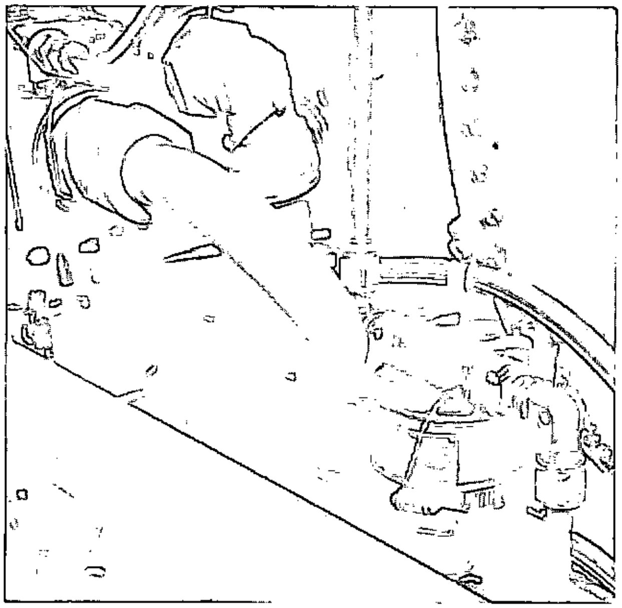
This schedule is adequate for interim use until an optimum schedule can be developed that will least affect maximum endurance while retaining adequate stall margin.

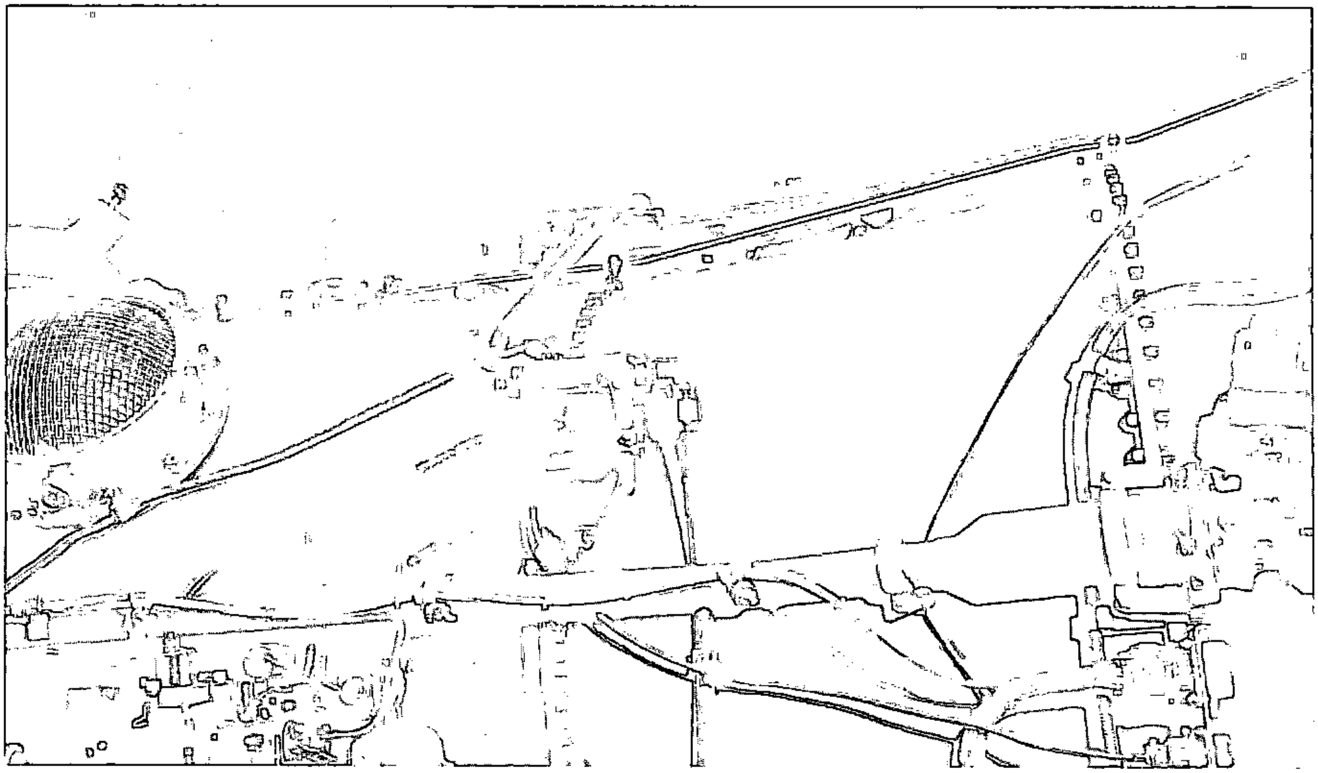
☐ *Cooling Air Bleed Duct:* Seven cooling air bleed ducts failed during the first 200 hours of Phase VI flying time. These failures were attributed to abnormal air pressures created by engine surge. This resulted in the eventual rupturing of these N_2 bleed ducts. Ducts of heavier gage material were designed and only one failure of a duct of this type occurred during the remainder of this program.





☐ *Oil Filter — Sundstrand Unit:* Difficulty was encountered in gaining access to the return oil filter for the Sundstrand Unit. The position on the engine was such that the engine had to be removed from the aircraft in order to inspect this filter assembly. Applicable technical orders required that this filter be inspected at 25-hour intervals. Since the operating time between engine inspections was 50 hours, the filter was relocated to a point approximately 15 inches lower on the engine to permit accessibility through the right-hand wheel well so that it is not necessary to remove the engine to inspect this single item.

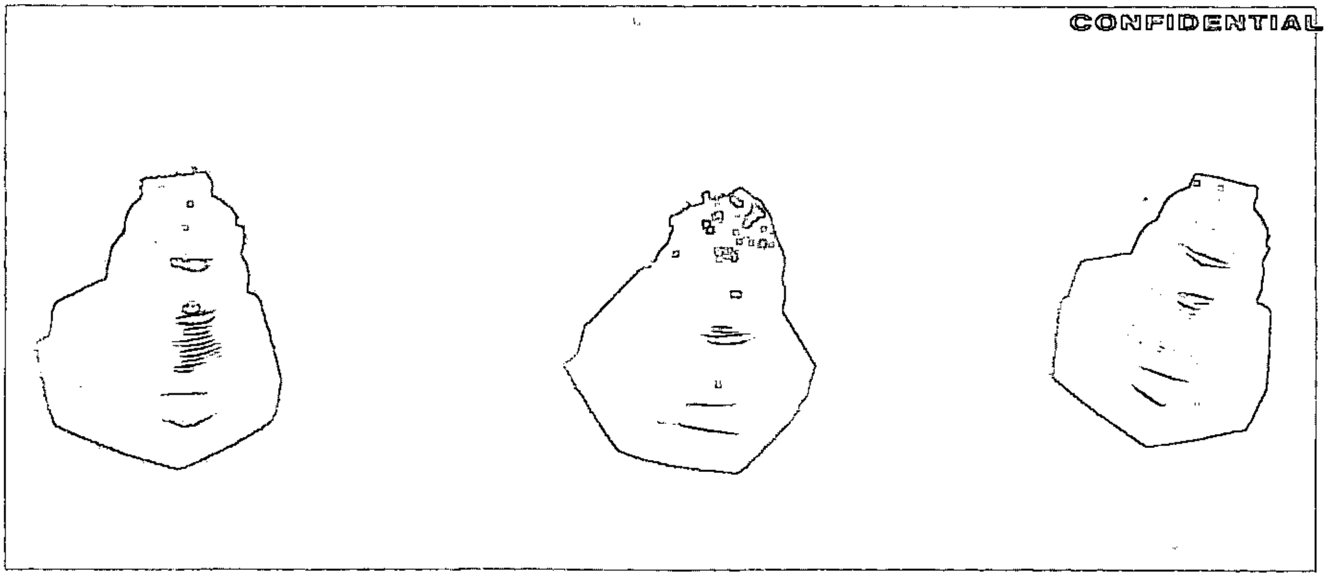




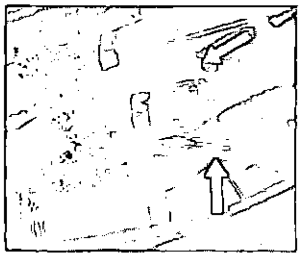
☐ sundstrand constant speed drive unit

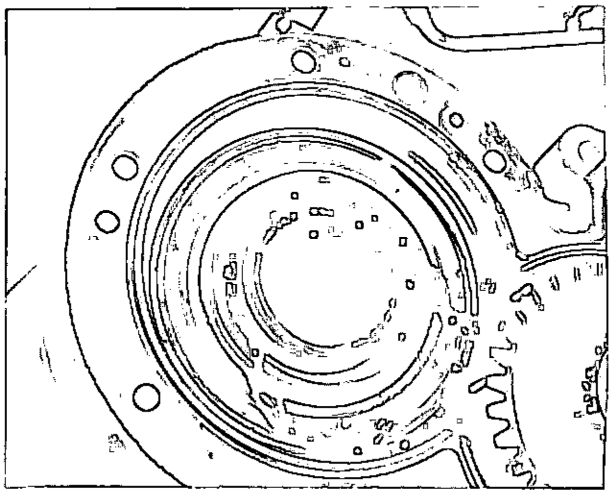
The Sundstrand drive unit demonstrated relatively good service throughout the test program; however, a number of failures did occur in the engine mounted gear box, the drive, and the transmission.

☐ *Gear Box:* Three failures of the engine mounted gear box showed the service life which could be expected from this unit. In each case, internal failure was indicated through discovery of metal particles on the

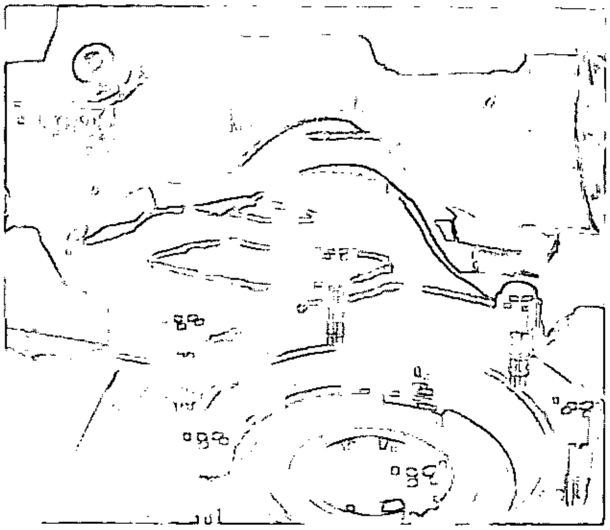


magnetic sump plug. The operating times on the gear boxes at the time of failure were 121:30, 143:00, and 161:20 hours. Disassembly and inspection of each of the gear boxes revealed failures of bearings and bearing retainer rings. In one case, bearing seizure had been induced through the resultant friction and attendant high temperatures. A pending failure was discovered after a spot inspection of another gear box had revealed a small number of metal particles clinging to the magnetic sump plug. Further inspection revealed that a ball bearing retainer ring had begun to deteriorate. The gear box had been operating for 129:20 hours and would have failed during the next flight had the pending failure not been discovered.

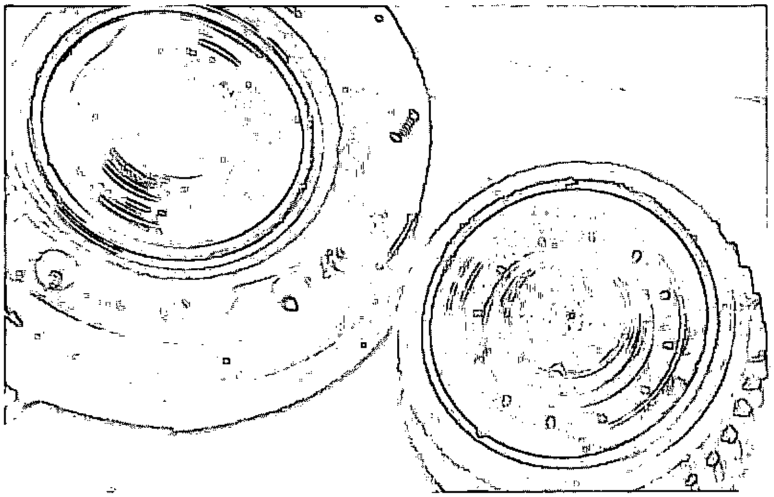




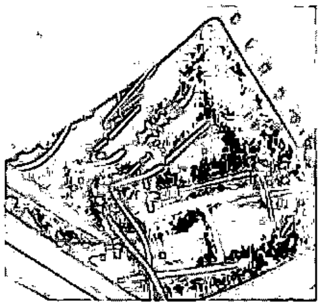
☐ *Drive Shaft:* Loss of a-c power while airborne occurred on several occasions. In two cases the loss resulted from failure of the interconnecting shaft which transmits rotational power from the engine mounted gear box to the transmission. The failures had occurred through separation of the shaft from a universal joint which is integral with the transmission. This separation resulted in random whipping rotation of the shaft and consequent damage to the shaft cover and the refuel crossover tube which is installed adjacent to the shaft. This whipping action also caused the failure of one engine mounted gear box.

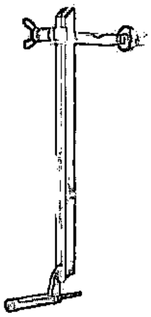


☐ *Transmission:* During a periodic inspection a number of metal particles were discovered on the remote gear box sump plug. Further inspection revealed that one of the idler gear roller bearing assemblies installed in the remote gear box housing on the drive transmission contained pitted rollers, and that the bearing inner rollway was badly scored. No cause for the bearing failure could be determined, and the assembly was released to the contractor, Sundstrand Machine Tool Company, for study.



☐ *Cover Plate Bolts:* A design deficiency exists in the proximity of the cover plate and filter element retaining cap screw heads to the raised portion of the cover plate. The inspection requirement for the removal of the cover plate and filter element is very difficult to perform with the drive assembly installed in the aircraft. The use of an open end or socket wrench to remove these screws has proven to be very time-consuming and often results in damage to the cap screw heads.





☐ *Universal Joint Alignment:* The alignment of the Sundstrand constant speed drive unit universal joints, which is required each time the engine is reinstalled, repositioned, or a new drive unit is installed on the aircraft, is a tedious and time-consuming operation. Technical Order 1F-102A-2-4, Paragraph 9-5, specifies the use of a 12 inch straight edge and a 6 inch scale with decimal increments for measuring the "gap" and "interference" of the universal joints. These universal joints connect the engine mounted gear box to the airframe mounted Sundstrand transmission by means of a shaft. The 16 measurements which must be taken require two crew men approximately 1 hour. There is also considerable chance for an error to develop from misreading any one of the measurements. To reduce the possibility of an error and to permit a reasonably rapid and accurate alignment of the universal joints, a specially designed precision alignment tool should be made available.

☐ *Engine Oil System:* The fitting, P/N 8-27961, which directs engine oil to the engine oil pressure switch in one direction, and lube oil to the Sundstrand gear box in the other direction, was improperly installed on one engine. This resulted in a low oil pressure reading and in an improper quantity of oil to the Sundstrand gear box. The design of this fitting should be such that it can only be installed in the proper position.

☐ *Avica Hose Assemblies:* Prior to engine shutdown following a test flight, fuel was observed to be leaking from the fuselage in the engine bay area. The engine was shut down immediately. The lower engine bay had filled with fuel and investigation revealed that the fuel flow equalizer control line, a flexible Avica hose, had failed. Visual inspection of the hose assembly was difficult because these assemblies had a braided wire external covering, but this hose was determined to have failed internally. Because of the nature of the failure, an emergency Unsatisfactory Report was submitted. This UR resulted in a requirement from AMC to inspect all Avica hose assemblies on F-102A aircraft for indication of failure. Each hose determined to be faulty was replaced with a like serviceable item. A total of 42 hoses were replaced on Phase VI aircraft as a result of this inspection.

☐ fuselage system

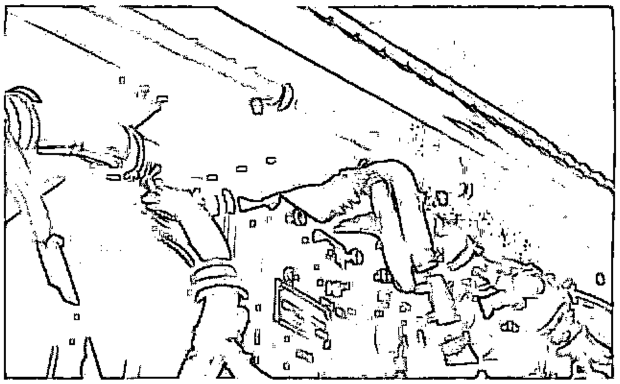
The fuselage of this aircraft is considered to be extremely well fabricated. No discrepancies of any note were revealed except the lack of an access panel to the afterburner igniter valve. A few instances of elevon edges chafing the fuselage at the neutral position occurred, but were remedied by slight hand fitting.

☐ *Fuselage Access Panel:* The lack of a fuselage access panel for the engine mounted afterburner ignitor valve and afterburner nozzle position switches constitutes a serious design deficiency. The engine must be removed from the fuselage to perform maintenance on either of these units. This caused unnecessary delay in the flight test program and precipitated other malfunctions which might occur during engine removal and reinstallation. In attempting to repair one particular leaking ignitor valve which was later damaged in engine removal, the engine had to be removed and reinstalled three times. This time-consuming process resulted in a total of 3 days out-of-commission essentially for the lack of one small access door.

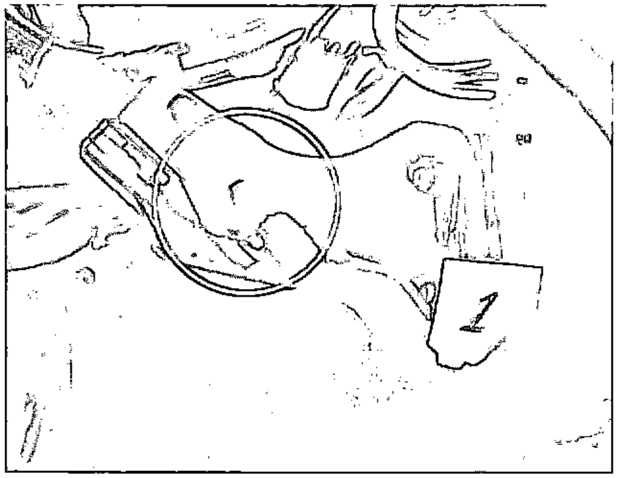
The contractor completed a design study of this problem and incorporated an access panel in the Phase VI aircraft. Installation of this access panel permitted maintenance personnel to perform repairs to either the afterburner ignitor valve or afterburner nozzle position indicator without the former requirement to remove the engine.

☐ *Left-Hand Aft Radar Access Door:* During a routine Phase VI flight on aircraft S/N 53-1808, the left-hand aft radar access door was torn from the aircraft. The door separated shortly after take-off while the aircraft was in a climbing turn to the left at 450 knots IAS at 8000 feet pressure altitude. The loss of the door was accompanied by a loud explosion and serious airframe vibrations. Throttle reduction lessened the severity of vibration and the flight was immediately terminated. Inspection of the aircraft structure and engine condition revealed the following:

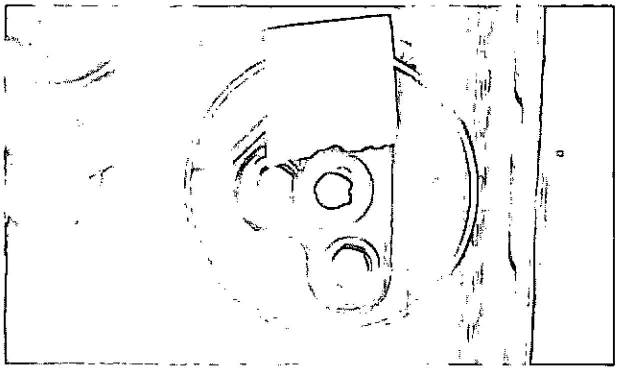
1. The hinge pin which connected the access door to the aircraft had sheared in a number of pieces, some of which remained with the aircraft. Those pieces of the hinge pin which could not be found are assumed to have been lost overboard or to have entered the engine through the left-hand engine inlet duct.



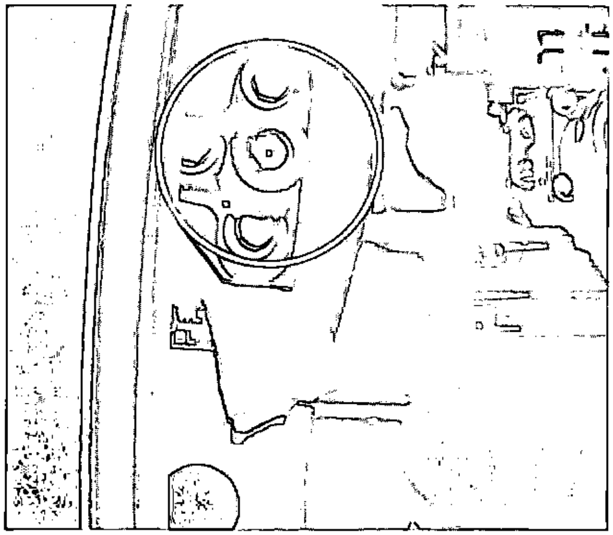
2. A piece of metal had broken away from the aft door latch support assembly.



3. Slight scoring was noted on the door assembly aft fitting in the vicinity of the shear pin hole.



4. Material around the forward fitting in the vicinity of the shear pin hole had been heavily gouged away.



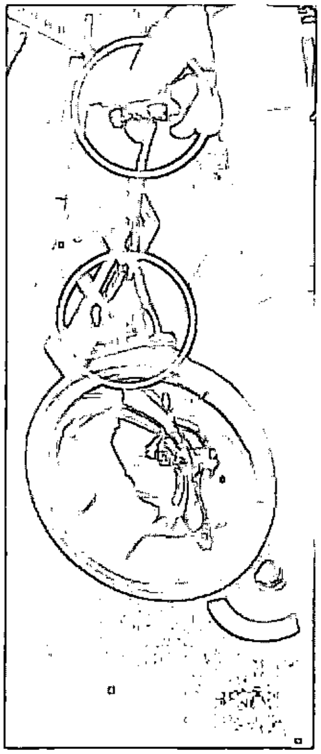
5. The vertical support assembly to which the forward fitting is attached was severely twisted and damaged.

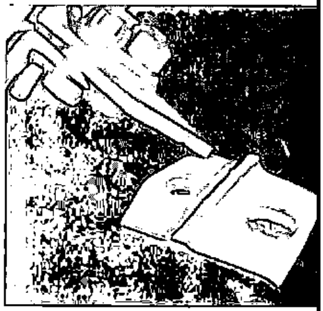
6. A number of first stage blades in the compressor section of the engine sustained damage in the form of nicks which were of sufficient magnitude to require engine removal for further inspection.

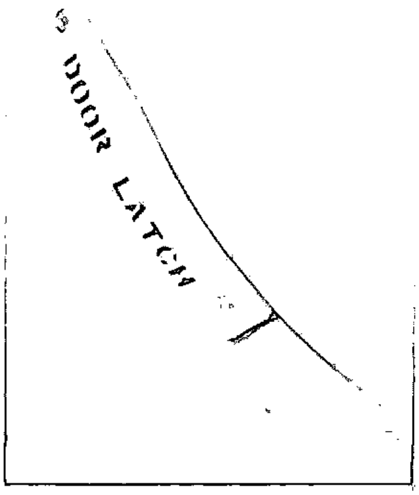
Prior to this particular flight, the aircraft had made a radar flight of 1:30 hours and a radar pressurization discrepancy had been noted. Maintenance personnel corrected the discrepancy by gaining access through the right-hand aft and forward radar access doors only. At completion of the work the doors were secured. Since the lost door was not found, the reason for the separation was not definitely determined. However, such factors as material or structural failure were considered possibilities. It is especially noted that with pressure the radar door latch covers can be made to fit over an unlocked door latch. The gap varied on different aircraft which were inspected from a very slight amount to approximately 1/4 inch as compared to the structural curvature of the fuselage.

☐ wing

There were very few malfunctions of the wing components during the Phase VI program. The most predominant problem was failure of the landing light brackets. These failures were attributed to metal fatigue induced by vibration and airloads, and resulted in breaking the light and loss of landing light capability. This problem was eliminated by a contractor modification which consisted of using heavier ma-

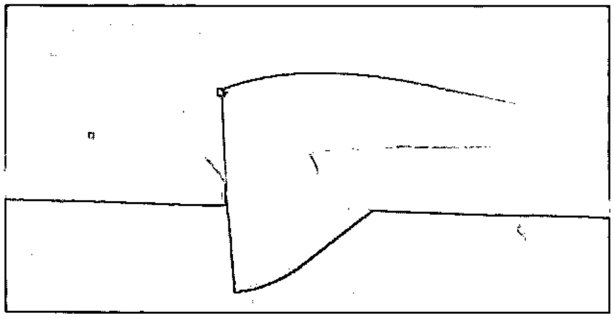






chined aluminum rather than the original light aluminum cast bracket.

Another structural problem consisted of the partial crushing of the three wing fence honeycomb sections. This was caused by poor installation and resulting interference with the protruding edge of the adjacent leading edge skins. In order to preclude similar failures in the future, the contractor has strengthened the forward section of the wing fence assembly and initiated more stringent quality control measures.

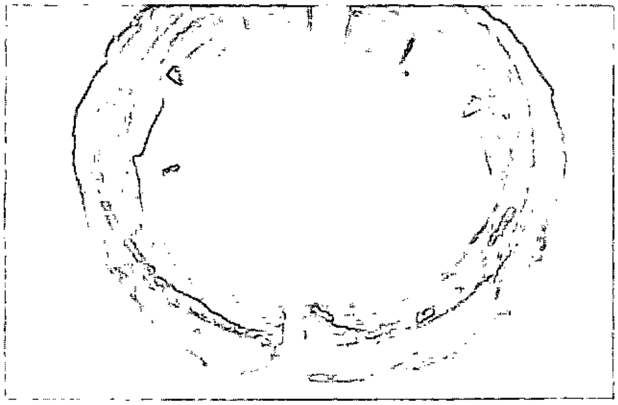


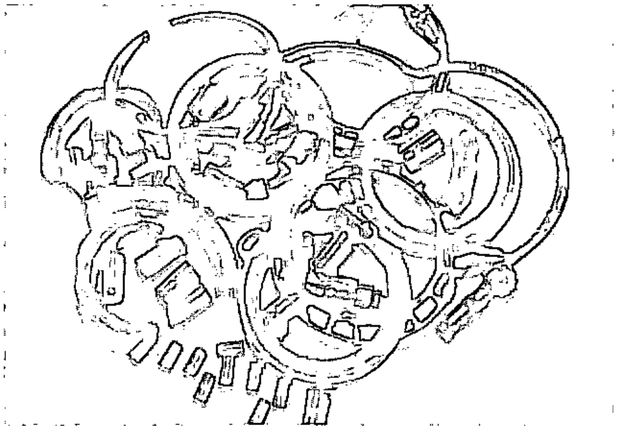
☐ landing gear

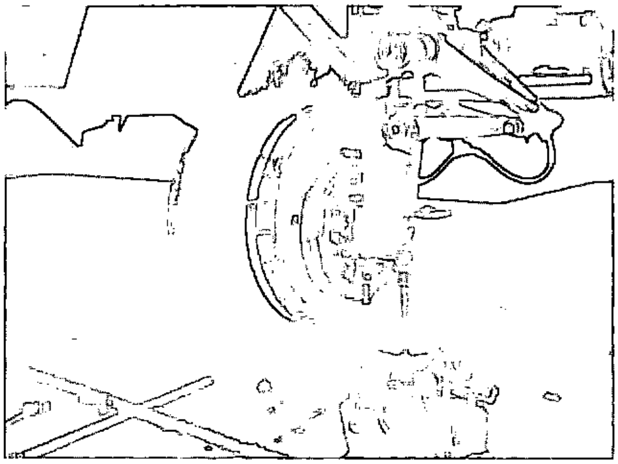
In general, the landing gear system has proved very satisfactory under conditions encountered at the AFFTC. During the missile firing phase at Holloman Air Force Base, excessive tire wear occurred due to rocks on the runway and taxi strips. A comparison of tire reliability between Holloman Air Force Base and Edwards Air Force Base indicates an average of 5 landings per tire versus 15 landings per tire, respectively. Both of these figures are based on minimum of braking due to the very long runways at both installations. Construction in progress at Holloman accounted for the rocks on the runway and normally would not be a problem.

☐ *Wheel Bearing Failure:* During the take-off roll of a projected test flight, the pilot elected to abort the mission because of low tailpipe temperature. As the roll continued towards its termination, the pilot received indications that the right-hand main landing gear tire had blown out. As the aircraft decelerated, the pilot executed a gradual turn to the left-hand side of the runway and stopped after a total roll of 9600 feet.

Investigation disclosed that the main landing gear wheel retaining nut was in two pieces with the hexagonal shoulders of the nut broken off; the pieces were found 7800 feet from the start of the take-off roll. Wheel bearing rollers were deposited from 8,000 to 8,800 feet. It was indicated that the tire blew out at 9200 feet. The wheel separated into two sections as the retaining bolts were sheared.





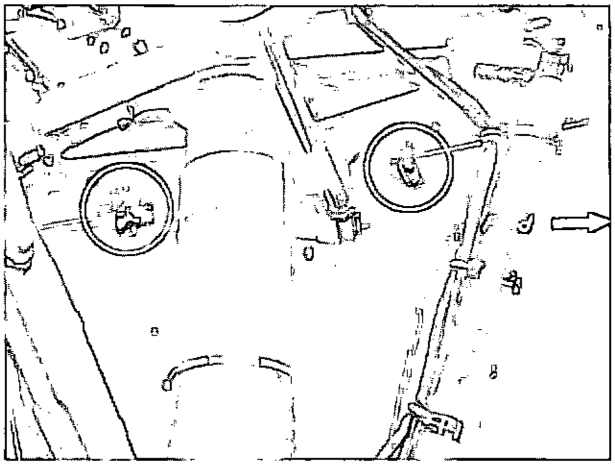


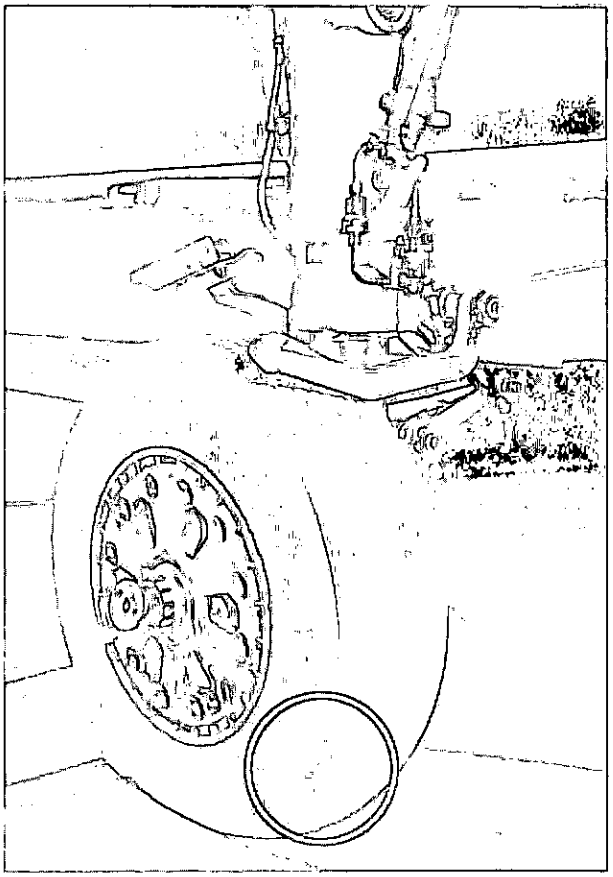
The bolt which retains the wheel retaining nut was still in place in the axle, but the head and nut were missing. The axle journal and the axle end were severely mutilated and replacement of the inner strut and axle assembly was required.

Although the exact cause of the incident was not definitely determined, the outer wheel bearing apparently seized in the cone. The wheel casting disintegrated outside of the cone area, eventually breaking the hexagonal shoulder from the retaining nut and causing the nut to separate.

Since the aircraft involved had been temporarily based at Holloman Air Force Base for a period of approximately 2 months and had been exposed to several thunderstorms, it is conceivable that the bearing could have been subjected to water infiltration, with subsequent rusting and corrosion which could have precipitated the bearing failure.

☐ *Main Landing Gear Fairing Door Failure:* During the testing program three landing gear doors failed in flight. These failures occurred during letdown and on take-off. In all cases, the failures were attributed to excessive speeds at time of gear extension. To prevent future landing gear door failures, the maximum gear down speed was reduced from 240 knots IAS to 220 knots IAS. A mechanical change was later incorporated to prevent door failures by strengthening door linkage and attaching pins which will allow the pilot to attain his placarded speed without damaging the doors.





☐ air conditioning, pressurization, and anti-ice systems

Cockpit Air Conditioning and Pressurization: The cabin air conditioning and pressurization system was unsatisfactory throughout the entire testing period. Cabin pressure surges, failure of cabin pressurization, and failure of the cabin air heating and cooling system occurred repeatedly, disrupting orderly progress of scheduled missions.

Air conditioning is obtained by bleeding air from the main engine compressor through strut bleed ports at high temperature and pressure. The air is then routed forward through a flow control and shut-off valve to a refrigeration unit. Pressure and temperature are reduced by first passing the air through an air-to-air heat exchanger and then expanding it through a cooling turbine. An axial flow fan, mounted on a common shaft with the turbine wheel, absorbs the energy of the turbine wheel by drawing cooling ram air across the heat exchanger. Turbine discharge air is then discharged into the cockpit through the distribution tubes.

Ventilating air is supplied for the forward electronics compartment, intermediate electronics bay and motor generator by ram air. The ram air is bled from the left-hand boundary layer duct through a pressure regulator and temperature operated shut-off valve.

To eliminate smoke in the cabin when a missile is fired, a flow control shut-off valve is closed with a "missile fired" signal. During this period, cockpit pressure may drop due to leakage and a rate sensor unit is provided to control the re-opening rate of the shut-off valve. This prevents an initial pressure surge in the cockpit. During engine ground operation when no ram air is available for electronic cooling, conditioned air is automatically diverted from the cockpit to the electronic equipment.

☒ *Pressure Surges:* During the early portion of the program, cabin pressure surges occurred when the test aircraft were flown at speeds in excess of 440 knots IAS at altitudes below 10,000 feet. These surges occurred at a rate of from 1 to 1½ cycles per second and covered a pressure differential of plus or minus 1000 to 2000 feet of altitude.

At 10,000 feet cabin pressure surging stopped and continued climb could be effected satisfactorily. On several occasions pressurization failed completely at altitudes of 35,000 feet and above. As pressurization failed, pilots would descend immediately to lower altitudes. Pressurization in most instances would return quite suddenly at approximately 25,000 feet.

Descents below 10,000 feet at airspeeds greater than 440 knots IAS would once again be accompanied by the same pressure surges as those experienced during climb.

To avoid subjecting pilots to this pressure surging, until a more positive remedy could be found, all flying below 10,000 feet was restricted to a maximum of 400 knots IAS.

In an effort to determine the reasons for cabin pressure surging the cabin pressure rate sensor relay was inspected and found to be incompatible with a pressure tube because of the 52½ degree flare on the rate sensor could not match the 37½ degree standard flare on the tube. The Stratos Corporation, manufacturers of the pressure rate sensors, furnished new sensors having the proper standard flare angle. This reduced pressure surging to a great degree, but did not completely eliminate it. No specific reason was found for the intermittent surging which now exists.

☐ *Turbine Assembly:* The turbine assembly unit used in the refrigeration package failed six times. These malfunctions were attributed to bearing failure. All indications are that the shaft mounting design is faulty and that the bearing materials could be improved to eliminate this problem.

☐ *Cabin Heat:* The cabin heat control box failed five times. With the control in automatic range the pilot sometimes receives no heat at all or too much heat. The dependability and service life of this unit is not satisfactory. A redesign of this control is mandatory.

☐ *Windshield De-fogging:* Windshield de-fogging is accomplished by electrical heating of NESA glass windshield panels and is controlled through the windshield switch located on the pedestal switch panel which is placarded "NESA," "NORMAL" and "OFF." With the NESA switch in the "NORMAL" position, the NESA control applies voltage to the heating elements which are imbedded in the vinyl layer of glass. The NESA control maintains the temperature of the vinyl layer of glass at a pre-set value and within a plus or minus 5 degree F range.

The NESA windshield de-fogging system did not operate properly during any part of the test program. The principal deficiency consisted of a faulty relay which would stick and ultimately burn out, rendering the system inoperative. On several occasions pilots whose windshields had fogged over as a result of a fast descent, were required to fly at low altitudes at high power settings and increased cockpit heat for as long as 5 minutes in order to clear the windscreen sufficiently to insure safe visibility for landing.

An attempt was made to provide a means of clearing the windscreen more effectively than the method mentioned above. A flexible hose, which had been tapped into a hot air line, was provided in the cockpit for the use of the pilot. In the event fogging occurred the pilot needed only to direct the flow of hot air from the hose onto the windscreen, thereby de-fogging it. This method was satisfactory and a sufficiently large enough portion of the windscreen could be de-fogged to insure safe visibility. However, the crude installation was not considered to be a proper correction for this deficiency.

The contractor (Convair) was aware of the NESA deficiency at the beginning of the Phase VI test program, and recommended modifications. The necessary kits containing these improvements were not received in time to be tested during the program.

☐ hydraulic system

In general, the hydraulic system of the aircraft is acceptable. There were malfunctions and failures of various parts of the system, but a definite failure trend was not established. Hydraulic power is supplied by two completely independent, continuously operating, closed center systems. The primary system supplies power for the operation of the flight control valves and actuating cylinders, in addition to the elevon manual lock-out valves, elevon and rudder control servo actuators, speed brakes, landing gear, landing gear doors, nose wheel steering damper and the emergency alternator. Normal operation of both hydraulic systems is automatic whenever the engine is running.

An emergency system supplies hydraulic power for the primary system only, thus providing sufficient power for limited maneuverability and a safe, normal approach and landing. This emergency hydraulic pump is driven by a ram air turbine which the pilot can displace into the air stream by means of the high pressure pneumatic system. This turbine cannot be retracted in flight. This emergency pump delivers 6.3 gpm minimum at 1000 psi at 250 knots IAS. The primary and secondary pumps are 22 gpm, 3000 psi, constant pressure, variable volume, engine driven pumps.

A total of 14 failures on the emergency hydraulic system pumps indicated an abnormally low life expectancy of approximately 75 operating hours. The basic cause was external leakage at the rear cover seal. Near the end of the testing program, 6 end cap failures on the pump compensator control occurred which had not previously been prevalent. As a result of these failures, the secondary system of airplane S/N 53-1805 was instrumented to determine whether abnormal system pressure surges were a contributing factor.

Instrumentation consisted of transducers and flow meters being installed at strategic points such as the hydraulic pump inlet, drain to reservoir, pump outlet to control actuators, and at the accumulator. All readings from these instruments were recorded on an oscillograph. The tests were performed first on the ground to insure proper operation, and then in the air at near maximum altitude. The pilot performed various violent maneuvers at g loadings. No malfunctions occurred.

It is to be noted that these tests were primarily designed to obtain data pertaining to two specific trouble areas: (1) rear cover seal leakage, and (2) pump compensator cover failures. Pump case pressure conditions are related to the rear cover seal problems and pump outlet pressure conditions relate to the compensator cover failures. Outlet pressures have no bearing on the rear cover leakage and, conversely, case pressures have no bearing on the compensator cover failures.

Review of the oscillograph test records indicate that no unusual outlet pressure conditions exist over the full range of system operation. Pressure surges of 3300 to 3600 psi were the maximum experienced, which is well within the 4000 psi design requirement. An analysis of broken specimens by Vicker's Incorporated determined that cap-screws of poor strength quality caused the compensator cover failures. A program has been undertaken whereby Vicker's field service personnel will rework all delivered pumps to install an improved cover and cap-screw of proper strength on the pump compensator. Identification of the rework will be made by addition of a suffix letter after each pump serial number.

For normal ground operation and flight maneuvers, pump case pressures appear to be within the design limits of the pump. It was determined that under some extreme flight maneuver conditions case pressure peaks of approximately 280 psi can be experienced. This is considered excessive over an acceptable value of 150 to 175 psi. Accordingly, Convair has initiated an Engineering Work Order to relocate the case drain connection close to the system reservoir. This will provide a more direct case drain line and should substantially reduce pressure peaking within the case. Upon satisfactory mock-up and checkout on the controls test stand this change will be scheduled into production with Service Action recommended. This change will assist in alleviating the pump rear cover leakage problem.

In addition, Vicker's Incorporated has made an engineering change to replace an initial flat gasket at the rear cover with a superior molded "O" ring seal. The new seal is currently being assembled into delivered pumps. Existing pumps returned for repair are also being modified to take the new seal. This change, in combination with the proposed Convair plumbing change, should eliminate the pump rear cover leakage problem.

☐ pneumatic system

The pneumatic system was not satisfactory because the pneumatic pressure regulator repeatedly malfunctioned. The principal failure trend consisted of the regulator releasing pressure before 3000 psi (normally capacity) had been reached.

☐ *Pressure Regulator:* Fifteen specific failures have been reported in Unsatisfactory Reports. These failures were attributed to leakage of air through the vent ports as a result of faulty internal seals. No new design change was received from the contractor. A description of the high pressure pneumatic system will aid in the identification of this deficiency.

Air at 3000 psi is stored in fiberglass flasks of approximately spherical shape for operation of the high pressure pneumatic system components. Two flasks are required when the aircraft carries missiles; two additional flasks are required when the aircraft carries rockets only. The system is charged from a ground source. The air is filtered in the system prior to storage in the flasks.

The high pressure pneumatic system performs the following functions:

1. Opens and closes missile bay doors.
2. Lowers armament racks into and retracts racks from firing position.
3. Provides pressure for rudder feel cylinder.
4. Extends ram air turbine into operating position for emergency hydraulic power.
5. Provides air pressure for emergency landing gear extension.
6. Provides air pressure for drag chute deployment.
7. Provides air pressure for main wheel brakes.

One of the permanently installed air flasks (the priority or isolated flask) is directly connected to the manually operated valves which control ram air turbine extension, emergency landing gear extension, drag chute deployment, and is also connected to the ground charging line and the line supplying pressure to the main wheel brake drag brace reservoirs. The connection to the remainder of the system from this flask is made through a priority valve which will not permit air to flow from the isolated flask to the remainder of the system when pressure in the isolated flask is 1200 psi or less. Therefore, discounting leakage, because of the priority valve the pressure available at any time to the three manually operated valves mentioned above and to the brakes is a maximum of 3000 psi and a minimum of 1200 psi.

Air pressure for the rudder feel cylinder, missile bay door actuation and armament rack operation is held to 1100 psi by a regulator and relief valve. The relief feature of this valve protects the system components downstream by venting system pressure above 1250 psi to the atmosphere. The valve reseats at 1155 psi.

A pressure switch, located in the pressure line, operates a warning light on the pilot's warning light panel. The light comes on when there is insufficient pressure remaining in the pneumatic system for a combat cycle (1500 plus or minus 50 psi). A combat cycle consists of opening the missile bay doors, extending and retracting the armament racks and closing the doors.

☐ *Ground Air Supply:* Another problem was the lack of provisions for charging the high pressure pneumatic system in flight. Leaks can and do occur. If a leak occurs in the priority valve, pressure may be lost to the entire system exclusive of the wheel brake reservoirs. If the leak occurs in a wheel brake valve assembly pressure is lost in that brake as well as in the rest of the system exclusive of the other brake. In such an event, the pilot will be landing with only one brake operative and the drag chute system inoperative.

The warning light alerts the pilot that the total system pressure is down to 1500 psi or less, but does not isolate the condition described above. Thus, the pilot has no method of determining whether or not the loss of pressure involves the priority system. Even with one brake inoperative (due to a leak) the pilot will not know that a leak has developed in the priority system. This is because the brake pedal pressure is supplied by the hydraulic system and is not affected by the pneumatic portion of the brake system.

Since normal depletion of the pneumatic system pressure to the point of actuating the warning light (1500 pounds) will occur frequently as a result of using the armament system, the pilot will make many landings without knowing whether these landings will be normal. Thus, for the purposes of normal operation as well as for emergency operations (priority system leak), the pilot should be able to isolate the cause and location of system depletion.

A gage should be provided in the cockpit to indidate priority system pressure to the pilot. This gage could be similar to and possibly replace the gage in the right-hand wheel well. If the present pneumatic system pressure warning indication is retained, the gage need not be in a prominent location and can be located where most convenient. The source of air pressure for the drag chute should be independent or isolated from the remainder of the priority system by a check valve. This provision would insure that the drag chute will be available when the priority system failure described above occurs. Consequently, even with one brake inoperative, the pilot could bring the aircraft to a safe stop by combined use of the drag chute and nose wheel steering.

☐ *Electronic Cooling:* Five failures of the ram air regulator occurred in the jet pump system used to cool the electronics bay area. In all cases, the regulator failed to open when the oil pressure warning switch was actuated during engine idle. This regulator affects supply of reverse air flow cooling and is one of the major causes of radar operational difficulties due to the radar electronic equipment becoming overheated.

One reason for these failures is insufficient inspection provisions of the regulator valve. The required inspection (post-flight) of this valve consists of checking the position of the pressure valve indicator arm. It is possible for this to be normal and the valve to be inoperative. To insure against this occurrence, the post-flight inspection should include an operational check which can be accomplished by using a mirror and visually observe the forward end of the butterfly shaft which is marked at the open and closed position to see if it actually changes position when desired.

Failures of these valves could be due to the pressure switch being open when the engine is removed. The valve is electrically opened and d-c power is on on for lengthy periods during periodic inspections, thus causing possible damage to the valve.

To prevent this occurrence the cabin air circuit breaker should be pulled whenever the aircraft is undergoing periodic maintenance of any type. This becomes mandatory in the ADC squadrons where the bulk of the radar periodic maintenance is performed while the engine is removed. If the cabin air circuit breaker is not pulled the ground cooling air will be dumped overboard through the left boundary layer air duct.

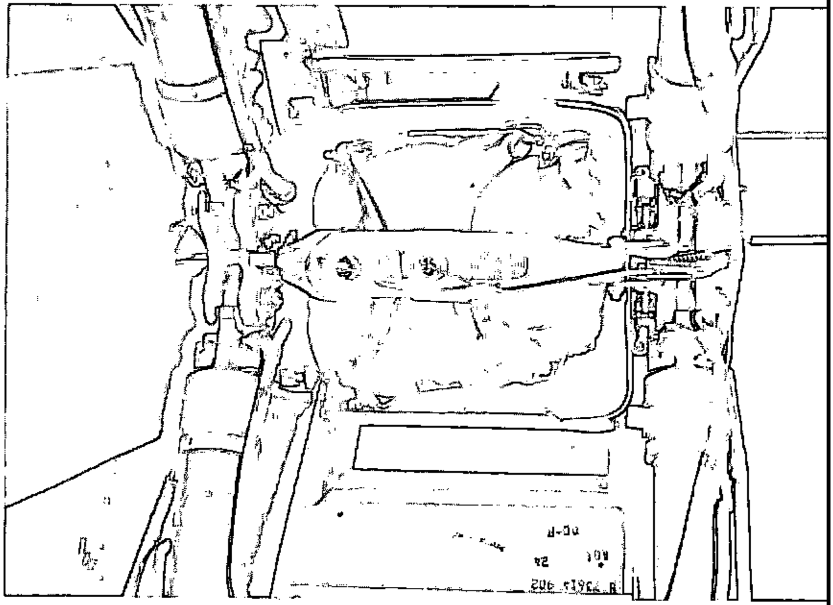
☐ flight control system

The flight control system was satisfactory in most respects. The flight surfaces are actuated by an irreversible, full hydraulic power control system which includes artificial feel, gyro-stabilized pitch and yaw damper systems and trimming devices. Full hydraulic power is used because of the high forces produced on elevons and rudder at high speeds. Both primary and secondary hydraulic systems (or one of them in case of failure of the other) supply the required pressure for operation of the flight controls. The elevon control system combines elevator and aileron motion in a single pair of elevons located on the wing trailing edge. The elevons move in parallel motion as elevators and in opposite directions as ailerons, or simultaneously in any combination of the two motions. pilot control motion mechanically moves the spools in the hydraulic control valves which meter hydraulic pressure to the surface actuating cylinders. The controls move only as long as the pilot moves his controls, and when control motion stops a mechanical follow-up mechanism shuts the control valve off.

A pitch and yaw damper system is provided to damp out short period oscillations of the airplane and to provide automatic turn coordination. Pitch, roll, and yaw rate gyros measure angular velocities about the airplane axis and apply signals to the elevator and rudder servo actuator amplifiers. The pitch and yaw damper system is augmented by the installation of a sideslip feedback system. This system is better known as Beta Feedback and was designed to sense sideslip and to automatically supply sufficient rudder deflection to maintain a zero sideslip angle. All Phase VI aircraft were received and accepted with this system inoperative. A retrofit modification to the aircraft, which increased the area of the vertical tail by approximately 40 percent, had already been approved at the start of the Phase VI program. The "Big Tail" modification eliminated the requirement for the Beta feedback system in production aircraft; therefore, no functional development tests were performed. A discussion of the operation and effectiveness of this system can be found in the F-102A Phase VI Stability and Control Technical Report.

A 14.5 foot diameter drag chute is carried in a structural housing located above the fuselage tail cone. The speed brakes are used as drag chute housing doors. The chute attachment jaws, which are spring loaded to the open position, permit the chute to jettison if it is inadvertently deployed in flight. Closing the jaws to connect the chute to the airplane structure and pulling the chute rip cord is accomplished by means of the high pressure pneumatic system. The jaws are unlocked and spring loaded in the stowed position. When the pneumatic cylinder is actuated, the jaws are closed and locked. The pilot chute pulls the deployment bag out to the maximum length of the risers and then pulls the main chute from the deployment bag. The chute is mounted on the aircraft center line and horizontally in line with the airplane cg when the airplane is in a normal landing attitude. This location permits deployment prior to main wheel touchdown. Jettisoning is accomplished by removing pressure from the drag chute cylinder. The riser shackle is assembled by means of a shear pin which is designed to fail if the chute is deployed at speeds in excess of 180 knots IAS.

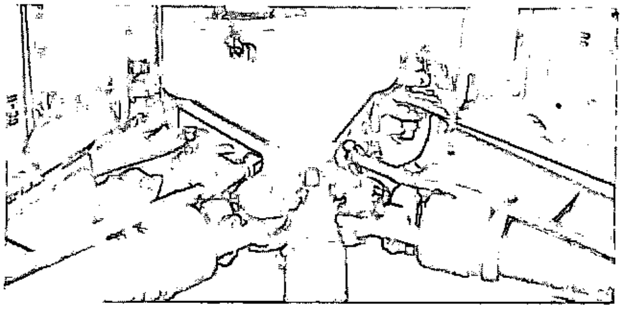
☐ *Drag Chute:* During the initial portion of the test program (2 month period) 100 drag chute deployments were made of which, 15 failed. The majority of the failures were due to improper installation of the drag chute into the canister. The riser strap was being pushed into the back of the canister and at the same time the "D" ring was being wedged in the locking jaws. This prevented them from closing and producing the motion required to pull the ripcord cable which releases the pilot chute. The remaining failures were due to the improper packing of the chute so that the pilot chute retaining pin was inserted at an angle. Thus installed, the retaining pin cannot be pulled out and chute deployment cannot be effected.

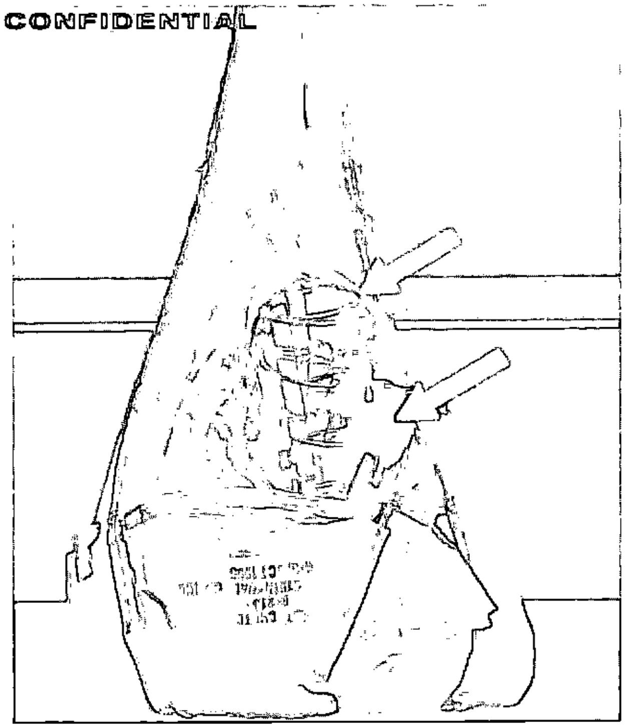


A number of modifications were made on the system. These include:

1. Shorten the riser straps from 54 inches to 18 inches, thus minimizing the problem of riser straps fouling the release of the chute.

2. Shorten the cable assembly, P/N 8-45359, from 9 inches to 6⅛ inches so that the "D" ring will not prevent the locking jaws from closing.





3. Modification of the pilot chute to prevent the spring in the cone from coming loose.

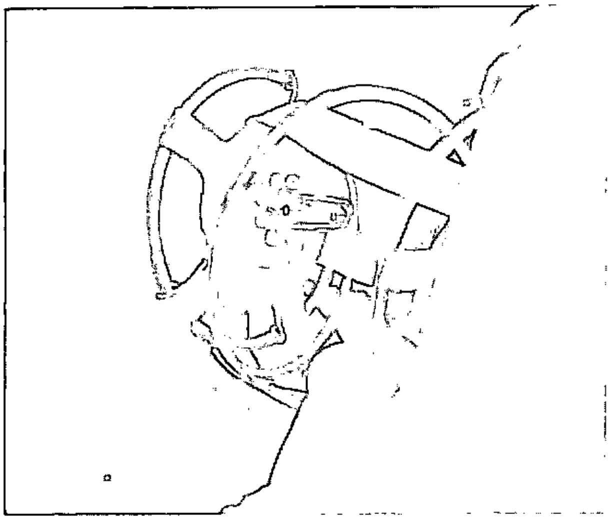
4. A new jaw mechanism was installed and sheet metal work around the jaws was performed by a Convair team to minimize possibility of faulty "D" ring release.

A period of 40 flights covering 50 hours of flying with the above modifications, resulted in no reported drag chute failure.

☐ *Yaw Rate Gyro:* The yaw-rate gyro consistently failed to function properly. The primary deficiency is that it does not erect after making a turning or banking maneuver. These gyros are hermetically sealed and could not be disassembled for inspection. Several gyros were sent to the contractor (Hughes Aircraft Company) for evaluation of the trouble. To date no report of findings has been furnished to this Center.

☐ *Elevon Trim Actuators:* The elevon trim actuators are not functionally acceptable. The actuators either operate slower than desired or are unstable in operation. A total of 10 malfunctions of these actuators occurred during the program.

Difficulties encountered in the mechanism on one aircraft were isolated to a faulty snap ring in the clutch assembly. Failure of this snap ring permitted the clutch to spring apart, breaking the power links between the motor and the actuator.



☐ fuel and oil systems

Fuel System: All fuel in the F-102A is carried in the wings, outboard of the fuselage in a left and right fuel tank system, each of which includes three pressurized integral tanks. Except for a common single point refuel fitting and an interconnected totalizing and low level warning system, each wing system is completely independent of the other. Fuel from both systems is delivered to the engine by a common supply line; however, either side is capable of supplying engine and afterburner requirements with the other side inoperative or shut off.

An engine driven pump draws the fuel from both wing systems. These systems are connected by the fuel joiner valve. The fuel flow equalizer is located in this valve and is designed to draw equal quantities of fuel from both wing systems for balance purposes.

There is a rubber diaphragm in the equalizer which actually senses the difference in fuel flow from each system and compensates accordingly. The equalizer nozzles are located on either side of this diaphragm.

During the Phase VI tests nine failures of the fuel flow equalizer were directly attributed to this rubber diaphragm. Fuel surges in the equalizer caused the diaphragm to contact the equalizer nozzles, eventually chafing through the center and destroying the equalizing function of the diaphragm. This caused uneven feeding between the right and left wing tanks with variance in available fuel per wing as high as 2800 pounds.

☐ *Fuel Quantity Indicating System:* A fuel quantity gage normally totalizes the contents of the wing systems as selected by the quantity gage switch. With the switch in "TOTAL" and the fuel in one wing unavailable to the engine, the indication of total fuel remaining will not be an indication of usable fuel. With the switch in "3L" or "3R" the indicator reads the amount of fuel in the inboard tank.

A low level warning switch, installed in each aft inboard tank, is set to energize a fuel tank low level warning light circuit whenever either No. "3L" or No. "3R" tank drops its level to a predetermined point. The lights are set to operate when 88 gallons of usable fuel remain in each side. This will permit about 10 minutes of cruising at 35,000 feet and a letdown at 200 knots IAS from there to sea level for a landing.

This low level warning system has malfunctioned several times. On some occasions, a low level warning light has illuminated well before it should have, and on one occasion a warning light failed to illuminate at all. This latter failure indirectly caused a flameout and a night dead stick landing. The cause of the flameout was fuel starvation which was attributed to failure of the right wing low level warning system. The pilot had turned off the left wing tank booster pumps after receiving a left wing fuel low level warning light. Since he had received no right tank low level warning, he assumed that he was receiving uneven fuel feeding (which occurred many times due to ruptured fuel flow equalizer diaphragms) and had over 88 gallons of fuel in the right tanks. Those tanks were in reality very nearly empty. On the pitch-out to the left from the initial approach for landing fuel flow was interrupted, causing the flame-out. The pilot managed to make a safe landing on a different runway under rather adverse conditions. Residual fuel was drained and found to be 93 gallons, 90 of which was in the left No. 3 tank.

☐ *Oil System:* The oil system is relatively simple. From the tank the oil passes through a gear pressure pump, a strainer, and then divides, part going through a restrictor valve to the Sundstrand constant speed gear box and accessory drives, the other part going to the four engine main bearings. From the bearings the oil passes through a scavenge pump to the air oil cooler and, if over 185 degrees F, the oil goes through the fuel-oil cooler and then back to the oil tank.

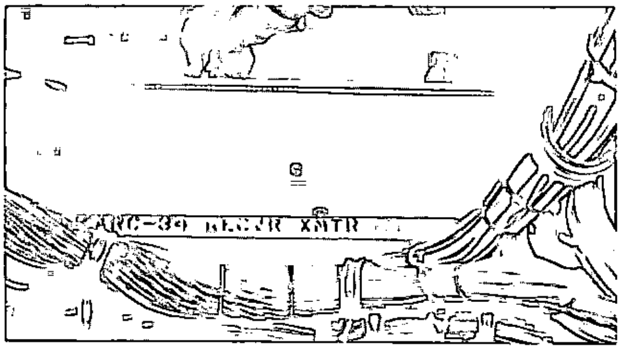
This system performed satisfactorily except for one item. Five ruptures have occurred in the oil cooler air duct assembly. These failures are attributed to insufficient thickness of the ducting. In every case it is believed the duct ruptured when a sudden compressor surge caused excess air pressure in the duct.

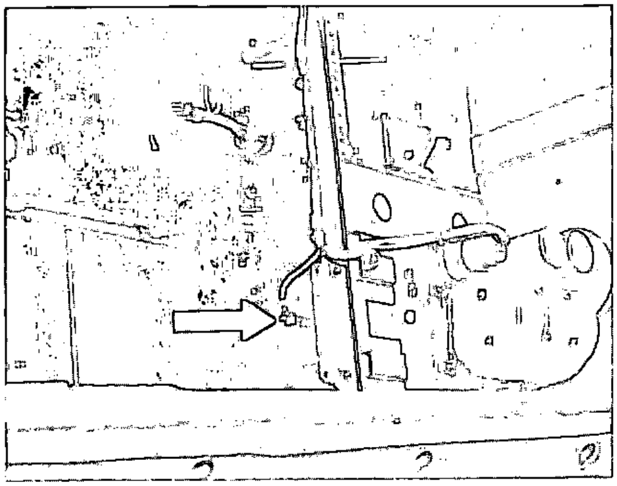
☐ communications system

The communications system is acceptable in most respects; however, some problems exist in component location.

☐ *AN/ARC-34 Receiver-Transmitter:* The inaccessibility of the ARC-34 receiver-transmitter resulted in excessive maintainence man-hours for removing and installing this equipment. In addition there is increased possibility of damage to the equipment during these operations. A hot air line is routed across the top of the ARC-34 unit, clearing the unit by approximately 1 inch. Removing the unit requires lifting it from the rack approximately 1½ inches and pulling it forward.

The microphone and earphone jack sockets are located on the forward end of the receiver-transmitter which precludes easy access because of inadequate clearance between the set and the bulkhead.





☐ *AIC-10 Intercom Amplifier:* The hot lead to the noise filter for the AIC-10 intercom amplifier, located on a shelf in the upper section of the mid electronics compartment, has no protective covering on the exposed terminal post. On several occasions maintenance personnel working in this compartment suffered burns when accidentally contacting this terminal post. A local modification to correct this condition consisted of placing a nipple-type cover over the exposed terminal.

☐ electrical power supply system

The engine accessory drive gear box provides power for the generation of electrical power. A variable input speed, constant out-put speed unit is installed between the engine and an a-c generator and operates at a speed which varies as the engine speed varies. The 120/208 volt, 400 cycle, 3-phase a-c generator is rated at 30 KVA. The 30 volt d-c generator is rated at 200 amperes.

Emergency electrical power is supplied by 120/208 volt, 3-phase, 400 cycle, 1-KVA a-c generator and a 24 volt, 24 ampere-hour battery. The emergency a-c generator is driven by a hydraulic motor which is powered from the secondary hydraulic system.

Power is normally distributed to the essential and non-essential bus systems. The alternator provides 200 volt, 3-phase, 115 volt single phase, and through transformers, 115 volt 3-phase and 26 volt single phase power. When emergency power is selected the non-essential buses are de-energized. A master switch is provided to de-energize all buses regardless of the position of any other switch. An under-speed switch on the constant speed unit de-energizes the a-c generator if the frequency of the normal power supply falls below 350 cycles.

One deficiency in this system concerns the lighting of the instrument panel. Only the flight instrument group in the center of the instrument panel is illuminated through use of the 28 volt d-c essential power. All other interior lighting (with the exception of the thunderstorm light which uses 28 volt d-c non-essential power) utilizes power from the 115 volt essential bus. In event of failure of the main a-c generator, the hydraulically driven emergency a-c generator must be selected manually to provide power to the essential buses. No automatic change-over feature is provided. Since the electrical control panel is illuminated by a-c power, it is obvious that failure under night conditions will result in loss of illumination of the panel which must be utilized in restoring that power. It would be advantageous for purposes of efficient restoration of a-c power and complete cockpit lighting if the electrical control panel were illuminated by use of 28 volt d-c essential power.

Following a severe rainstorm at Holloman Air Force Base, during the AFFTC firing program, an electrical fire occurred on ground operation of aircraft S/N 53-1803. An external power supply had been operating radar equipment for approximately 6 hours when smoke was observed emitting from the cockpit. External power was immediately disconnected and the fire extinguished itself. Investigation disclosed that the fire had originated in the connector assembly panel insulation which is located on the cockpit floor beneath the right-hand console. The connector and adjacent wiring were severely charred and the console fabric dust cover was burned beyond use. The wiring and damaged parts required replacement. Evidence of water remained around the connector. This water entered the cockpit during the rainstorm, and though the canopy was completely closed and caused a short circuit and subsequent fire. The smoke became so dense that had the aircraft been in flight the pilot would have been unable to see the instrument panel and probably would have been forced to abandon the aircraft.

☐ *Fuel Booster Pump Wiring:* Early in the Phase VI program considerable aircraft out-of-commission time was caused by what appeared initially to be booster pump malfunction. The 15 ampere fuses provided within the circuits for these pumps continually failed as the pumps were started. The cause of these malfunctions was traced to the circuit wiring. Contractor assistance was utilized in the installation of larger size wire (#12) and larger capacity fuses (20 amps) as a precaution against further overloading. This modification alleviated the problem for the remainder of the program.

functional analysis

ARMAMENT AND FIRE CONTROL SYSTEM

During the Phase VI program the fire control systems were maintained in an in-commission status as much as was practicable. Most Phase VI flights were accomplished with the fire control system in operation. Utilizing an ADC GCI station, actual intercepts were made on targets of opportunity within the local flying area. These targets ranged from B-52 to F-86D type aircraft. On one occasion early in the program a successful intercept of an unidentified aircraft was made at the request of the Air Defense Command. It was possible to detect a heavy bomber of the B-52 type at an average range of 15.4 miles. Average ranges for detection of medium bomber (B-47) and fighter (F-86D) aircraft were 14.3 miles and 9.1 miles, respectively. A total of 472.3 hours of airborne operation time was accumulated on the fire control system of the five test aircraft. In addition, 1116.4 hours of ground operating time were logged.

Detection and lock-on ranges for a representative group of aircraft are indicated in the following table.

TABLE No. 7

Type Target	No. of Contacts	Maximum Detection Range (Miles)	Average Detection Range (Miles)	Maximum Lock-on Range (Miles)	Average Lock-on Range (Miles)
B-52	13	25	15.4	15	9.7
B-36	12	15	11.8	14	9.9
B-47	15	22	14.3	14	9.6
B-45	4	7	7	5	4.3
C-133	2	13	10.5	5	4.5
C-97	1	15	15	14	14
C-54	2	12	11	11	9.5
F-86D	16	13	9.1	10	6.8
F-86F	1	7	7	5	5
TF-86	1	10	10	8	8
F-89	8	14	9.3	12	7.5
F-94	2	14	13	9	8
F-100	10	17	9.1	12	6.9
F-101	2	15	12.5	12	9.5
F-102	41	22	11	15	8.6
T-33	16	13	7.2	12	5.7
F3D	3	21	18.3	14	13
F4D	3	18	10.3	13	8.3
F8F	1	6	6	5	5
F9F	1	12	12	9	9
R4D	2	10	10	7	7

☐ missile firing program

Following the weapon system concept in Phase VI flight testing, a missile firing program was conducted on the F-102A/MG-10/GAR-1 system. Two of the five F-102A Phase VI aircraft, S/Nos 53-1801 and 53-1803, were flown to Holloman Air Development Center on 16 June 1956 to conduct this program. During the period 16 June to 16 August 1956, 49 flights totaling 42:45 flying hours, were flown on the two aircraft, and on 24 actual firing missions a total of 63 GAR-1 missiles were fired. Targets for the missiles included parachutes, balloons, Q-2 drones, and QB-17 drones.

The firing program was divided into three phases with each phase having a specific objective under the overall objective to investigate and prove the capability of the F-102A/MG-10/GAR-1 weapon system while gathering data on ground maintenance and airborne problems, failures and equipment deficiencies.

☐ *Phase 1:* The first phase, system verification and compatability, was designed to check the system and prove that all components were operating and that there were no major deficiencies in the system. Firings were to be conducted under the most ideal conditions at moderate speeds and altitudes. Generally, single or two missile salvo firings, one missile loaded forward and one aft, were used against parachute targets. During this phase any major deficiencies in system wiring or components were located and corrected.

☐ *Phase 2:* The second phase, system demonstration through its full envelope of performance, was conducted within the limits of available targets and Holloman range safety. During this phase the missiles were to be fired in salvos of three and six at subsonic and supersonic speeds and up to altitudes of 50,000 feet. Unfortunately, due to the limited amount of time and supply of missiles, and lack of OMNI range at HADC, no missions could be scheduled for IFR conditions or against targets equipped with ECM equipment.

☐ *Phase 3:* The third phase, investigation of limits of system performance, had the objective of determining the systems capability and restrictions at maximum altitudes of the aircraft. This was considered a requirement of the program since, according to the specification of the missile, its performance limit is 50,000 feet, while the aircraft is capable of considerably higher altitudes and would most likely be operating at those altitudes. No attempt was made to schedule an investigation of the minimum altitude limitations of the system because of the limited time and number of missiles available.

Aircraft maintenance personnel were provided by the Flight Test Center and were assisted by one field technical representative of Convair, and 6 airmen from the Air Defense Command receiving on-the-job training with the project. MG-3 radar maintenance was performed by 3 airmen from the AFFTC, 2 from HADC, and 1 from ADC, and 2 field technical representatives of Hughes Aircraft Company. Airmen personnel of the AFFTC at HADC, with the assistance of 1 Hughes Aircraft Company technical representative provided the entire missile check-out and handling capability.

☐ *Fire Control System:* The MG-3 radar of the MG-10 fire control system consists of four sub-systems: the radar and computing, the missile auxiliary, the power supply, and the cockpit sub-system.

The radar and computing sub-system provides the means for automatic radar searching and tracking, interrogation of radar ground beacons, and radar ground mapping. The ground mapping feature is used so seldom in this area that it was disabled and an additional method of using the radar in the search phase (30 mile short pulse) was used to replace it. During search the radar has a range of 30 miles and is stabilized so that normal roll and pitch of the aircraft will not change the area of search. Manual "lock-on" and automatic tracking in range and azimuth may be achieved within 15 miles of the target.

The missile auxiliary sub-system is used in conjunction with the GAR-1 and GAR-2 missiles normally carried in this aircraft. For both types of missiles the auxiliary sub-system performed the following functions:

1. Converts the radar antenna pointing data so that the missiles may be servoed on target prior to launch.

2. Computes and adjusts the missile control surface sensitivity for optimum performance for the particular air density and range rate at which the missile is to be launched.

3. Supplies external power to the missiles.

4. Performs the various switching operations necessary to permit transfer from external to missile power and warhead arming.

5. Servos the GAR-1 missile range gate into coincidence with the radar range gate.

6. Servos the GAR-1 missile receiver and radar transmitter into frequency correspondence.

7. Shifts the radar pulse repetition rate to that required by the GAR-1 missile.

8. Computes the optimum GAR-1 relative missile travel and computes and sets the optimum GAR-1 missile receiver sensitivity.

After all the required operations in the missile preparation sequence have been performed on signal from the computer (depending on the type of missiles loaded), the firing signal may fire the missiles.

The power supply consists of a motor generator unit which has a 400 cycle a-c motor and three d-c generators on a common shaft and their associated voltage regulators and protective circuitry.

The cockpit sub-system consists of the radar scope, a hand control unit (mounted on the control stick) and the system control panels. The radar scope displays the information from the radar and computing sub-system. The hand control unit provides a means of controlling antenna motion and range gate position in order to "spotlight" the target and achieve "lock-on." The system controls provide the means of turning on the system, selecting the mode of attack, selecting the number of missiles fired, controlling the intensity of the display, controlling the receiver IF gain, controlling the antenna scan pattern (in search), and erecting the artificial horizon.

A normal missile attack proceeds through the following phases:

☐ *Search:* The pilot can detect a target at ranges up to 30 miles and fly the aircraft in such a manner to reduce the range to that required for lock-on.

☐ *Automatic Track:* The pilot, having reduced the range to 15 miles and "spotlighted" the target using his hand control, becomes locked on. Steering information is presented on his scope which allows him to fly a lead collision course to the target. After lock-on is achieved, the fire control system computer determines the following timing periods:

☐ *"A" time;* the transmitter is tuned to missile frequency and the system starts preparing the missiles for launch.

☐ *"B" time;* the transfer is made from external to missile power. This normally occurs 15 seconds prior to firing.

☐ *"D" time;* the doors are opened, the missile launchers extended, and the missile parameters set.

☐ *"E" time;* the firing signal, sent to the missile intervalometer, fires the missiles in the sequence which has been selected by the pilot.

☐ *Missile:* The GAR-1 missile is an air-to-air semi-active radar guided missile, which is propelled at supersonic speeds to intercept and destroy aerial targets upon impact. The GAR-1 is launched on a collision course with a computed lead angle, and is provided with a guidance system designed to overcome effects of launching errors and target maneuvers. The missile passes through five phases in completing a tactical mission: pre-launch, launch, boost, guidance, and impact. An auxiliary phase, which terminates in self-destruction, is provided in the event of a miss. The pre-launch phase begins with the energizing of the fire control system and ends 2 seconds before transmission of the firing signal. During this phase external power is supplied to the missile, the missile is transferred to internal power, the missile antenna is trained on the target, the missile range gate is set, and automatic frequency control begins.

The launch phase begins 2 seconds prior to the firing signal and ends when the rocket motor ignites and the missile is released. During this phase, parameter circuits are adjusted and the hydraulic power supply and rocket motor igniter of each selected missile are activated by the firing signal.

The boost phase begins with firing of the rocket motor igniter and ends when the rocket motor has burned out. During this phase the missile accelerates at a constant rate, radar tracking of the target is maintained, steering signals are shorted out, and roll rate of the missile is controlled.

The guidance phase begins at rocket motor burnout and ends when the missile hits the target. During this phase the operation of the missile guidance and control system alters the straight-line flight of the missile to correct for launching errors, missile slowdown, and target maneuvers. The interceptor radar continues to illuminate the target throughout the guidance phase.

Impact may occur anywhere from 5 to 15 seconds after launch, depending upon attack conditions. Upon impact, the warhead is triggered by breaking or grounding any one of four trigger wires located in the leading edge of each stabilizer. A timing mechanism detonates the warhead and destroys the missile 22 seconds after launch in the event the missile does not hit the target.

☐ preparations

Fire Control System: During operation at HADC the following procedures were normally followed prior to flight.

☐ *Missile Range Gate Harmonization:* In this procedure time delays inherent in the wiring between the missile and the range gate servo unit and the associated wiring internal to the missile were cancelled out by an adjustment made on the servo unit. This adjustment was necessary for each individual missile to be fired and is a function of the launcher on which the missile is installed. Harmonization was performed in the following manner.

a. Prepare fire control system for normal ground operation, turn on and allow system to "Time In".

b. Install missile on rail to be harmonized and connect power and test cart to missile.

c. Connect signal interrupter switch box to appropriate plugs on range servo unit (P/N 463106-120) or "106" box, for rail to be harmonized.

d. Lock radar on to corner reflector placed from 3,000 to 6,000 yards from the aircraft.

e. Using missile auxiliary test set or special switch box (locally manufactured) artifically place system in "B" time (switch box plugs into J8601 on computer mode plug).

f. Using a vacuum tube voltmeter, monitor voltage from test point 1 on "106" box for rail to be harmonized while adjusting delay line on this box. Adjust delay line so that minimum voltage is obtained with the switch on the interrupter box in both positions. This completes the harmonization for that particular rail with that particular missile installed.

☐ *Pre-flight Inspection:* A normal radar pre-flight as outlined in Technical Order IF-102A-2-12, was followed and all discrepancies discovered were corrected using normal test equipment and procedures.

☐ *Test Equipment:* All items of test equipment used were standard Air Force items with the following exceptions.

a. Missile power and test cart used to supply power to missiles being harmonized and to monitor certain voltages in the missile.

b. Interrupter switch box used to break range voltage circuit during harmonization adjustments.

c. Computer timing signal simulator used to supply synthetic computer timing signals to the system during harmonization.

It should be noted that the first item of equipment is necessary for range gate harmonization with the live missile only. The other two items or their equivalent will be necessary for any harmonization no matter how performed.

The modifications listed below were already incorporated in the equipment upon delivery to the AFFTC.

Table No. 8

ECP	Unit Affected	Mod. No.
HUG-(E-9/MG-3)-0022	423020-140	MD-2
HUG-(E-9/MG-3)-0023	423057-130	MD-2
HUG-(E-9/MG-3)-0052	423008-120	MD-1
HUG-(E-9/MG-3)-0094	479086-140 479096-130	MD-1 MD-1
HUG-(E-9/MG-3)-0110	423057-130 423032-121 423004-130	MD-1 MD-1 MD-1
HUG-(E-9/MG-3)-0117	479023-130	MD-1
HUG-(E-9/MG-3)-120	423106-110 423057-130 423008-120	
HUG-(E-9/MG-3)-0059	423033-130	MD-1
HUG-(E-9/MG-3)-0065	423107	MD-2

The following units were modified by Air Force personnel at the AFFTC.

Table No. 9

Unit Affected	Description	Purpose
403004-100 Interconnecting Unit Missile	Change values of certain capacitors per ECA 20233	To eliminate undesirable transients
463057-110 Relay Assembly Armament Control	Delete wire from relay K-34 per HAC instruction	To prevent abort if firing signal occurs during launcher extension
463057-110 Relay Assembly Armament Control	Modify wiring per ECP 0210	To prevent undesired missile parameter settings
463106-X120 Range Gate Servo Unit	Modify aircraft wiring per TVA 46557	To adapt system to modified units
463086-110 Computer Mode Plug	Modify wiring per ECP (E-9, MG-10)-0201	To advance firing time
463003-110 Electrical Synchronizer	Modify in accordance with HAC instructions	To reduce Video blooming
463255-100 Control Radar Set	Rewire Unit	To give 30 mile short pulse capability
463083-110 Flight Control Group	Reverse leads to R-10A	To reverse range gate motion with hand control action

☐ *Missile:* Sixty-three GAR-1 missiles were delivered to Holloman Air Force Base for the F-102A Phase VI test. All of the missiles were checked out on the MA-1 (DSP) field check-out console by Air Force personnel. In addition, one telemetered missile and one oscillograph missile were repeatedly checked out during the first portion of the test program.

One modification was made to the missiles. It consisted of modifying Lot #4 and #5 missiles to a boattail configuration to accommodate three larger flares that were originally installed on each missile to aid tracking of the missile by ground cameras. Lot #8 and higher missiles were delivered with no provision for flares, and as far as possible, the standard boattails were replaced with the large flare holder type. Shortage of flare holder boattails prevented equipping all missiles with flares.

☐ *Check-out Console Operation:* Several modifications were made to the MA-1 (DSP) check-out console and associated equipment. The first of these was the addition of two resistors and removal of one switch connection in the multimeter circuits in accordance with Hughes Engineering Order 75690. The repetition rate generator "B" was modified as prescribed in Hughes Engineering Orders 75708 and 75710, which consisted of eliminating one connection. On the maintenance bench, a Hubbel power plug was replaced by an R & S plug strictly because of the availability of the latter plug. On the console only very minor modifications were made; one of these being the repositioning of the "T" return line connector to allow easier access for filling with oil.

A total of 29 repairs or part replacements had to be made on the console and associated equipment. Twenty-one of the repairs or replacements were of minor significance, being the replacement of tubes, fuses, relays, or repairing poorly soldered joints or broken contact pins.

Of greater importance was the replacement of three RF attenuators in different units, replacement of one inverter, replacement af an "O" ring to stop excessive nitrogen pressure leakage and replacement of the automatic unloading valve. The most time consuming replacement was that of the umbilical harness which had shorted and subsequently burned out. Total "down time" on the console was approximately 1½ days.

Operation of the check-out console during this program and on earlier F-89H Phase VI program indicates that maximum efficiency and safety is attained when the operation is performed by a 6 man crew. The NCO in charge should be seven level airman (31170E). One airman should record history and data at all times. This can be a three level airman (31130E). A five level airman (31150E) with a knowledge of armament should be at "A" station. "A" station is the station at the console where continuity checks are made on the igniter, fuse, battery, and trigger circuitry. A five level airman (31150E) and a three level airman (31130E) should be at "B" station. "B" station is where guidance and control checks are made on the missile. A three level airman (31130E) or a five level airman (31150E) should be at "C" station. At "C" station, the missile is secured and inspected before its release for flight. The "A" and "C" station men actually work together at both ends of the console.

☐ test conditions

Flight Conditions: All missions of the F-102A Phase VI guided GAR-1 missile firing program were conducted at the Holloman Air Development Center over the Holloman Air Force Range. Mission control was exercised by HADC personnel from the flight requests of the Air Force Flight Test Center project officer at Holloman Air Development Center via the Air Force Missile Test Branch at HADC. Ground control interception was performed by the use of modified SCR 584 radar (each aircraft carried a beacon) with the horizontal position and altitude information recorded on a plotting board. The physical location of the control point for all missions was at "King I" and all vectoring of launch aircraft was perfomed by officers assigned to the Phase VI program. During the early part of the firing program, including all of the first phase, a maximum effort was made to run all missions under perfect weather conditions in order to facilitate maximum tracking and miss distance data from the Holloman photo theodolite range. However, this condition was impossible to maintain throughout the program, since, due to system development delays, it was necessary to conduct the program at HADC during the rainy season where very often theodolite coverage is impossible. To restrict flights to clear weather when theodolite coverage was possible would have extended the program by at least 2 months.

All missions were conducted as nearly to a tactical condition as possible, still considering the necessity of range safety and of covering the operational range of the weapon system. Generally, the F-102A aircraft were not sent aloft until the target reached the desired altitude and then firing passes were attempted as soon as the interceptor reached the same approximate altitude.

During the last 10 seconds of all firing passes an attempt was made to restrict the roll angle to 10 degrees or less. The equation solved by the delta "H" computer solves for a difference in altitude between the target and the interceptor. This term is used in the air density servo and this air density output is used in computing the missile firing range and in parameter setting functions. The difficulty originates in the phase of the input voltage to the delta "H" controller (247 units), causing the term to be subtracted instead of being added. Up to 10 degrees roll the error in the launch range is approximately 6 percent but as roll increases above 10 degrees the error increases rapidly.

☐ *Instrumentation:* The instrumentation installed in both F-102A aircraft consisted of 2 N-9 strike cameras mounted one in each elevon actuator fairing, two launch cameras viewing the missile bay area mounted one in each elevon actuator fairing, and one scope camera viewing the pilot's indicator through a 2-way mirror and mounted above the scope. The instrumentation was installed to provide additional launch and miss distance information that was not available from the theodolite data and to permanently record any steering errors.

To assist in determining deficiencies and failures in the system, several telemetry equipped missiles were borrowed from Hughes Aircraft Company. These missiles were not fired, but were used to gain information on the preparation of the missile during a firing run. These missiles telemetered the following information:

1. Receiver automatic gain control (AGC).
2. Angular position of antenna head with reference to pitch and yaw axis.
3. All power supply voltages.

Also, a missile simulator was borrowed from Hughes Aircraft Company. This missile simulator or test missile was used whenever possible and provided an oscillograph record of everything that happened within the missiles during a firing pass. By having the launch aircraft continue on past the path of the target, this test missile would indicate any transients in the radar system, and, if the missiles were fired, it would also indicate whether those missiles were able to maintain "lock-on" on the target and guide to it.

☐ *Maintenance:* All aircraft and radar maintenance was conducted on the ramp at HADC in front of the Butler Hangar which housed the MG-3 bench set and the missile check-out console. There was no cover available for the crews while working on the aircraft; therefore, whenever time permitted, the radar maintenance was performed in the morning or after dark. In this way, more work could be performed by the crews without the extremes of fatigue. The missile loading carts were not available so all missiles were loaded by hand from carts similar to storage racks.

During this firing program a number of system failures caused aborted missions. These failures and their frequency are outlined as follows:

Failure	Function Affected	No. of Occurances
Failure of 305 box	Missile malfunction	3
Failure of Ledex Units, computer mode plug unit	Flight abort	4
Failure of nose wheel micro switch	Flight abort	2
Failure of interlock switches	Flight abort	5
Failure of radar pressurization	Flight abort	2
Failure of afterburner to relight	Flight abort	1
Radar adjustment (lock-on sensitivity)	Flight abort	2
Failure of missile ignitor	Missile malfunction	1
Failure of Sundstrand drive gear box	Flight abort	1

☐ *Targets:* Four different type targets were used during the firing program: parachutes, balloons, Q-2A drones, and QB-17 drones. The parachute targets consisted of a 48 foot flat chute made of a metallic impregnated material (developed by Irving Air Chute Company) supporting a 35 pound weight. No corner reflectors were required since the entire chute was radar reflective. When descending, the chute produced a target that was approximately 44 feet long and from 6 to 8 feet deep. These targets were carried aloft in containers that were similar to a 500 pound bomb, by F-86H support aircraft stationed at Holloman Air Force Base. The parachute provided a scintillating target with an unknown radar center.

The balloon targets were also made of a metallic impregnated material that reflected radar energy. They had a 40 feet diameter bag that was nearly spherical when above 50,000 feet. The balloons could be preset for any desired altitude and were launched by personnel of the HADC balloon squadron. The launching sites were moved about so that when the balloon reached the desired altitude it would be within the desired limits of the theodolite range. At altitudes above 50,000 feet, the spherical balloon made an excellent radar target with a reflective center at the center of the balloon with very little scintillation.

The Q-2A (Ryan Firebee) drone is a turbojet propelled, radio controlled drone. Because of the small size of this target, its reflectivity was insufficient to obtain contact and lock-on ranges great enough without some reflectivity augmentation. Two different corner reflectors were obtained to provide the augmentation; one was a wing tip mounted reflector and the other was a reflector mounted in the tail cone aft of the parachute compartment. Since the corner reflector is monostatic, its installation as a reflectivity augmentor precluded any beam attack on the Q-2 drone. No usable bistatic reflectors were available at the time of the program.

The QB-17 drones were modified B-17's with radio control equipment installed. Unfortunately, due to previous losses, no firing error indicator (FEI) equipment was available for installation on the QB-17's provided for this project. Because of the very short supply of QB-17 drones, explosive warheads were clipped on the GAR-1 missiles which were fired at the QB-17 so that in the event the missiles scored a hit the possibility of destruction of the drone would be lessened substantially.

The F-102A Phase VI project was allocated, for the first quarter of fiscal year 1957, 7 Q-2A flights with 3 kills permitted, and 3 QB-17 flights with no kills and only one operational loss permitted. Midway through the program the AFFTC requested, and was granted, 5 additional Q-2A flights and 3 additional QB-17 flights.

☐ test results

Project Operation: A total of 49 flights were flown on F-102A aircraft S/N 53-1801 and 53-1803 during the period from 16 June to 16 August. During 24 of these flights a total of 63 missiles were fired in various attack patterns, altitudes, speeds, and against various targets. As shown in Chart No. 4 an attempt was made to cover the entire operational range of the aircraft. However, a maximum speed (1.2 Mach number) firing attack was not obtained. On a firing pass at 38,000 feet against a Q-2 target, F-102A S/N 1801 reached that speed, however a malfunction in the armament interlock circuits prevented the GAR missiles from firing. No low level attacks, below 10,000 feet, were made because the low elevation angles make it very difficult to obtain theodolite coverage. The parachute targets were used solely for the first phase of the test program and all attacks were made at altitudes and speeds which would permit maximum theodolite and photographic chase (F-94C aircraft) coverage. Only one hit was scored on this type target.

The Q-2A targets were used (except for two flights) entirely for the second phase of the program. Because of engine and control problems with this drone, altitudes were limited to approximately 40,000 feet. The speed of the Q-2A as operated for this test, was 270 knots IAS or Mach 0.76, whichever was the lower. In firing against the Q-2A, a racetrack pattern was used with the F-102A aircraft following the Q-2A at approximately 7 miles range until the final turn on a firing run when the F-102A's were accelerated to desired firing speed. For the firings above Mach 1.0 a slight diving attack was used to bring the F-102A aircraft up to speed within a 3 minute time duration. Of the total of 29 missiles fired, estimated miss distance is available on 22. It is estimated that only 9 came within 20 feet of the target. This is considered to be substandard accuracy against a corner reflector target.

The QB-17, as used for the second phase of the program, was the most realistic target in size and form available. It was flown in a racetrack pattern and attack passes were made at angles of 70 to 90 degrees off the tail. A total of 13 missiles were launched at this target and only 1 hit was scored. From the estimated miss distances only 5 of the total 13 were within 20 feet of the target. One again, this is a very low percentage of hits and near misses.

The balloon targets were used entirely for the high altitude performance phase of the operation. Attacks against the balloons were made at co-altitudes or in slight climbs starting approximately 2,000 feet lower than the balloon at a range of 10 miles. Reasonably good results were obtained with 3 balloons being destroyed on 5 firing passes. A total of 12 missiles were fired in groups of two and three. The fact that 2 of the balloons were destroyed during 3 firing passes at approximately 55,000 feet when the maximum design altitude for the system firing GAR-1 missile is 50,000 feet was considered of great importance. On the 2 passes where hits were scored, the steering dot was centered while on the third a considerable elevation error was present. On this third pass, the missiles were observed to pass below the target still climbing, and it was evident that at the extreme altitude the GAR-1 missile was unable to achieve a sufficient load factor to make the required corrections.

It should be noted that the majority of miss distances presented in this report are from visual estimates by the launching aircraft pilot, the chase aircraft pilot, and the phototheodolite operators. The complete and detailed data on each flight is given in Table No. 15.

☐ *Missile Accuracy:* The accuracy of the missile in conjunction with the MG-3 radar appears to be less than predicted by the contractor. It is estimated from the limited Phase VI firing program that the probability of a hit when attempting to fire a 6-missile salvo in a beam attack against a bomber-type target 113 feet in length, is 0.72. The probability of a hit when firing a single missile is 0.19; with a 3-missile salvo the probability is 0.47. Considering the in-flight abort rate the overall probability of a hit, utilizing 6 missile salvo starting at take-off, is 0.44.

CHART NO. 4

F 102A GAR I

Launch Conditions

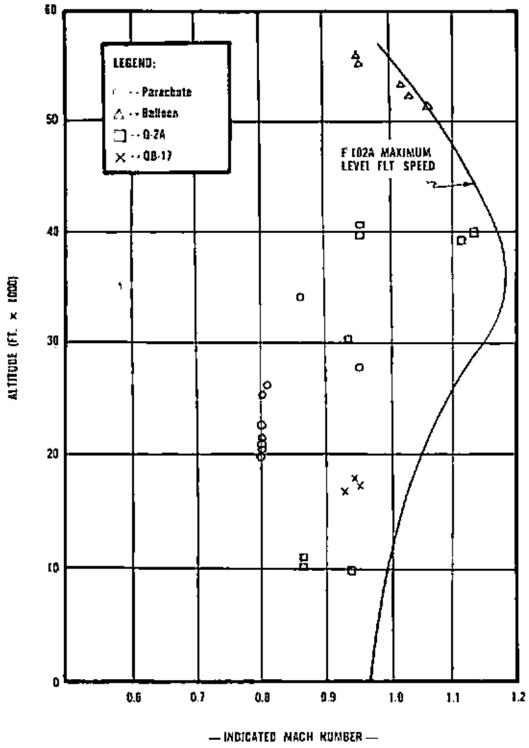


CHART NO. 5

- MISSILE FIRING PROGRESS -

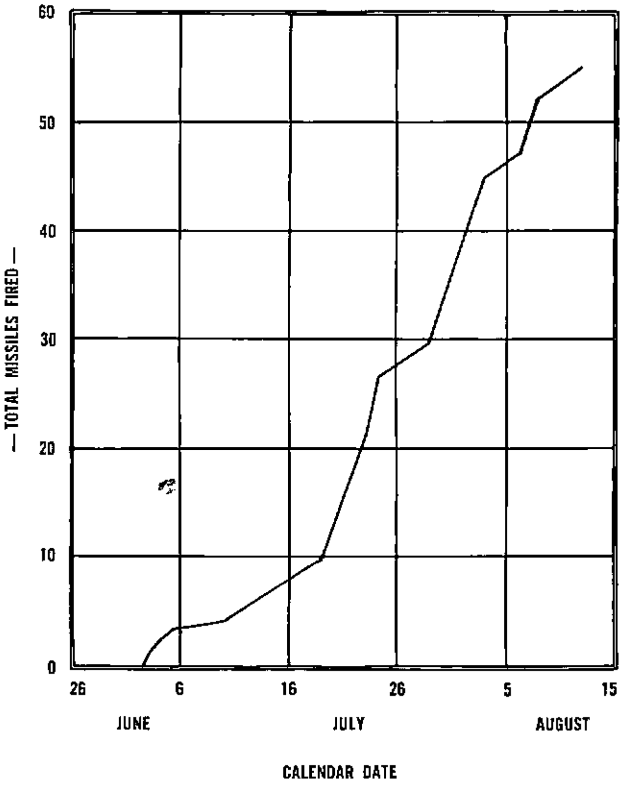


Table 11 — Gar-I Firing

	1st Phase	2nd Phase	3rd Phase
Missions Flown	18	24	7
Missions Aborted (After T.O.)	0	3	3
Airframe Failure	6	2	1
Fire Control System Failure	1	9	0
Other	6	0	0
Single Missile Salvo	2	1	3
Two Missile Salvo	1	8	1
Three Missile Salvo	0	1	0
Four Missile Salvo	0	1	0
Five Missile Salvo	0	1	0
Six Missile Salvo	—	—	—
Total Missiles Fired	13	41	9
Total Missiles Guided	8	27	8
Guidance Unknown	0	7	1
Target Hits	1	2	2

Table 12 – Target Miss Distance Distribution

Miss Distance	Parachute	Balloon	Q-2A	QB-17
Hit	1	3	0	1
0 - 10 Ft	0	0	7	0
10 - 20 Ft	1	1	2	4
20 - 50 Ft	3	1	7	1
50 - 100 Ft	0	2	1	0
100 - 200 Ft	0	0	1	5
Over 200 Ft	3	0	4	2
Unknown	1	5	7	0

Table 13 – Missile Salvo Distribution

Type Salvo	Parachute	Balloon	Q-2A	QB-17
Salvo of 1	5	0	1	0
Salvo of 2	2	3	1	0
Salvo of 3	0	2	7	1
Salvo of 4	0	0	0	1
Salvo of 5	0	0	1	0
Salvo of 6	0	0	0	1

Table 14 – Hits Scored

Type Salvo	Type Target	Altitude (Ft)
3 Missile	Balloon	51,000
3 Missile	Balloon	53,000
2 Missile	Balloon	55,000
2 Missile	Parachute	35,000
6 Missile	QB-17	18,000

TABLE No. 15 - FLIGHT LOG

Flight No.	Date	TEST OBJECTIVE	MSL No.	Pos No.	Target Alt/Ft	Launch Alt/Ft	Lock-On Range	Launch Range	Launch Speed	Guidance	Miss Dist/Ft	REMARKS
1	28 June	Check radar and missile auxiliary function	2345 2501	6 4	Parachute 20,000		7.5 Miles		0.82 M			Captive flight — could not maintain lock-on
2	30 June	Check radar and missile auxiliary function	2345	6	F-102A 25,000		4 Miles		0.85 M			Captive flight — Operation apparently proper
3	2 July	Determine proper functioning of system	2319 2362	4 6	Parachute 25,500	26000	8 Miles	4170* yds	0.82 M	No Cap		After flight 8 reason for nonguidance was found to be intermittent failure of 305 box
4	3 July	Determine proper functioning of system	2342 2345	4 6	Parachute 21600	21900	10.5 Miles	3778* yds	0.82 M	No Cap		Same as for flight 3
5	6 July	Determine proper functioning of system	2358 0188	4 6	Parachute 22000	22200	9.25 Miles	3200 yds	0.81 M	Yes TM	20-50	
6	10 July	Determine proper functioning of system	2324 0188	5 4	Parachute 20000		9.5 Miles		0.82 M			System received no signals after "A" signal because of faulty Ledex relay
7	10 July	Determine proper functioning of system	2324 0188	5 4	Parachute 22,000		11.5 Miles		0.82 M			Same as for flight 6
8	11 July	Determine proper functioning of system	2340 0188	5 4	Parachute 20,600	20600	11 Miles	3670 yds	0.81 M	No TM		Intermittent failure of 305 box caused non-guidance
9	12 July	Determine proper functioning of system	2324 0403 2051	2 3 1	Parachute 20,800					TM Test		System received no signals after "A" signal because of faulty Ledex relay
10	13 July	Verify proper functioning of system	2051 2324 0403	1 2 3	Parachute 21,000	22000	11 Miles	3200 yds	0.82 M	Test Yes TM	10-20 30-50*	Only missile SN 2324 was fired. Others used for data.
11	17 July	Verify the proper functioning of the system firing multiple missile salvos from fore and aft bays	2392 047 2389 2385	1 3 4 5	Q-2A 31,700	31200	7 Miles	3000 yds	0.88 M	UN TM Yes UN	0-10	
12	18 July	Verify the proper functioning of the system firing multiple missile salvoes from fore and aft bays	647 2590 2591	2 3 4	Parachute 31000		10.5 Miles		0.83	TM		Missile bay doors failed to open.
13	18 July	Verify proper functioning of system	2345 0404 2051	4 5 6	Parachute 25000		4 Miles		0.81	TM Test		To late lock-on on one pass and target too low on second pass
14	19 July	Verify proper functioning of system	2345 0404 2051	4 5 6	Q-2A 12700	13500	4 Miles	1500 yds	0.94	Yes TM Test	20-50	
15	20 July	Verify proper functioning of system firing in salvo from fore and aft rails	2380 0404 2051 2522	1 2 3 4	Parachute 26800	27900	9.5 Miles	4420* yds	0.84	Yes TM Test Yes	UN 10-30	No carrier on telemetry missile Test missile showed proper operation
16	20 July	Verify proper functioning of system	2591	4	Parachute 13000	13500	8 Miles		0.84			Missile doors did not open. Failure of armament micro switch in nose wheel well was cause
17	20 July	Same as flight 15	2587 2590 2051	1 5 3	Parachute 34600	39600	7.75 Miles	5000 yds	0.81	Yes Yes Test	Hit 20-50	
18	23 July	Verify system capability in firing typical 3 missile salvoes	2401 2598 2396 2505	4 5 6 2	Q-2A					TM		Target went out of control soon after launch
19	23 July	Same as flight 18	2051 2312 2509 2991	1 3 4 5	Q-2A					Test		Same as flight 18
20	23 July	Demonstrate the performance of system and gather data when firing 3 missile salvo	2401 2598 2398 2505	4 5 6 2	Q-2A 12300	12700	5.5 Miles	2000 yds	0.80	Yes Yes Yes TM	10-20 20-50 50-100	
21	23 July	Same as flight 20	2051 2312 2508 2591	1 3 4 5	Q-2A 12400	13400	4.25 Miles	1720* yds	0.84	Test Yes Yes UN	20-50 532* 1103*	Actual miss distance data received on two missiles
22	24 July	Same as flight 20	2427 2518 2501	5 4 6	Q-2A 39700	38700	4.25 Miles	5000 yds	0.94	Yes Yes Yes	0-10 0-10 20-50	
23	24 July	Same as flight 20	2519 2430 2463 2051	3 4 5 1	Q-2A 40000	40700	5.5 Miles	5400 yds	0.92	MF Yes Yes Test	0-10 0-10	Misfire caused by loose launcher down interlock switch
24	25 July	Same as flight 20	2352 02531 2348	4 5 6	QB-17							Mission aborted because of weather
25	25 July	Same as flight 20	2403 2408 02538 02450 2579 2327	1 2 3 4 5 6	QB-17							Mission aborted because of weather
26	27 July	Same as flight 20	0625 2429 2510 2051	5 4 6 3	Q-2A							Q-2A drone spun in on final turn to a firing run
27	27 July	Same as flight 20	02450 2467 0880 02505	4 2 1 3	Q-2A							Q-2A drone spun in on final turn to a firing run
28	30 July	Same as flight 20	0625	5	Q-2A	41500	0	4500	1.05	UN	UN	Believe magnetron detuned when

27	27 July	Same as flight 20	[illegible] 2467 0880 02505	4 2 1 3	Q-2A							Q-2A drone spun in on final turn to a firing run
28	30 July	Same as flight 20	0825 2429 2510	5 4 6	Q-2A 40100	41500	8 Miles	4500 yds	1.05	UN UN UN	UN UN UN	Believe magnetron detuned when nose gear armament switch failed immediately after firing
29	30 July	Same as flight 20	0880 2467 2461 2051	1 2 4 3	Q-2A 41000							First pass aborted because rate of closure was too low. No additional passes made because of loss of power a GCI control
30	31 July	Investigate performance of system at extreme altitudes	0401 2464 2463	1 3 2	Balloon 56000							Radar pressurization went out at 50,000 feet
31	31 July	Demonstrate system at limit of altitude operational specifications	0880 2467 2538 2461 2579 2327	1 2 3 4 5 6	Balloon 52000	51400	13 Miles	7000 yds 7417*	1.07	Yes Yes Yes	Hit UN UN	Second firing of 3 missiles not made because of destruction of target
32	31 July	Same as flight 20	0880 2467 2538	1 2 3	Q-2A 28000	28700	4.25 Miles	3200	0.94	UN UN UN	UN UN UN	
			2390 2391 2519	4 5 6	Q 2A 25700	28100	5 Miles	3000	0.94	Yes Yes UN	20-50 20-50 UN	
33	1 Aug.	Same as flight 20	2352 02531 2346	4 5 6	Q-2A 30000							No missiles were fired because of poor weather and GCI control failure
34	1 Aug.	Demonstrate system at limit of altitude operational specifications	2352 02531 2346	4 5 6	Balloon 51000							Mission cancelled because of weather warning
35	1 Aug.	Same as flight 34	0401 2463 2464 02505 2368 2362	1 2 3 4 5 6	Balloon 50000							Radar lost pressurization at 49,000 feet
36	2 Aug.	Same as flight 34	2352 02531 2346	4 5 6	Balloon 51000							Mission aborted when could not get relight on afterburner
37	2 Aug.	Same as flight 34	0401 2463 2464 02505 2368 2362	1 2 3 4 5 6	Balloon 60000							Mission aborted — radar lock-on sensitivity to high and would not track.
38	2 Aug.	Investigate extreme altitude capability of system	0401 2463 2464	1 2 3	Balloon 54000	53000	12.5 Miles	7800 yds	1.07	Yes Yes Yes	Hit UN UN	Missile war head exploded when it hit the balloon
			02505 2368 2362	4 5 6		51500	11 Miles	7500 yds 7582*	1.1	MF Yes Yes	 UN UN	Misfire caused by igniter failure
39	3 Aug.	Same as flight 38	02531 2346	5 6	Balloon 55000	52000	9.5 Miles	UN	0.94	Yes Yes	50-100 50-100	Missile passed below target. Large miss believed caused by large launch error at high altitude
40	3 Aug.	Same as flight 38	2403 2408 00185 2368	1 2 4 5	Balloon 58600	54000	15 Miles			 MF MF MF		Misfire caused by faulty launcher down interlock switch
41	6 Aug.	Same as flight 38	2515 2521	4 6	Balloon 55000		12 Miles			MF MF		Misfire caused by faulty launcher down interlock switch
42	6 Aug.	Same as flight 38	2368 00185	4 6	Balloon 55000							Mission aborted when AC-DC power drive gear box failed
43	7 Aug.	Same as flight 38	2502 2505	4 6	Balloon 55900	55300	11 Miles	7000 yds 7583* yds	0.94	Yes UN	Hit UN	
44	8 Aug.	Demonstrate system operation at maximum level flight speed	2517 2408 2460	4 5 6	Q-2 41500	41000	10.3 Miles		1.15	MF MF MF		Misfire caused by faulty launcher down interlock switch
45	8 Aug.	Same as flight 44	2523 2382 2362 2352 02505	1 2 3 4 5	Q-2A 40800	42500	7 Miles	4500 yds 4723* yds	1.11	Yes Yes Yes No Yes	0-10 0-10 20-50 20-50	612 ft.* 22 ft.* UN* 6 ft.* 22 ft.*
46	13 Aug.	Demonstrate capability of system in firing complete salvoes in beam attacks	2403 2460 2520 2462 2517 2468	1 2 3 4 5 6	QB-17 18500	19000	10 Miles			MF MF MF MF MF MF		Mission aborted because of failure of No. 1 launcher rail up interlock switch
47	13 Aug.	Same as flight 46	2515 2512 2368 2521 00185 2405	1 2 3 4 5 6	QD-17 10500	18800	0.8 Miles	4000 yds	0.90 –	MF MF MF Yes Yes Yes	 100-200 100-200 100-200	The forward three missiles failed to fire because of a faulty inter valometer
48	15 Aug.	Same as flight 46	2512 2460 2360 2403 2520 2462	1 2 3 4 5 6	QB-17 18200	17800	11.5 Miles	3700 yds 3930* yds	0.91	Yes No Yes Yes Yes Yes	10-20 10-20* 50-100* 10-20* Hit*	Warhead on all missiles were clipped. Hit did very little damage to vertical fin
49	15 Aug.	Same as flight 48	2517 2515 01183 2462	1 4 5 6	QB-17 10400	18800	7 Miles	3500 yds 2575* yds	0.93	Yes Yes No Yes	100-200 20-50 10-20	UN* 55 ft.* 130 ft.* 662 ft.*

NOTE:
MF — Misfire
UN — Missile guidance and miss distance unknown
CAP — Captive missile used to measure parameters
TM — Telemetry missile (never fired)
TEST — Test missile (missile Simulator) with recording oscillograph

*Information marked with asterisk is reduced data from the Holloman Range. Miss distances from such data are measured from the center of the target. Miss distances determined from visual data are estimated from nearest point on target.

☐ armament system deficiencies

The armament and fire control system as a whole constituted a large portion of the maintenance effort expended on the Phase VI aircraft. A total of 297 malfunction reports were generated against both systems. The paragraphs which follow comprise discussions of the major malfunctions which occurred within each system during the entire program and within the missiles which were fired at HADC.

☐ *Launcher Rail Snubbers:* On four occasions while performing maintenance in the missile bay, the missile launcher rails were inadvertently retracted without snubbing air. This caused the rails to engage in the "UP AND LOCKED" position with maximum force, causing extensive damage to up-lock fittings and buckling of several bulkheads in the missile bay area. Ground crews must be especially careful of this condition and be closely supervised.

The precautions taken in the past for preparing the aircraft to insure safety of personnel while working in the missile bay regions resulted in deterioration of the pneumatic system. Contractor Engineering Design Sections have offered a possible way of safetying the aircraft so personnel may enter the missile bay area to reposition missile launching gear, and load missiles without the possibility of an inadvertent operation of the armament gear. The following steps are necessary to de-activate the armament system and prevent movement of the components in the missile bay.

Open the doors pneumatically (do not lower launchers pneumatically) and proceed as follows:

1. The two armament circuit breakers in the main wheel well must be safetied in the de-energized position. This step removes all pneumatic power to the launcher and door cylinders. No movement (other than manual) is possible with circuit breakers in this position.

2. Missile bay door selector valve electrical disconnect plug (station 316, right-hand center missile beam) shall be disconnected. This is a double safety precaution in addition to item #1 above. When the doors are opened pneumatically, the selector valve will remain in "door open" position, and when the d.c. electrical power is disconnected the valve will shuttle to a neutral position and will vent air from the door actuating cylinders and connecting lines. Install missile door safety bars as soon as doors have been opened and armament system de-energized. Pneumatic power cannot be delivered to close the doors until the door selector valve is re-connected and armament electrical power is reapplied at the main wheel well circuit breakers.

3. Manually lower all forward or all aft missile displacing gear. Repositioning of the gear places the system in a non-rest position and requires the system to be re-set before any door or launcher operations can be accomplished. This is a double safety precaution to prevent operation of launchers.

With the armament power and re-set circuit breakers in the de-energized (open) position, no voltage is available to the system from the 28 V d.c. bus to operate the door selector or the launcher selector valves, the intervalometer, the armament control relay panel, the "ARMED-SAFE" switch, the armament control relay box, or the ground control panel in the main wheel well. This removal of voltage will not affect radar pre-flight operation or a radar check-out operation.

Disconnect the forward and aft launcher selector valve electrical disconnect plugs on all aircraft that do not have instrumentation platforms installed in the missile bay area.

The nose landing gear limit switches may be activated in any position without interferring or removing any safety precautions, but it should be noted that the nose gear limit switches should be in a ground borne position (not actuated) before the armament system is reactivated by energizing the armament circuit breakers.

☐ *Interlock Switches:* Five failures occurred in the armament interlock switch system. These failures were caused by failure of the launcher up or down switch to make contact, thus preventing some or all of the missiles from being fired. A failure of any of the forward or rear launcher down interlock switches will prevent all of the missiles in that group of three from being fired since they are all connected in series. A failure of any one of the forward launcher up interlock switches will prevent all the missiles, both fore and aft, from firing. This condition is unsatisfactory. For safety reasons it is desirable to prevent the missile behind the one that is not indicating up and locked from firing, but not the remaining missiles. Difficulty was experienced in keeping these switches in adjustment, therefore, after several failures they were checked before every flight. However, this was still not sufficient, because airloads could not be simulated.

☐ *Radome:* Difficulty was experienced in removing and replacing the radome assembly. Mounting bolts are extremely difficult to reach with normal tools available. A special Allen wrench was made for this purpose. It consisted of a piece of hexagonal bar stock approximately 1 inch in length which was fitted into the end of a 6 inch racket extension. At present the task of removing or replacing the radome requires approximately 4 manhours of work.

☐ *Cable Marking:* Lack of sufficient marking on coaxial cables caused much needless maintenance delay. If a coaxial cable is not marked with some easily visible code number, it is virtually impossible to determine to which plug it should be connected when there are a number of choices.

☐ fire control system deficiencies

Range Gate Harmonization: The principal difficulty in maintaining this system was the excessive ground operating time necessary for range gate harmonization. At present, using the range gate servo unit (P/N 463106-X120), range gate harmonization is considered necessary to insure proper guidance of the missiles. It was found that once set, the tap setting on the range gate servo unit did not vary more than one tap from a given average setting for that unit for the majority of missiles installed. There were, however, variations from this form in significant number such that it is believed with this type of range gate servo unit installed, it is desirable to harmonize each missile. With the new range gate servo (P/N 463106-120) installed, harmonization showed that though there is an optimum setting for each unit-missile combination, the movement of the adjustment tap did not affect the missile and radar range gate coincidence significantly. Only three of the new units were tested, however, and further testing to determine the requirement for harmonization with the new unit was indicated.

☐ *Transmitter-Receiver Unit:* Arcing in the magnetron and pulse forming networks, and magnetron failing to tune to missile frequency were the most frequent troubles encountered with this unit. Four units were replaced in one aircraft at HADC.

☐ *Computer:* The most serious difficulty found in this unit was the failure of the Ledex units in the computer mode plug to make contact, causing the system to "hold" in early attack phases such as "A" time. This difficulty can be corrected by cleaning the contacts or bending the contact arm slightly until it makes more positive contact.

☐ *Relay Assembly:* This unit contains many relays, which are subject to sticking in one position or chattering. Since this unit supplies signals to many other parts of the system, a malfunction in this unit may adversely affect other portions of the system.

☐ *Electrical Synchronizer:* The most common difficulty caused by this unit was excessive range gate drift which could normally be corrected by adjustment. This adjustment is necessary approximately every 25 hours of operating time because of changing tube characteristics.

☐ *Automatic Frequency Control Tuning Unit:* These units were found to have a relatively high failure rate (8 failed during the 2 month firing program at HADC). The most serious defect was that over a period of time the automatic frequency control crossover point drifts until it cannot be brought into the prescribed limits with the adjustment means available.

☐ *Radar Pressurization:* While flying at altitudes of 40,000 feet or above the radar transmitter cut off due to lack of radar pressurization. This difficulty was found to be caused by the air intake filter to the pressurization control. This deficiency was corrected by changing the filter from a diameter of 10 microns to 40 microns, and cleaning it every 10 operational hours.

☐ *Motor Generator Unit:* This unit was considered satisfactory, however, all leads to this unit were connected by a terminal strip arrangement. If this method of correction were changed to a cannon plug type connector much confusion would be eliminated.

☐ *Pilot's Indicator (Radar Scope):* Many of these units were received from the contractor with the astigmatism and focus so far out of tolerance that the resulting fuzzy display could not be corrected by adjustment. The light filtering supplied with this unit was found to reduce the light from the display to such an extent that the filter had to be removed for daytime operation.

☐ *Target Simulator:* This unit was found difficult to maintain and inadequate for use in range gate harmonization. New units were received but could not be made to operate in time to be of use in these tests.

No provisions have been made to enable ground crew men or pilots to perform certain desirable checks of the armament and fire control systems either on the ground or in the air. The following methods of checking are required to insure satisfactory operation:

1. A method for completely checking the operation of the armament control relay assembly separate from the system should be established. Frequently, troubles have been traced to this unit and excessive time is necessary to determine which of the many relays in the unit is malfunctioning.

2. A method should be established for supplying power to a missile installed on a launcher rail in lieu of an expensive, short-lived missile battery. Some method of supplying power to a missile is absolutely necessary when performing range gate harmonization when the -110 or -V120 range gate servo units are installed. It is also necessary when the system is operationally checked with a live missile or a recording missile installed.

3. In performing range gate harmonization a method of artificially supplying computer timing signals ("A" and "B") and a method of interrupting the range voltage to the range gate servo unit is necessary. Since no method was provided, units to perform these functions were locally manufactured for use during this program.

4. A method for checking the arrival and sequence of firing pulses to the missile rails and to the rocket tubes should be provided. At present, the only method of checking the arrival of these signals is by wiring squibs or electrical fuses to the proper points and then checking whether these have blown after an armament cycling check. Though this will determine whether a firing signal has arrived, it will not check whether these signals have arrived in proper sequence.

☐ missile deficiencies

Missile Hydraulic Power Supply: A serious problem affecting missile parts consumption and repair time was uncovered during the Phase VI firings. While having a.c. power on for harmonization at the same time missiles were being loaded, 6 missile hydraulic power supply squibs were blown as the umbilical plug made contact with the missile. This was attributed to mis-mating of the plug and receptacle as the plug lowered to meet the missile. In such a case the possibility existed of either contact 32 or 36, which carried 30 volts a.c. and 28 volts d.c., respectively, touching the pin leading to the hydraulic power supply squib.

The reason for loading missiles when AC power was "ON" was to reduce the total time required in preparation for a mission by simultaneously loading some missiles and harmonizing others. It is expected that to achieve short turn-around time under operational conditions, a.c. power will be "ON" for fire control system checks, repairs, or adjustments, while new missiles are being loaded. Therefore, this problem represents an unsatisfactory condition requiring correction.

☐ *Missile Heat Control:* Although only one malfunction of this unit has been detected, it is assumed to be the possible cause of the failure of missiles to guide. Though a check of the system with the missile auxiliary test set showed this unit to be operating properly before flight, an intermittent failure of a relay in this unit subsequently occurred in flight. This failure was only detected when it occured again on the ground after flight, and caused the destruction of a converter unit in a missile being harmonized.

☐ *Missile Batteries:* In the check-out of the missiles it was noted that several batteries which failed the test at HADC were considered satisfactory in a re-check at the Hughes Aircraft Company Tucson facility. At HADC, an ohmmeter type of test with a direct short giving full scale deflection on a 50 microampere meter was used. A deflection corresponding to 5 microamperes or higher is cause for rejection of the battery for too high leakage. The test at Tucson by Hughes Aircraft Company is a voltmeter type of test. The difference in tests probably accounts for the difference of opinion on the acceptance of the battery.

☐ *Chassis V:* Chassis V in the missile had the highest number of failures of any individual missile component. Again, the Tucson facility of Hughes Aircraft Company has approved some Chassis V's after they were rejected at HADC. The reason for this may be the complete missile type of test at HADC as opposed to the unit type of test at Tucson. A new method of testing requiring a given radio frequency power level rather than a given automatic gain control voltage may eliminate the difference between HADC and Tucson results.

☐ *Missile Guidance Units:* Guidance units experienced a high failure rate. The failure rate was greatest in the case of Lot #8 and higher missiles. Leakage of air into the evacuated guidance unit was suspected because this would reduce the speed of the head when power was removed (at launch), allowing the head speed to decrease too rapidly.

☐ *Missile Harmonization:* Harmonization (alignment of range gate servo with missile) was especially difficult in the HADC climate where ambient temperature was high. Without missile cooling equipment (blower and hose adapter to fit missile) power could be applied to the missile for only 1½ minutes at a time without shortening missile life by overheating. A new range gate servo has been designed with lower gain so that the voltage sensitivity to a small time delay is greatly reduced. This should eliminate the requirement for harmonization with every missile. Six of the new range gate servos were received near the end of the F-102A Phase VI firing. Only three of these were usable and were installed in the aft bay of F-102A S/N 53-1801. Because the new range gate servos were acquired very late in the program, insufficient information was obtained to be certain that harmonization of each missile will no longer be required. If harmonization will be required, then missile cooling equipment will be needed in addition to a special missile auxiliary power unit or rechargeable batteries.

☐ *Missile Ignition:* One igniter failed when the firing signal was received. The squibs were blown but the black powder in the igniter failed to burn and fire the rocket motor. Such a malfunction is a safety of flight deficiency.

☐ *Failure Summary:* The following is a summary of the unit failure as detected by the missile check-out console.

TABLE No. 16

Missile Component	Part No.	Number of Failures
Converter Unit	IBHA 125340	14
Guidance Unit	IBHA 122300	11
Hydraulic Power Supply	IBHA 122470-801	0
Electrical Power Supply	IBHA 125995	4
Chassis I	IBHA 125896	3
Chassis II	IBHA 125516-801	5
Chassis III	IBHA 125444-801	6
Chassis IV	IBHA 125630-801	9
Chassis V	IBHA 124210	24
Chassis VI	IBHA 125975-801	8
Servo Positioner	IBHA 138391	0
Battery	IBHA 999078-1	6
Rate Sensor	IBHA 104360	1

GROUND SUPPORT EQUIPMENT

Tests have demonstrated that the F-102A weapon system is totally contingent upon the quality and quantity of the ground support equipment. This fact repeatedly manifested itself through operational delays caused by lack of immediate availability of such commonplace items as high pressure air or electrical power. Some of the major faults noted in the support equipment are:

- *Joy Compressor:* Considerable difficulty was encountered with the clutch unit. It was difficult to engage the clutch and bring the engine up to speed with a load on the compressor unit. Clutch wear was excessive and frequent adjustment and replacement of clutches were necessary.

- *Canopy Safety Bar:* The wooden canopy support wedge which was supplied to prevent inadvertent closing of the canopy during ground maintenance could become dislodged. A locally manufactured metal, hinged, split tube placed around the actuator cylinder rod was used to prevent the canopy from closing inadvertently.

- *Engine Removal Stand:* The engine removal stand, manufactured by the Hammond Manufacturing Corporation, has several design deficiencies which are noted as follows:

1. The aft rail brackets strike the engine shroud blankets on both left and right-hand sides. Local modifications consisted of cutting the corners off the brackets to prevent further damage to the brackets.

2. The No. 1 and No. 2 braces on the left-hand rail support obstruct entrance of the engine into the fuselage by chafing the engine transition duct. These braces were modified by removing the protruding angle which strengthens the side brace.

3. There is no warning placard on the unit to instruct personnel to pull the lock pin in the tow bar tongue. Failure to pull the pin results in damaged tow bar tongue plates.

During an evaluation of a modified Hammond engine stand the following discrepancies were noted:

1. There were no stops at the aft end of the engine stand rails to prevent the engine from rolling on or off the stand.

2. There was not enough clearance for the upper half of the roll-over stand to clear the tail cone fairings. To get the stand even close to engine removal rails, the upper half of the stand must be removed.

3. The engine stand rails do not match up with the engine removal rack in the fuselage.

- *MA-1/MA-4 Air Conditioner:* The present method of cooling the MG-3 radar during ground operation has been through the use of the MA-1 air conditioner. The MA-4 air conditioner, designed specifically for this aircraft, did not arrive at this Center until the latter portion of the test program. The MA-1 cooler is a KECO air conditioning unit procured for the B-52 aircraft. It was designed for use in situations where the ambient temperature is not less than 70 degrees F. When the temperature drops below this limit, the lubricating oil within the until will frequently congeal. When this occurs, the unit immediately stops running and cannot be restarted until the oil becomes sufficiently warm. Intermittent operation of these units considerably hampered the ground maintenance effort on the MG-3 radar.

A major fault with the MA-4, aside from its extreme size and weight, was that the starter rod was much too hard to operate. Several times the rod was broken in an effort to start the cooler.

☐ *MA-1 Air Starting Unit:* The major fault with this unit was the fitting which connects to the aircraft. This fitting will match only the F-102A aircraft. If it could be redesigned so that it could fit other aircraft as well, utilization of the starter would be greatly improved.

OPERATIONAL ANALYSIS

Pilots, almost without exception, are enthusiastic regarding the handling characteristics of the F-102A aircraft. Many pilots dislike certain operational features of the aircraft, such as instrument unreliability, poor brake release, engine compressor stalls, and afterburner flame-out at altitude. The discussions which follow summarize the pilots' comments relating to the various phases of flight and the operational characteristics of the aircraft systems.

☐ taxiing and ground handling

Adequate taxi speeds can be maintained with the throttle at or near the "IDLE" rpm position. Visibility to the front and sides of the aircraft is satisfactory. Visibility from each wing tip to the rear of the aircraft is non-existant with the F-102A canopy installation. Taxi guidance is accomplished with nose wheel steering which is generally satisfactory. Slightly excessive rudder pressures are required, particularly when executing a turn of minimum radius. Steering response is immediate and smooth except when returning to neutral operation after a short radius turn. The response in this case is sudden and jerky.

It is difficult to determine the actual amount of braking action being obtained through brake pedal forces. The pedal forces required during application of the brakes are relatively constant from initial application to the point where the wheels skid. This is further aggravated by the large pedal force required to obtain braking action. Brake pedal release is followed by a momentary lag before actual release of the brakes. Since the actual amount of braking action cannot be accurately determined, the momentary lag in brake release can cause a skidding tire to rupture. An anti-skid feature should be incorporated in this brake system to alleviate this problem. This lag sometimes results in a non-simultaneous release of the wheel brakes, which reduces effectiveness of ground control, particularly on take-off. For this reason nose wheel steering must be used for directional control during the initial take-off roll.

☐ take-off and initial climb

A nose-high attitude is necessary at take-off which, from observation, appears to be excessive and to require a special pilot technique. This attitude is required because of the characteristics of the delta wing. However, to the pilot in the cockpit this attitude appears normal. Trim changes after take-off are satisfactory. Longitudinal forces during acceleration and configuration changes are relatively light.

Phase VI testing of the aircraft resulted in development of a positive take-off technique. The rudder and elevon controls are automatically positioned to take-off trim by actuating a button in the cockpit. The aircraft is aligned with the runway, making full use of nose wheel steering to center the nose wheel. After engine check-out at military power is completed, the brakes are released with nose wheel steering engaged. After directional control is assured, during initial acceleration, the afterburner is lighted. A distinct increase in thrust will indicate a successful afterburner light. At approximately 70 knots IAS, the speed at which rudder control becomes effective, the nose wheel steering may be released. Differential braking should not be required. At 125 knots indicated airspeed, the aircraft easily rotates to the take-off attitude and will become airborne at approximately 155 knots IAS. Rapid acceleration after take-off requires that a steep climb attitude be maintained in order for the landing gear to fully retract prior to reaching gear limit airspeed. Take-off using military power is similar except for slower acceleration, longer take-off roll, and lower rate of climb.

maneuvering flight

The F-102A does not have a stall characteristic as conventional or as well defined as the stall characteristics of those fighters prior to the "Century Series" aircraft. There is no stall warning, since low speed pre-stall buffet appears at speeds well above the stall. The handling characteristics at low speeds are excellent with very good control response available throughout the low speed ranges. Because of the unusual stall characteristics and the fact that the aircraft has not yet been spin tested by the contractor, investigation of stalls was limited to a minimum speed of 110 knots IAS.

Longitudinal stability and control of the aircraft is considered satisfactory. A pitch damping system is employed to dampen longitudinal oscillations and is considered an adequate and necessary stability augmentor for high speed flight, particularly during radar tracking. With the pitch damping system operating, longitudinal oscillations are dampened very rapidly. Supersonic damping is most noticeably improved.

At 0.9 to maximum Mach number the aircraft exhibits continual lateral-directional oscillation with yawing of 2 to 3 degrees. In this speed range the aircraft is operationally unsatisfactory with the yaw damper inoperative. With the damper operating these inherent oscillations are immediately dampened. The presently installed yaw damper is satisfactory and is required to augment the poor inherent lateral-directional stability.

landing

The inherent stability of the aircraft at low airspeeds offer good control qualities in the landing pattern. The recommended initial approach speed is 300 to 350 knots IAS. The pitch-out turn toward the downwind leg is made using speed brakes and reducing power to arrive on that leg at 220 knots IAS. Landing gear is lowered prior to turning onto the base leg. After the gear has been lowered, the speed brakes may be closed to eliminate light buffeting. A speed of 200 knots IAS is maintained on the base leg. The turn to final approach is started at 190 knots IAS and finished at 180 knots IAS. The optimum approach speed is 160 knots IAS with power being used to maintain airspeed. "Over the fence" speed is 150 knots IAS where power is reduced to "IDLE" rpm and touchdown made at approximately 135 knots IAS. Immediately after touchdown with the nose wheel still off the ground the drag chute is deployed. At 100 to 110 knots IAS, the nose wheel should be eased down, controlling the rate of fall with the control stick. If the landing is made with the speed brake closed, a slight delay is required for drag chute deployment. Nose wheel steering can be effectively used to maintain directional control and braking action is satisfactory except as previously described.

☐ runway barrier tests

A total of 18 runway overrun barrier engagements attempts were made with F-102A air craft S/N 53-1800 to determine its compatibility with the standard Air Force MA-1 runway overrun barrier. The test aircraft incorporated pre-production main landing gear with a tread of 23 feet, 3 inches. The tests were made in two phases. The first phase, consisting of 7 runs, was performed at groundspeeds from 35 to 135 knots. There were three successful engagements. The landing gear doors were removed for the last three runs. The second phase consisted of 11 runs. During this phase landing gear doors were installed for all runs. A deflector was installed forward of the nose gear door and one over the landing light to prevent premature triggering of the webbing. Two different types of hooks were used on the nose gear strut to prevent the nose wheel from overriding the webbing. The hooks and deflectors were prototypes, the intent being to design final configuration hooks and deflectors of the same geometry if they were determined to be necessary and functional. These tests were made at ground speeds from 40 to 95 knots. There were five successful engagements.

Based on the results of these limited tests, the aircraft cannot be expected to engage the barrier at ground speed below 60 knots. At speeds between 60 and 70 knots engagement is marginal. Above 70 knots the aircraft can be expected to engage the barrier, if the webbing is not overridden. The probability of engagement increases as the speed increases. Without the hook over the nose wheel, which is not now a standard item on the F-102A, there is a possibility of overriding the webbing at any speed. If the hook is installed, the nose gear can be expected to engage the webbing. The inability of the aircraft to engage the barrier at low speeds is attributed to the unusually long distance between the nose wheel and main gear. The production model aircraft will have the main gear skewed forward with a shorter tread which could possibly reduce the lowest speed at which engagement can be expected.

A successful barrier engagement will, in most cases, result in torn main gear doors, broken door arms, and serrated main gear struts which will require replacement prior to further flight. The damage to these parts will be greater at the higher engagement speeds. Structural failure of the gear is not likely, and the decelration forces experienced by the pilot are negligible.

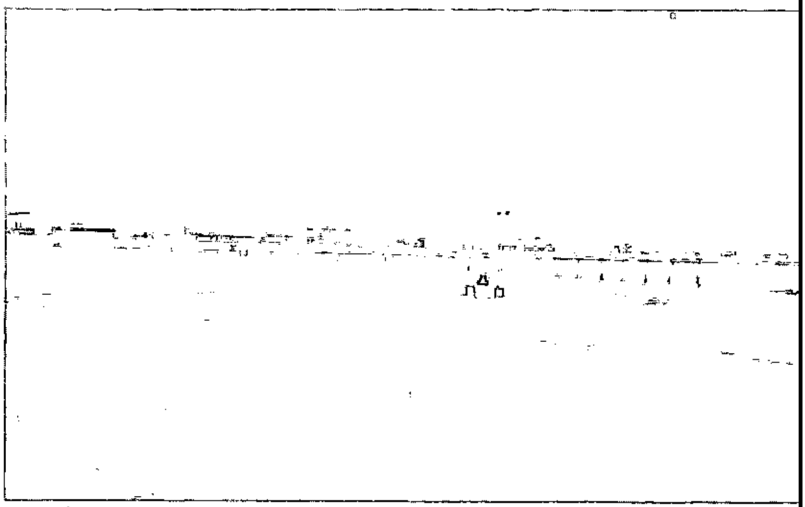
Directional control of the aircraft is normally not difficult to maintain if both main gears are engaged. The pilot should have nose wheel steering engaged and the throttle should be "stop cocked" before entering the barrier. The aircraft should enter the barrier at a 90 degree angle, although some deviation will not preclude successful engagement. Above 70 knots ground speed, nose wheel steering becomes extremely sensitive, making directional control more difficult. If only one gear is engaged, the aircraft will veer in the direction of the engaged gear. This tendency increases as the engagement speed increases. Directional control is maintained by using nose wheel steering and differential braking. The brakes should not be applied when entering the barrier because braking will lower the nose of the aircraft and possibly engage the webbing with the pitot boom.

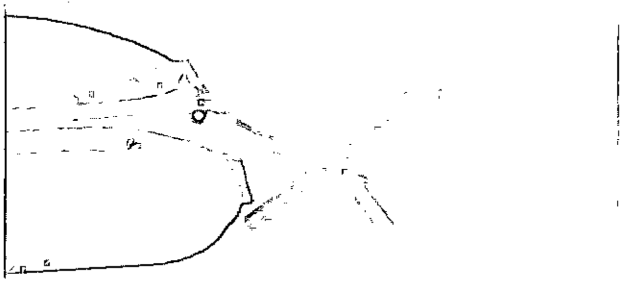
☐ description of tests

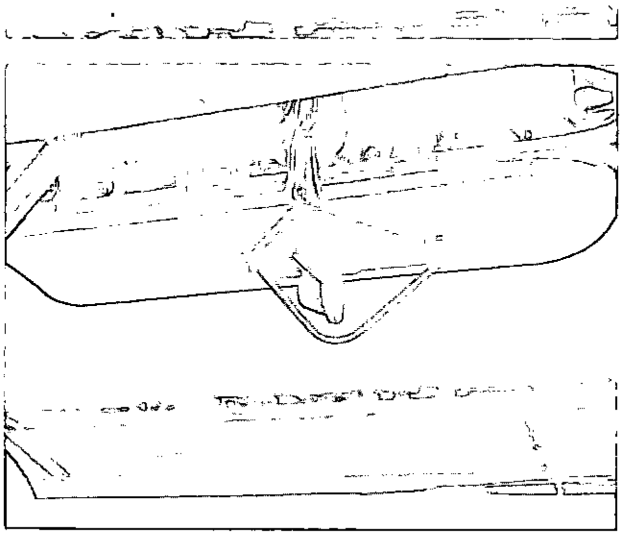
A standard USAF type MA-1 runway overrun barrier, as described in Technical Order 35E8-2-2-1, was used on a 300 foot wide runway.

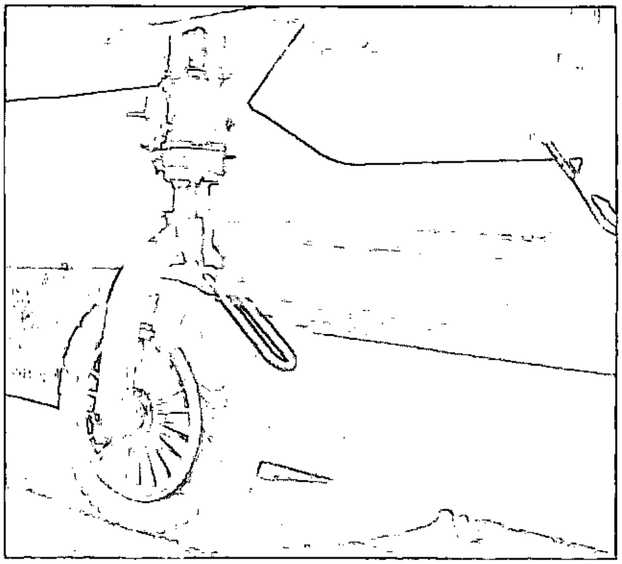
The F-102A aircraft, S/N 53-1800, was used for all tests in standard configuration except for the alterations listed below.

☐ *1st Phase:* Runs 1 through 4, the nose wheel door was removed. Runs 5 through 7, the nose wheel doors and outboard main gear doors were removed.









☐ *2nd Phase:* The gear doors were reinstalled for these tests. Runs 1 through 7, a deflector was installed forward of the nose wheel door as shown in Fig. 36, to prevent premature triggering of the barrier by the door. A deflector was installed over the landing light, as shown in Fig. 37, to prevent premature triggering of the barrier. A hook extending downward over the wheel was installed on the nose gear strut, as shown in Fig. 38, to insure engagement of the webbing.

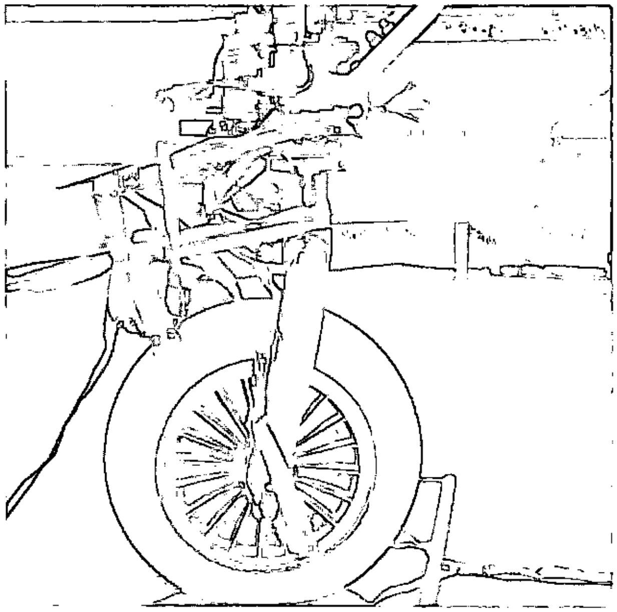
Runs 8 and 9, the landing light and deflector were removed. The hook over the nose wheel was removed and a modified smaller hook was installed higher on the nose wheel strut, as shown in Fig. 39.

Runs 10 and 11, the original hook, as installed in runs 1 through 7, was reinstalled.

☐ *Measuring Devices:* A Fairchild data camera was used on all tests to provide a measurement of the speed.

For the 1st Phase of the tests the only immediate-reading speed measuring device used was the standard airspeed indicator. This proved unsatisfactory because of the inaccuracy of the indicator in the low speed ranges and because ground speed rather than indicated airspeed was required.

For the 2nd Phase of the test, two speed measuring devices were employed. The standard airspeed indicator was replaced with a low range helicopter-type airspeed indicator to provide a more accurate measurement of the indicated airspeed in the desired ranges, and a speed measuring device similar to the type used by Highway Patrols to check the speed of automobiles was used. This latter device consisted of



a stop clock and two pneumatic hoses located 90 feet apart. The last hose was located just short of the barrier. The nose wheel started the clock when it passed over the first hose and stopped the clock when it passed over the second hose. This provided an accurate and immediate check of the speed. By making one or two runs with the barrier down just previous to the test run, the pneumatic speed device provided a calibration for the airspeed indicator under the prevailing weather and altitude conditions. It also provided a measurement of the speed for the test run.

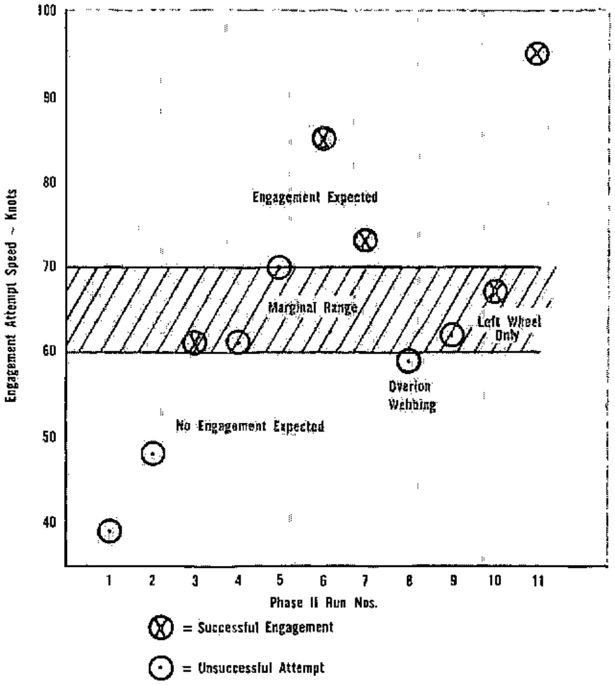
- *Photographic Coverage:* High speed black and white remotely controlled and hand operated cameras provided photographic coverage of the test. High speed color hand operated cameras provided documentary coverage.

TABLE No 17

BARRIER ENGAGEMENT TEST RESULTS

Run Number	Ground Speed-Kts	Engagement	Gross Weight-lbs	Stopping Distance-Ft
1	35	Unsuccessful — overrode webbing	NA	NA
2	44	Unsuccessful	NA	NA
3	52	Unsuccessful	NA	NA
4	61	Successful	20,900	735
5	100	Unsuccessful — overrode webbing	NA	NA
6	85	Successful at right wheel only	20,500	855
7	135	Successful	20,900	1365
		2ND PHASE		
1	39	Unsuccessful	NA	NA
2	48	Unsuccessful	NA	NA
3	61	Successful	21,300	670
4	61	Unsuccessful	NA	NA
5	70	Unsuccessful	NA	NA
6	85	Successful	21,000	880
7	73	Successful	21,500	721
8	59	Unsuccessful — overrode webbing	NA	NA
9	62	Unsuccessful	NA	NA
10	67	Successful — left wheel only	21,400	685
11	95	Successful	21,100	962

CHART NO. 6



first phase successful engagements

Run 4, Engagement Speed 61 Knots.

The engagement was successful and the aircraft was brought to a stop in 735 feet with no directional deflection of the aircraft. The only damage was turn main gear doors.

Run 6, Engagement Speed 85 Knots.

The right main strut only was engaged, causing the aircraft to swerve to the right in a fairly tight curve. A blow out of the left main tire resulted and the swinging action of the cable serrated the right main strut. No main gear doors were installed. The aircraft was brought to a stop in 855 feet.

Run 7, Engagement Speed 135 Knots.

The engagement was successful but the high engaging speed resulted in complete deployment of the chain, permitting the two chains to strike each other. This impact caused the chain connecting link to fail. The cable encircled the gear, pulling a portion of the chain with it. The chain struck the right main gear, broke the right strut above the wheel, and caused the aircraft to collapse on the right wing. Extensive damage was incurred to the wing and right main strut. The aircraft was brought to a stop in 1365 feet. All chain connecting links were replaced with a type which are believed to be safe from failure and additional chain was added to prevent complete run out.

second phase successful engagements

Run 3, Engagement Speed 61 Knots

The engagement was successful and the aircraft was brought to a stop in 670 feet. Both main landing gear doors were torn by the cable, and the door arms and hinges were broken.

Run 6, Engagement Speed 85 Knots

The engagement was successful and the aircraft was brought to a stop in 880 feet. The only damage was torn main gear doors and broken door arms.

Run 7, Engagement Speed 73 Knots

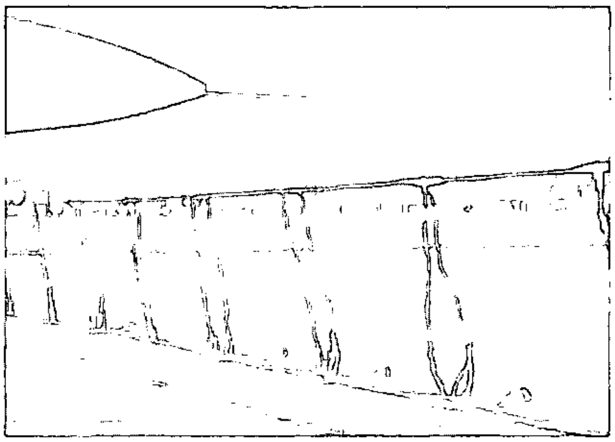
The engagement was successful and the aircraft was brought to a stop in 721 feet. The only damage which resulted was torn main gear doors and broken door arms.

Run 10, Engagement Speed 67 Knots

The left main strut only was engaged but the pilot was able to avoid veering from his course by steering with the nose gear. The stopping distance was 685 feet. The left main gear door was torn and the door arms were broken.

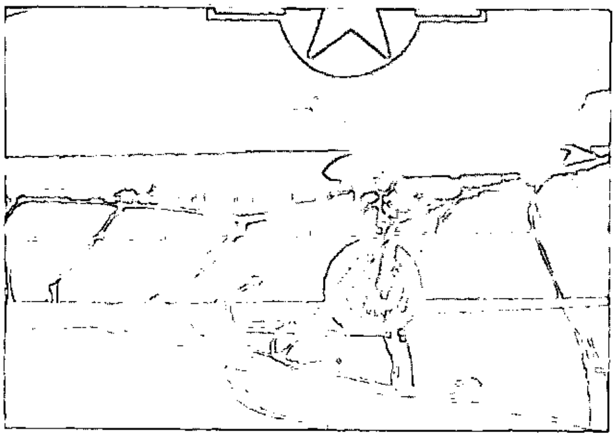
Run 11, Engagement Speed 95 Knots

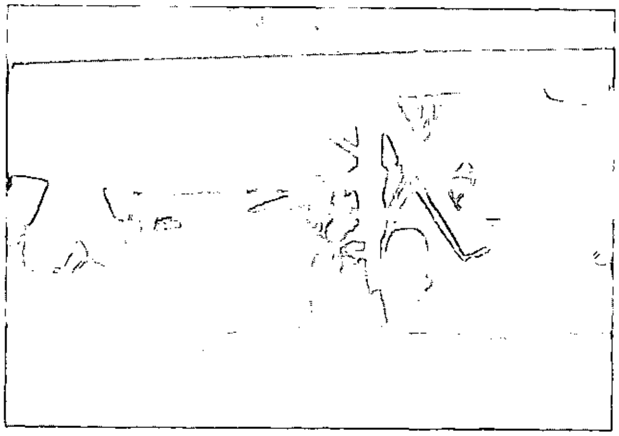
The engagement was successful and the aircraft was brought to a stop in 962 feet. The main gear doors were torn and the door arms were broken.

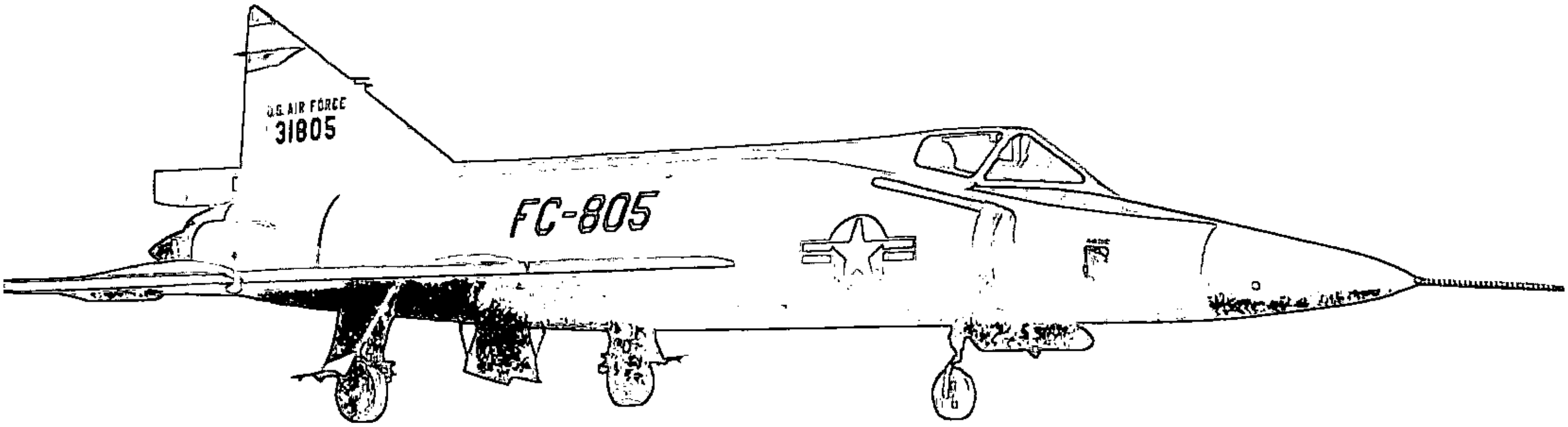


unsuccessful engagements

In the case of Phase 1, Run 1, at 35 knots; Phase 1, Run 5, at 100 knots; and Phase 2, Run 8, at 59 knots, the nose wheel overrode the webbing, thereby failing to trigger the barrier. In each case except Run 8 the hook over the nose wheel was not installed. In the other cases where engagement was not successful, the barrier was triggered and the cable rose but fell too low before reaching the main gear.







OPERATIONAL DEFICIENCIES

The F-102A weapon system is considered to be generally acceptable. There are several areas, however, which need considerable improvement. The deficiencies listed below adversely influence the operational effectiveness of the aircraft.

☐ cockpit general

Cockpit Entrance Ladder: Entrance to the cockpit is accomplished by climbing a ladder which is attached over the left-hand engine air duct adjacent to the cockpit. The outward angle at which the ladder hangs is unsatisfactory. The pilot should be able to climb either straight up or angled slightly in toward the aircraft as he ascends.

☐ *Anti-reflection Curtain:* The F-102A aircraft are provided with an anti-reflection curtain aligned longitudinally to split the space above and ahead of the glare shield. Its purpose is to prevent light entering one windscreen panel from crossing over and reflecting backward from the other panel into the pilot's eyes. It is fabricated from metal and cloth and is provided with a retracting device so that it may be stored.

Such retraction is considered unnecessary, since no known phase of flight or ground operation was interfered with by the erected curtain. (The optical sight was not available for Phase VI tests.) The service life of this curtain will be low, since stretching and sagging have already been noted, and some obstruction to forward vision can be expected with deterioration.

☐ *Glare Shield:* The shape of the instrument panel glare shield is considered unsatisfactory. Personnel utilizing the cockpit must exercise extreme care to avoid contacting and damaging the shield. The material is a semi-rigid plastic with an over-lay of black fiberglass which tends to crack with continued bending. This shield was intentionally constructed from flexible material to avoid pilot injury during bail-out, but the service life of the shield has proved to be less than desired. A retractable shield would be more desirable.

☐ *Canopy:* The engineering of the cockpit canopy system necessitates that the canopy be closed for nearly all taxi operation. Under hot weather conditions this procedure causes high temperatures in the cockpit. The cockpit air conditioning system is neither intended for nor capable of alleviating this condition during ground operation. It is estimated that such operation during hot weather in desert areas with the canopy closed will result in cockpit temperatures reaching unusually high values. Temperatures at this level will be intolerable for reasons of comfort, efficiency and safety.

☐ *Canopy Cinchdown:* To facilitate manual raising and lowering of the canopy in F-102A aircraft, a pneumatic counter-balance cylinder is installed which equalizes the weight of the canopy. A clamping, or cinchdown force (from the high pressure pneumatic system) is applied by the counter-balance cylinder during the final movement of the canopy closing action which compresses the canopy seal and facilitates engagement of the latches. The canopy cinchdown mechanism has been a source of trouble and delay in the general process of preparing for flight. Approximately 50 percent of the initial attempts at canopy cinchdown result in a failure which prevents completion of the canopy locking procedure. This occurs in spite of the fact that pilots, as a matter of procedure, assist in the operation by 1) placing their forearms on the sides of the canopy before it has reached the point where the cinchdown force is applied, and 2) exerting downward force during and throughout the process (including the normal delay period for pneumatic operation). Usually, by the second or third try the pilot is able to complete canopy closing and locking satisfactorily. Occasionally he must enlist the air of a ground crew man who places a ladder against the side of the aircraft, climbs up, and presses down on the canopy from the outside. Such unreliability of operation, together with the attendant delay when initial closing attempts fail, is incompatible with the scramble capability requirement of Air Defense Command aircraft.

☐ instruments

Accelerometer: The production aircraft instrument configuration does not include an accelerometer. This is considered unsatisfactory since the aircraft has g force limitations imposed for reasons of structural safety. Maximum tactical utilization will frequently require accelerations approaching these limits. An accelerometer would provide for maneuvering at optimum rates while staying within g limitations, and would provide a means for determining when limits are exceeded, thereby assuring proper inspection and contributing to flight safety.

☐ *Turn and Slip Indicator:* The turn and slip instrument installed in the production configuration initially indicates a turn in the opposite direction to the actual direction of turn. The indicator does not indicate the proper direction of turn or bank until the bank angle has been established and roll rotation has been stopped. Rolling out of a turn also produces false indication. This condition is caused by the fact that the instrument panel is tilted 15 degrees from vertical. The contractor has initiated an Engineering Change Proposal which installs a 15 degree wedge between the instrument and panel to correct the condition.

☐ *Afterburner Nozzle Position Indicator:* Malfunction of the afterburner nozzle position indicator system has resulted in considerable unscheduled maintenance. This system becomes unreliable after a few hours of operation. In several instances, engines had to be removed in order to perform maintenance on the indicating system when the nozzles were actually operating properly. In all cases of nozzle malfunction it was necessary for the pilots to use the ratiometer, tailpipe temperature, and fuel flow indication as a cross check to effectively ascertain afterburner nozzle operation.

☐ *J-2 Gyro-Magnetic Compass:* Precision in turns is reduced because of the low fidelity representation of actual aircraft heading which is inherent in the J-2 gyro-magnetic compass system and the radio magnetic indicator (RMI). This manifests itself as an error at

2. The return to search button at the top left side of the hand control is difficult to locate and awkward to operate because of its position and mode of actuation. Often when it is desired to break lock-on on return to search in intercept operations, the time available will be critically short. Speed and ease of actuation of this switch would be increased if it were located on the forward side of the hand control at a position more natural and convenient for operation by the tip of the index finger.

The majority of pilots expect the motion of the range gate on the radar scope to correspond to the motion of the nose of the aircraft. Thus, forward motion of the hand control should cause the range gate marker to move down (reduced range) and vice versa, eliminating confusion from the units being mounted on the control stick.

Some pilots have found that the position of the elevation vernier on the top of the control makes it difficult to operate while depressing the action switch.

☐ *Speed Brakes:* The speed brake is unsatisfactory due to ineffectiveness and excessive pitch change with brake operation. Pilots report that it is difficult to determine, through aircraft performance, whether the speed brake is open. Many aspects of the Air Defense mission, especially the interception and identification of slow flying aircraft, require effective speed braking. It is further felt that formation flying is unnecessarily difficult with the present speed brake. The contratcor has initiated Engineering Change Proposal No. 1093 which proposes incorporation of enlarged speed brakes on the F-102A.

☐ *"Flight Mode" Selector Switch:* The "flight mode" selector switch is provided to allow pilot selection of either direct manual control of the flight surfaces or control with stability augmentation (manual). Interpretation of this placarding is ambiguous. Pilots have reported situations where they were uncertain whether they were using the pitch and yaw damping with the switch in a given position. Since it is a safety requirement that pilots have complete control of the flight surfaces (no damping) during take-off and landing, it is necessary that the two positions be differentiated with proper placarding.

☐ lighting system

Cockpit Lighting Controls: Selections and intensity adjustments of the various portions of the cockpit lighting system are made several times during each night flight. The lighting control panel is located ahead of the yoke control stick at the bottom center of the instrument panel. In such location, the lighting control knobs are extremely difficult to see, identify, grasp, and operate. There are five of these controls on the panel: 1) flight instrument group edge lighting and standby compass; 2) instrument panel (other than flight instruments) edge lighting; 3) console edge lighting; 4) instrument panel flood lighting, and 5) console flood lighting. To allow easier identification and discrimination between controls, as well as to simplify operation, these lighting functions could be combined into three circuits and control by only three switches.

☐ *Landing Gear Warning Light:* Two aspects of th landing gear warning system are not satisfactor from a human factors standpoint.

The location of the red warning light within the gear actuating handles causes it to be so far removed from the pilot's normal line of vision that its attention-getting power is seriously impaired. During daylight flight the light is not easily discernable. Under night flight conditions the gear unsafe light tends to blend with the red lighted background. This condition is considered doubly critical by reason of the lack of an audible landing gear warning signal.

The landing gear warning light test switch is similar and adjacent to the emergency landing gear retraction switch. Inadvertent retraction of the landing gear could occur as a result of actuating the wrong switch when the intention was merely to test the landing gear warning light. Desired emergency retraction could be prevented or critically delayed through the necessity to visually determine which switch to actuate.

☐ *Warning Light Dimming:* F-102A warning lights are automatically dimmed when the center instrument panel lighting is turned on (if thunderstorm lighting is off). This is accomplished in order that the high intensities of illumination necessary for daytime flying in sunlight will be reduced to a level more compatible with the high sensitivity of the pilot's eyes under night conditions. However, night flying experience has revealed that the dimmed level of these lights is still too high for visual comfort and efficiency. Pilots report further that illumination of one or more of these lights causes a reduction in visibility and creates a considerable distraction of attention. Since illumination of any one of the lights on the warning light panel is accompanied by illumination of the master warning light, which can be re-set to off, their "dimmed" intensity should be reduced still further to a level no higher than necessary for good legibility of the legend on the lights.

▪ *Landing Lights:* The landing lights are inadequate in two respects. The general intensity of illumination provided by the lights is too low, and the light beams do not penetrate far enough along the projected approach path to allow proper anticipation of the flare-out maneuver for landing.

▪ *Formation Lighting:* Night flying during Phase VI evaluation of the F-102A has revealed deficiencies in the exterior lighting provisions as related to formation flying. The aircraft has no formation lights. The position lights are used in the "STEADY" position for night formation flying. These lights are concentrated in groups: on the fuselage above and below the wing root leading edges; at the wing tips; and at the tail cone fairings above inboard trailing edges of the elevons. This configuration of lights provides very poor definition of aircraft attitude and size for the formation wingman. Thus, the wingman does not perceive the lead aircraft in proper perspective. Pilots report that perception of the movement of the lead aircraft is delayed and sometimes reversed because of these factors and that effectiveness and safety of formation flying are reduced as a result. Part of this problem is due to the unique pattern of lights visible to the wingman along with the fact that the lights are not separated sufficiently to provide good indications for depth and relative positions. In support of this, it was reported by two pilots that formation flying was made easier if the lead pilot directed the beam of a flashlight through the windshield onto the radome.

Another deficiency is that the lights are too bright, even in the "DIM" position, for close-in formation flying. Visual comfort and efficiency are compromised as a result, thus further reducing the safety of night formation flying.

□ emergency landing gear extension

The average time to extend the landing gear in the emergency mode to the down and locked position is approximately 38 seconds. This excessively long time required for extension, using the emergency system, compares unfavorably with the normal extension time of 4 to 6 seconds. In the interest of good forced landing procedures in the event of a flameout, it is desirable to conserve hydraulic pressure for actuation of the control surfaces. The pilot would, accordingly, elect to use the emergency extension feature which employs air to extend the gear. In forced landings, however, it is often necessary to delay the extension of the landing gear until such time as the pilot is sure he can make his intended touchdown point with the added drag. It is unlikely that any pilot could be able to make consistently accurate estimates of his touchdown point while still 40 or more seconds away from that point. Thus, the requirement that he start emergency gear extension 40 seconds ahead of touchdown introduces a considerable reduction in the flexibility he has for controlling point of touchdown. Another consideration is that a pilot will delay too long in starting extension and make touchdown with partially extended gear.

□ flight controls

Lateral Trim Control Effectiveness: Pilots participating in the Phase VI test program are unanimous in the opinion that the lateral trim control action of the F-102A is unsatisfactory. The effectiveness of trim control in various maneuvers is less than optimum because of the excessive delay between initiation of lateral trimming action and the accomplishment of the desired trim. The delay is attributed to initial response lag coupled with too slow a rate of trimming action. Thus, the system does not provide the capability of "beeping" in lateral trim changes, a capability which pilots consider highly desirable on the basis of experience with other trim systems which provide it. This problem is considered important because of the F-102A's high sensitivity in roll response to lateral control movement. With such sensitivity an adequate trimming action is a necessity for effective control of the aircraft. Pilots report that it is very difficult to maintain a stable, wings level condition with the present combination of high control sensitivity and ineffective trimming action.

The problem of high lateral control sensitivity may be partly a result of shortcomings of the artificial feel system for the small range of lateral stick movement around the neutral position. In this connection, it is believed that the first small increment of stick movement from the neutral position affects the control servos but does not bring the artificial feel system into operation. Thus, what appears to be excessive sensitivity may be due in part to a small range of control movement which affects aircraft attitude but for which there is no kinesthetic feedback to the pilot.

The contractor should undertake a study of the overall problem of lateral control effectiveness in the F-102A aircraft, paying special attention to those aspects having to do with trimming action, control sensitivity and feedback.

☐ personal equipment

Pressure Suits: Operation of the F-102A by pilots wearing the MC-1 altitude suit and K-1 pressure helmet served to re-emphasize the general problems of comfort, mobility, and visibility which have been found universally among partial pressure suit wearers regardless of aircraft type. These problems have been well documented in more recent Phase VI reports, and require no elaboration since they are no more critical in the F-102A than in any other aircraft. The problems do become critical, however, when evaluated within the realm of the mission the F-102A has been assigned: air defense intercept. This mission will be accomplished only as needed, and the time of need cannot be predicted. Therefore, pilots as well as aircraft must stand ready for long periods on "alert" status. Because of the excessive time required to don the equipment, pilots ready for immediate "scramble" (as on 5 minute alert) must be completely dressed in the partial pressure suit and helmet. Such ready-room attire, with the attendant restrictions on comfort and activity, will be intolerable and will generally lessen pilot effectiveness through induced fatigue. To permit pilots to stand by on alert status without being completely dressed will, on the other hand, result in a considerable delay in a scramble take-off. This creates a situation which becomes intolerable from the standpoint of mission effectiveness. Thus, it will be readily recognized that the use of the altitude suit, in addition to imposing the usual restrictions on aircraft operation, will seriously reduce the capability of the F-102A to satisfy the rapid "scramble" requirements of the Air Defense Command.

☐ *MA-4 Lap Belt:* The MA-4 lap belt currently provided in F-102A aircraft is not satisfatcory in two respects:

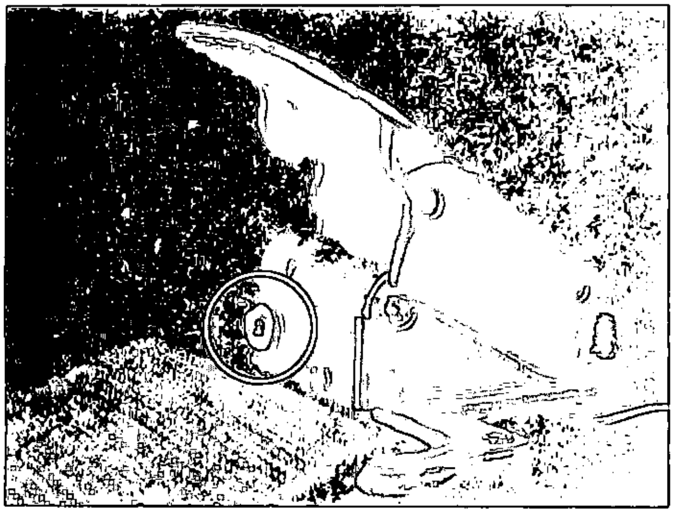
It does not have sufficient adjustment capability to allow the degree of take-up required by a pilot of average or smaller size torso and hips. This condition is noted in connection with use of the A-5 seat cushion which is delivered with the aircraft. This cushion is reported to be of equal or greater thickness as compared to the survival kit with which it will be used alternately. Thus, the problem of seat belt tightening will remain for all normal configuration of seat gear. A seat belt having adjustment capability sufficient to accommodate the range of pilot sizes is considered mandatory from a safety standpoint.

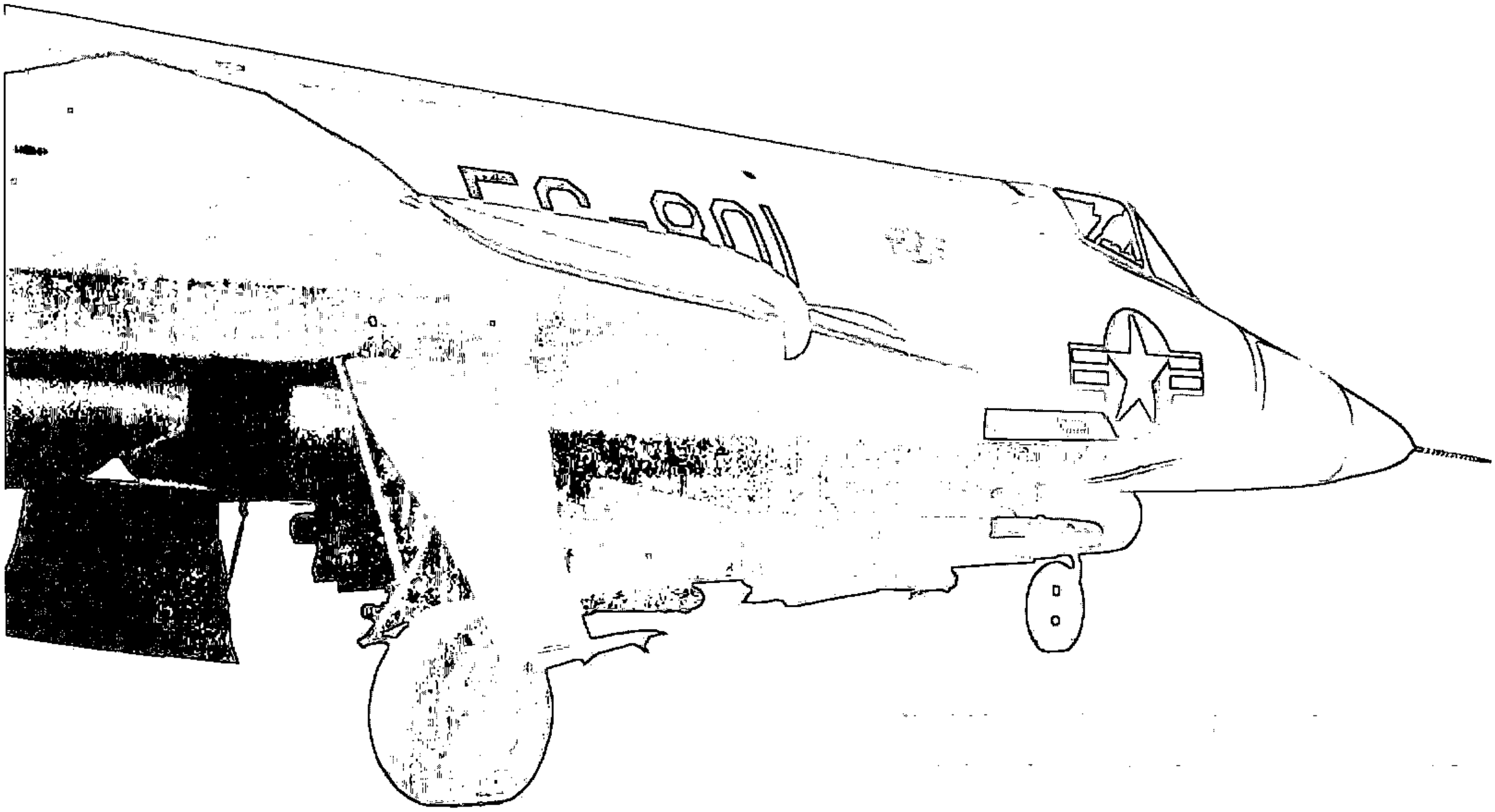
The aluminum threaded boss on the pilot's safety belt latch assembly where the initiator hose attaches has broken on two different belts. Flying safety was jeopardized since this failure would eliminate the automatic opening feature of the belt latch. The aluminum casting does not appear to have sufficient strength to withstand normal usage.

☐ fire control system

30-Mile Search Capability: The 30-mile search capability of the MG-3 fire control system is limited to long pulse operation. Prior systems, such as the E-4, have the advantage of both long and short pulse operation in the 30-mile search mode. Short pulse operation reduces the load on the transmitter with a slight reduction in the contact range of the system. The inclusion of the short pulse mode of operation will frequently allow mission completion when long pulse operation fails. Its alternative use with long pulse operation would lengthen transmitter life considerably.

▣ *Anti-Jam Action:* The anti-jam action of the MG-3 radar consists of transmitter retuning. Facility for continuous anti-jam tuning throughout the attack run is essential when and if sweep-type anti-jamming is employed by the target. Tuning is accomplished by depressing an anti-jam switch located on the upper left portion of the instrument panel. This switch must be physically held in the anti-jam position throughout the desired anti-jam period. This creates an unsatisfactory condition since it necessitates diversion of the pilot's hand away from more important functions during the attack run.





CONCLUSIONS

From a functional standpoint the F-102A is potentially an excellent fighter-interceptor aircraft. Operationally the aircraft cannot meet the time schedules imposed by Air Defense Command scramble tactics.

A number of unsatisfactory features must be eliminated before the aircraft can be considered satisfactory. The major deficiencies include compressor stalls and afterburner blowout at altitude; loss of pressurization at altitude; inadequate windscreen defogging; excessive fire control system ground maintenance which requires considerable specialized training; excessive armament interlock switch failures; and inadequate test equipment to perform system checkout. The automatic flight control system has not been proven and must undergo further testing.

The aircraft cannot be expected to engage the Standard Air Force MA-1 runway overrun barrier at ground speeds below 60 knots. Between 60 and 70 knots the engagement expectancy is marginal; above 70 knots the aircraft can normally be expected to engage the barrier.

Ground support equipment is adequate in most cases. The number of separate pieces of equipment required to support the aircraft is considered excessive.

RECOMMENDATIONS

Safety of Flight recommendations are indicated by asterisks.

aircraft systems

Power Plant:

1. A surge bleed valve governor "optimum schedule" be immediately developed to eliminate high altitude, high speed, engine deceleration stalls.

*2. AJA2 fuel controls be installed to eliminate low altitude engine acceleration stalls.

3. All aircraft be retrofitted with dual actuator "waffle" type afterburners to provide positive afterburner operation.

4. The number six swirl cup areas be provided with additional cooling to reduce cracking and buckling.

5. Further studies be conducted on the problem of turbine blade stretch and nozzle guide vane bowing.

6. Stronger cooling air bleed ducts be provided.

7. The Sundstrand return oil filter be relocated to a more accessible position to facilitate inspection without requiring engine removal.

8. Bearings and bearing retainer rings with greater service life be provided for the Sundstrand constant speed drive unit.

9. A more positive coupling be provided between the Sundstrand unit and its engine mounted drive gear box.

10. The cover plate be redesigned to permit a more rapid removal and replacement of the retaining cap screws.

11. A precision tool be provided which will greatly simplify alignment of the Sundstrand unit with its engine mounted drive gear box.

12. The oil fitting, P/N 8-27961, be redesigned to prevent inadvertent reversed installation.

*13. Further studies be conducted to provide fuel and oil hoses other than AVICA braided wire hoses. The type of hoses provided should be capable of being easily and thoroughly inspected.

☑ fuselage

14. A fuselage access panel be provided to permit rapid inspection, adjustment, or replacement of afterburner ignition valves and nozzle position indicator switches.

15. A more positive locking means for all radar access door handles be provided.

☑ landing gear

16. Structurally stronger landing light brackets be provided.

17. The existing outer wheel bearing seals be redesigned to positively prevent entry of water to the bearings.

18. Wheel bearings be lubricated after operation on wet ramps, taxi strips, and runway.

19. Consideration be given to reducing the period between wheel bearing inspections.

20. The structural integrity of the landing gear fairing door, linkage, and attaching pins be improved to permit greater maximum safe gear down speeds.

☐ air conditioning and pressurization

21. A study of cabin pressurization surging be conducted and that a positive remedy be immediately determined to eliminate this problem.

22. More suitable bearings and bearing retaining rings be provided for the refrigeration turbine assembly.

23. A more dependable cabin heat control be provided.

*24. A suitable and dependable windscreen de-fogging system be provided.

☐ hydraulic system

25. Further tests be conducted to determine the functional suitability of the modifications to the hydraulic pumps.

☐ pneumatic system

26. A more functionally dependable pneumatic pressure regulator be provided for positive pressure regulation.

27. The necessity of recharging the pneumatic supply while airborne be studied.

28. A means of determining priority system pressure be provided for the immediate reference of the pilot.

29. The source of air pressure for operation of the drag chute be made independent or isolated from the remainder of the priority system to insure drag chute availability.

☐ electronic cooling

30. A more dependable ram air regulator be provided to assure ground cooling of units in the electronics bay.

31. More frequent preventive maintenance inspections of the jet pump system be required to be accomplished.

32. A requirement be added to applicable technical orders directing that the cabin air circuit breaker be pulled at any time during which periodic maintenance is being performed on the aircraft.

☐ flight controls

33. The reliability of trim actuators and dampers for the control surfaces be improved.

34. The rudder trim switch be relocated to a more accessible location as near the throttle quadrant as possible.

35. Suitable speed brakes be provided.

36. The lateral control sensitivity be decreased. More stringent quality control should be exercised to insure that lateral trim motors provide the specified lateral trim rate.

37. The drag chute riser straps be shortened from the present length of 54 inches to 18 inches to minimize the problem of riser straps fouling the release of the chute.

38. Cable assembly, P/N 8-45359, be shortened from a length of 9 inches to 6⅛ inches so that the "D" ring will not prevent the locking jaws from closing.

39. The pilot chute be modified to prevent the spring in the cone from coming loose.

40. An improved jaw mechanism be devised and installed and that the drag chute canister be redesigned with a clean interior to minimize faulty "D" ring release.

☐ fuel system

41. The fuel flow equalizer diaphram be redesigned to satisfactorily resist wear inflicted by repeated contact with bilaterally positioned fuel jets.

*42. The existing low level fuel warning system be modified to make the system reliable. It is imperative that the fuel quantity gage be accurate, particularly in the low fuel remaining region.
A more suitable fuel indicating system should also be installed which will provide the pilot with the total amount of fuel remaining in each wing.

☐ communications system

43. The ARC-34 transmitter-receiver mount assembly be relocated approximately 3 inches aft of its present position.

44. The ARC-34 transmitter-receiver mount assembly be redesigned to permit easier removal and replacement.

45. The microphone and earphone jack boxes on the ARC-34 transmitter-receiver be relocated to permit more satisfactory access.

46. Covers be placed over all exposed electrical terminals.

☐ electrical system

47. The cockpit lighting functions (edge lighting, cockpit floodlighting, and instrument group lighting) be combined into three separate circuits with a control for each.

*48. Illumination for the electrical control panel be provided from 24 volt d-c essential power to assure electrical control panel lighting in the event of a-c power loss.

49. The possibility of electrical fires resulting from water leakage into the cockpit be eliminated.

☐ armament system

50. The armament interlock system be redesigned to permit the firing of all missiles except that missile which would create a safety problem.

51. The interlock switches be redesigned to provide a wider range of adjustment and to provide a positive ability to maintain adjustment through a minimum period of 50 hours.

52. The mounting bolts on the radome assembly be made more accessible.

53. All coaxial cables be properly marked or otherwise coded to eliminate mis-matching at connecting points.

54. An automatic frequency control be provided which will be more dependable in operation.

55. The contractor devise and provide a means of checking the arrival and sequence of the firing signal to each missile and rocket tube.

56. A rapid means of checking the operation of each relay within the armament control relay assembly separate from the armament system be devised.

57. The prime AMA and the contractor devise and provide a positive means of retaining sufficient snubbing air for safe launcher rail retraction.

☒ fire control system

58. All range gate servo units (P/N 463106-x120) be replaced immediately by the newer servo unit (-120).

59. A better means of protecting the Ledex units in the computer mode plug from dust and dirt be provided.

60. The diameter of the air intake filter to the radar pressurization control be increased from 10 microns to 40 microns and that the filter and all pressure points be cleaned after every 10 hours of operation.

61. The lead to the motor generator unit be redesigned to incorporate a cannon plug rather than a terminal strip arrangement.

62. Further testing of the target simulator be accomplished to determine its functional reliability and suitability.

63. A means of artificially supplying computer signals ("A" and "B") for use in range gate harmonization be designed.

64. A means of interrupting the range voltage to the range gate servo unit during range gate harmonization be provided.

65. The contractor immediately undertake a study of the radar in an effort to increase the radar range of the MG-3 radar set.

66. The contractor adopt more rigid quality control procedures for the magnetron and pulse-forming network sub-assemblies of the radar transmitter-receiver unit.

67. The method of mounting the missile interconnecting unit (P/N 403004-100) be changed to improve ease of access and replacement.

☐ missiles

68. The contractor initiate an investigation to determine and remedy the cause of missile hydraulic supply activation while loading missiles on the launcher rails.

69. A specific test be devised for checking missile batteries for leakage and that this test be incorporated in applicable technical orders.

70. The Chassis V unit be redesigned to reduce its high failure rate.

71. The hit probability of the Falcon missile be improved.

72. An evaluation be performed when possible to determine the functional reliability of the rocket firing system under actual firing conditions.

☐ ground support equipment

73. The clutch assembly on the MA-1 Joy Anti-Skid compressor be redesigned to permit more satisfactory and dependable operation.

74. A canopy safety bar be provided which, when placed in position, cannot become inadvertently dislodged.

75. An engine stand be provided which is satisfactory and dependable in all respects.

76. The MA-4 air conditioner be further tested in an effort to develop a more suitable and functionally reliable unit.

77. A universal-type coupling for the air hose on the MA-1 air starter be developed to permit the hose to be connected to aircraft other than the F-102A without the requirement to change couplings.

☐ operational improvements

78. A more satisfactory nose wheel steering system be provided.

79. A more satisfactory and less fatiguing nose wheel steering actuation switch be provided.

80. The brake pedal force system be redesigned to provide the pilot with a varying degree of pedal force and corresponding degree of actual braking action.

81. The lag in wheel brake release be removed.

82. An Anti-Skid feature be incorporated in the brake system.

83. Runway barrier engagement capability be improved.

84. A more suitable entrance ladder be provided.

85. A more durable anti-reflection curtain be provided.

86. A retractable-type glare shield be designed to permit its use without the existing requirement for exercising extreme caution when entering and leaving the cockpit.

87. Cockpit cooling during aircraft ground operation be considerably increased, or that a means of allowing the canopy to safely remain open for ground operation be devised and incorporated.

88. A more satisfactory means of successfully closing and locking the canopy be developed.

*89. An accelerometer be provided for the use of the pilot.

90. The afterburner nozzle position indicator be removed, or improved to the point that its intended purpose is fulfilled.

91. A directional indicating system be provided which will offer an accurate aircraft heading after turns of up to 75 degrees of bank.

92. The radio magnetic indicator be relocated to a less prominent position, and that a settable dial moving pointer indicator be utilized in its place.

93. The canopy latch handle be rotated 90 degrees to a horizontal position.

94. The drag chute deploy handle be provided with a catching device installed to prevent inadvertent forward movement of the control handle after operation.

95. The cockpit temperature control knob be replaced with a knob which can be more positively identified by touch.

96. The ram air turbine, emergency landing gear extension, drag chute, and canopy control handles be provided with identification lettering on the handles themselves.

97. The AN/ARC-34 channel selector be relocated to a position more accessible and more easily read. A remote channel indicator located on the forward instrument panel would be highly desirable.

*98. The radome anti-ice switch be redesigned to preclude the possibility of moving the switch into MANUAL ON by other than a deliberate movement.

99. The "Flight Mode" selector switch be replacarded with less ambiguous terminology.

100. The landing gear warning light be relocated to a point on the left side of the instrument panel more in line with the pilot's direct vision.

101. The landing gear warning light test switch be redesigned or relocated to positively prevent possible confusion with and inadvertent actuation of the emergency landing gear retraction switch.

102. The "DIMMED" level of intensity of the lights on the warning light panel be reduced to the greatest extent consistent with good legibility.

103. Landing lights with greater illumination intensity be provided.

*104. Formation lights which will define the extremities of the aircraft, including the tip of the vertical tail, be provided.

*105. The intensity of the aircraft running lights when in the "DIM" position be reduced to permit safer formation flying.

106. A more suitable emergency landing gear extension system be provided.

*107. Greater adjustment capability be provided in the MA-4 safety seat belt.

*108. A stronger latch assembly be provided for the MA-4 seat belt.

109. The anti-jam switch be relocated to the antenna hand control.

110. The action switch be relocated to a lower forward position on the antenna hand control and changed in shape to a bar-type switch.

111. The return to search button be relocated on front of the antenna hand control.

112. An altitude pressure suit be provided which will be compatible with the alert requirements of the Air Defense Command.

APPENDIX I

unsatisfactory reports

408 Unsatisfactory Reports were generated against the five test aircraft during the Phase VI program. The charts which follow represent the UR per flying hour ratio based on accumulated UR's and flying time. Chart No. 2 represents the Malfunction Report per flying hour ratio based on accumulated Malfunction Reports and flight time. It can be noted that during the last 300 flying hours, the UR per flying hour and malfunctions per flying hour trends appear to have stabilized at approximately 0.3. The second chart displays the number of Unsatisfactory Reports and the number of Malfunction Reports written per system during the program.

CHART NO. 7

AVERAGE NUMBER OF UNSATISFACTORY REPORTS
PER HOUR PER 50 HOUR FLYING PERIOD

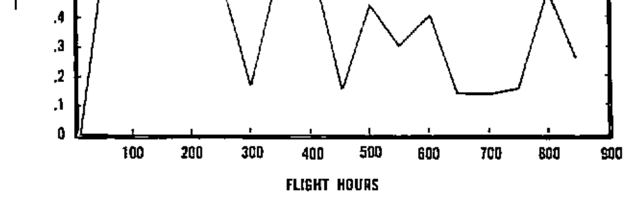


CHART NO. 8

AVERAGE NUMBER OF MALFUNCTION REPORTS
PER HOUR PER 50 HOUR FLYING PERIOD

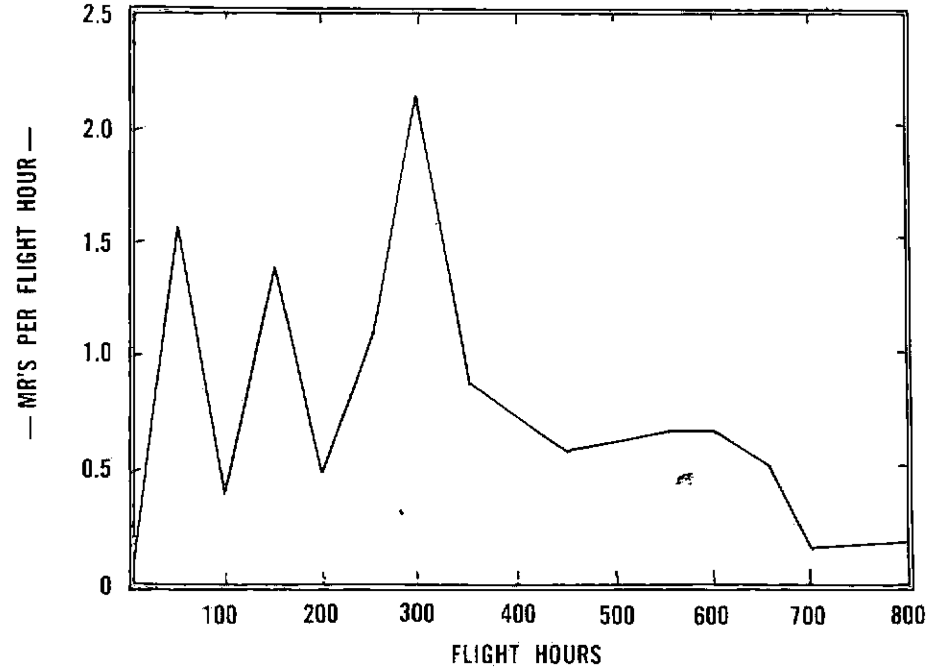
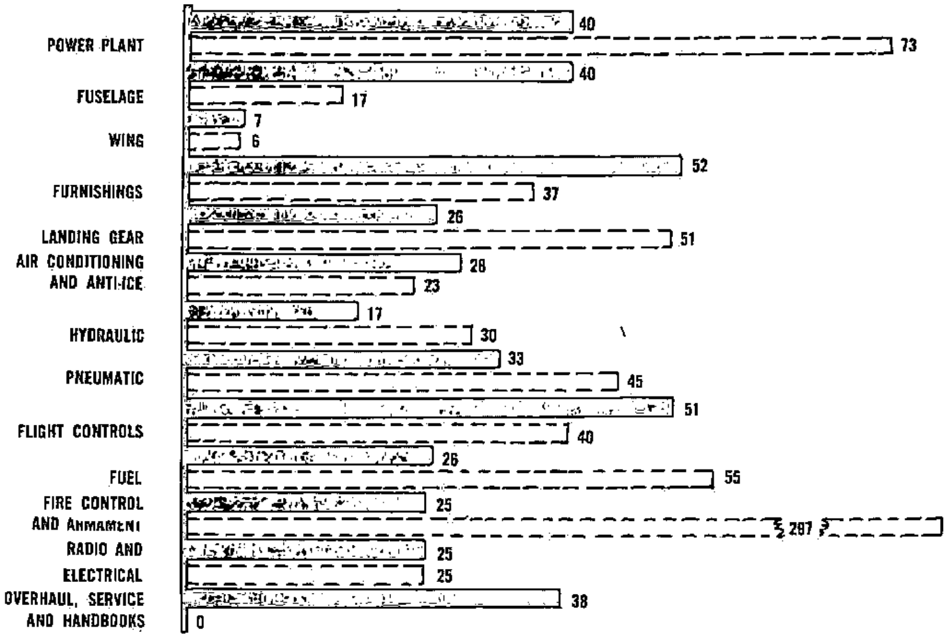


CHART NO. 9

UNSATISFACTORY/MALFUNCTION REPORTS PER SYSTEM



SYSTEM STATION S/N	DESCRIPTION
Power Plant	
56-49	Absence of warning placard constant speed (Sundstrand) drive.
56-165	Ruptured oil cooler duct assembly.
56-170	Silicon rubber gasket disintegrated in anti-ice valve.
56-208	MD-3 pneumatic starter failure.
56-253	Relocation of constant speed drive filter to provide accessibility.
56-264	Failed afterburner nozzle position indicator.
56-310	Replacement (spare) engine mount bushings not proper size.
56-342	Ruptured oil cooler duct assembly.
56-364	Ruptured oil cooler duct assembly.
56-405	Loose engine cooling shroud turnbuckle assembly.
56-421	Interference between oil cooler inlet duct and structure.
56-446	Plastic contamination in engine oil filter.
56-472	Hole in oil cooler duct assembly.
56-521	Interference between generator exhaust duct and constant speed drive oil tube.
56-528	Engine tail cone attach bolt problem.
56-539	Engine surge at high altitude and low power.
56-621	Warped afterburner nozzle weldments.
56-626	One-piece engine shroud installation requires engine and shroud removal for afterburner weldments and actuator access.
56-633	Engine installation problem.
56-649	Deficiency in afterburner closing action.
56-678	Warped afterburner nozzle weldments.
56-764	Leaking oil hose assembly.
56-767	Ruptured oil cooler duct assembly.
56-855	Engine surge at low altitude when accelerating from idle power.
56-888	Pneumatic starter leaking oil.
56-890	Engine cooling duct assembly chafing on on the floor of the engine compartment.
56-908	Bearing failure in Sunderstrand gear box.
56-916	Position and configuration of air valve is undesirable.
56-976	Inaccessible location of 6 cap screws on cover and filter element of Sundstrand drive assembly when installed on aircraft.
56-1036	Seized valve plate in Sundstrand drive assembly.
56-1040	Sundstrand interconnecting shaft separated from sleeve and yoke assembly.
56-1044	Defective fitted roller bearing in Sundstrand gear box.
56-1064	Engine mounted gear box not modified to permit interchangeability of starters.
56-1069	Acute bend in afterburner drain line.
56-1104	Failure of engine mounted gear box and and drive shaft.
56-1130	Buckling and cracking of combustion chamber weldments.
56-1135	Bleed valve governor bearing failure.
56-1159	Panel Assembly — Engine shroud adapter requires more than usual amount of sheet metal repair.
56-1160	Metal particles found in Sundstrand engine mounted gear box.
56-1170	Sundstrand drive alignment problem. No precision tool available to accomplish alignment.
Fuselage	
56-17	Broken tenite missile door stop gap assemblies.
56-17 (1)	Broken tenite missile door stop gap assemblies.
56-85	Lack of canopy handhold or lever.
56-105	Failed rivets — engine air inlet duct.
56-113	Failed rivets — engine air inlet duct.
56-143	Interference between canopy and auxiliary turbine access doors.
56-144	Interference between canopy and auxiliary turbine access doors.
56-145	Interference between canopy and auxiliary turbine access doors.
56-171	Removal difficulty, radar equipment rack.
56-236	Broken tenite missile door stop gap assemblies.
56-244	Worn radar door lock drive shaft assembly.
56-245	Removal difficulty, general control panel rack installation.

SYSTEM STATION S/N	DESCRIPTION
56-266	Lack of access to engine mounted components.
56-278	Interference between electronic access door and nose gear.
56-344	Cracked radar door lock plate bushing.
56-346	Radome attaching bolts are difficult to install.
56-372	Cracked radar door lock plate bushing.
56-373	Cracked radar door lock plate bushing.
56-374	Cracked radar door lock plate bushing.
56-375	Cracked radar door lock plate bushing.
56-426	Broken ram air turbine door hinge.
56-457	Canopy jettison system hoses being chafed by aileron stop bolts.
56-458	Canopy jettison system hoses being chafed by aileron stop bolts.
56-533	Canopy imposes taxiing restrictions.
56-579	Inadequacy of canopy counterbalance mechanism.
56-634	Inaccessibility of engine mounted components.
56-640	Broken landing light bracket.
56-666	Lack of access to elevon servo valves.
56-708	Lack of index pin holes in side slip transducer support assembly.
56-712	Ram air turbine door installation.
56-721	Broken landing light bracket.
56-743	Leaking canopy selector valve.
56-754	Broken landing light bracket.
56-759	Damaged radar access door seal.
56-777	Cracked tail cone fairing tip assembly.
56-782	Damaged radar access door seal.
56-850	Location of "T" shaped canopy latch handle obscures instruments.
56-901	Crack in the wing to fuselage attaching angle.
56-971	Left aft radar access door lost in flight.
56-1103	Weldment chafing launcher rail attaching link.
Wing	
56-182	Buckled outboard wing fence assembly.
56-228	Lack of screw gage for fuel tank access door.
56-237	Torn left-hand main gear fairing installation.
56-240	Broken right-hand landing light bracket.
56-341	Broken right-hand landing light bracket.

SYSTEM STATION S/N	DESCRIPTION
56-418	Broken right-hand landing light bracket.
56-609	Landing lights inadequate.
Furnishings	
56-71	Radio facility card holder catches hand when ram air turbine control handle is pulled.
56-86	Failed initiator hose fitting.
56-197	Malfunctioning fuel flow transmitter.
56-222	Lack of accelerometer in aircraft.
56-232	Malfunctioning turn and slip indicator.
56-246	Unsatisfactory fuel quantity indicator system capability.
56-249	False indications on turn and slip indicators.
56-256	Lack of identification on cockpit control handles.
56-280	Damaged screws in cockpit floodlight installation.
56-281	Damaged screws in cockpit floodlight installation.
56-283	Instrument panel glare shield cracked.
56-291	Afterburner position indicator inadequate.
56-299	Poor durability of windshield anti-reflection curtain.
56-313	Malfunctioning turn and slip indicator.
56-318	Leaking pressure ratio indicator case.
56-319	Leaking pressure ratio indicator case.
56-332	Failed air pressure accumulator gage.
56-358	Excessive manhour requirement for instrument removal from instrument panel.
56-380	Malfunctioning airspeed Mach meter indicator.
56-400	Missing ejection seat catapult hose fitting.
56-469	Ambiguous placarding of flight mode selector switch.
56-484	Interefrence right-hand pilot's panel and right-hand rudder pedal.
56-489	Interference right-hand pilot's panel and right-hand rudder pedal.
56-498	Radar scope glare shield inadequate.
56-509	Radio magnetic indicator presentation inadequate.
56-535	Location of cabin temperature control box.

SYSTEM STATION S/N	DESCRIPTION
56-542	Fuel quantity indicator failed to give accurate reading.
56-549	Interference pitot tube and locking handle on B-3 control panel.
56-625	Insufficient graduations on accumulator air pressure gage.
56-628	Stowage problem on safety pin in pilot's seat right-hand arm rest.
56-701	Broken armament selector knob.
56-762	Inoperative turn and slip indicator.
56-789	Location of rudder trim switch.
56-813	Erratic gyro compass operation.
56-835	Failed accumulator air pressure gage.
56-845	Airspeed indicator found to be faulty on delivery.
56-859	Need for modified altimeter to indicate the altitude below 16,000 feet.
56-860	Turn and slip indicator failure.
56-862	Turn and slip indicator failure.
56-910	Direction indicating system inadequate.
56-917	Design deficiency in the remote attitude indicator's "OFF" warning indicator.
56-918	Location of action switch.
56-919	Lap seat belt does not possess sufficient adjustment capability.
56-920	Anti-reflection curtain does not extend aft sufficiently to stop all glare.
56-967	Dirty connection indetector sensing system caused fire warning to light.
56-988	Bonded panel brackets on cockpit floor boards were abraded.
56-1010	Design deficiency of anti-ice switch.
56-1014	Improper position of ARC-34 channel selector.
56-1047	Oxygen regulator failed to deliver enough oxygen.
56-1080	Unsatisfactory length of pilot's oxygen hose.
56-1092	Insufficient warning light dimming.
Landing Gear	
56-115	Pitted wheel bearing cone and roller assembly.
56-120	Lack of adequate safety device to prevent retraction with aircraft on ground through landing gear emergency retract system.
56-199	Nose gear position indicator maladjusted by CVAC.
56-200	Spasmodic operation of nose wheel steering system.
56-297	Landing gear warning light installation is unsatisfactory.
56-301	Interference of nose wheel door and gear scissors.
56-308	Loose nut on nose gear torque link.
56-316	Leaking landing gear directional control valve.
56-368	Brake pedal feel forces are difficult to determine.
56-399	Excessive manhour requirement to replace main gear door closed switch.
56-504	Nose landing gear position switch installation.
56-590	Nose gear actuator Microswitch inoperative.
56-593	Main gear door closed switch setting slipped.
56-655	Pitted wheel bearing.
56-685	Gear down and locked harness caught by drag link.
56-686	Gear down and locked harness caught by drag link.
56-699	Excessive emergency landing gear extension time.
56-778	Cracked brake assembly piston.
56-816	Location of two upper axle attaching nuts on brake assembly inaccessible.
56-839	Failure of landing gear position indicator.
56-881	Running clearance was uneven on left brake.
56-983	Arrows on restrictor valve which indicate direction tube installed were backwards.
56-1012	Brakes tend to hold after release of brake pedal in normal operation.
56-1057	Gear indicator shows unsafe when right-hand main landing gear locked into position.
56-1150	Corroded main wheel bearings.
56-1151	Failure of main wheel bearings, retaining nuts, and tire.
Air Conditioning & Anti-Ice	
56-209	Refrigeration air cycle machine seized.
56-211	Cabin pressurization surge.

SYSTEM STATION S/N	DESCRIPTION
56-215	Cabin pressure rate sensor fitting incompatibility.
56-218	Failed air conditioning system control
56-407	Damaged anti-ice valve gasket.
56-415	Improperly installed cockpit air control switch.
56-436	Automatic cabin heat control box inoperative.
56-460	Inoperative cabin heat control box.
56-461	Inoperative cabin heat control box.
56-471	Broken flare, radome anti-ice tube.
56-499	Damaged anti-icing control valve seal.
56-520	Inoperative cabin heat control box.
56-766	Air cycle machine turbine seized.
56-769	Electronic equipment cooling duct chafing on bulkhead.
56-772	Inoperative caibn heat control.
56-818	Failed air cycle machine.
56-844	Malfunction of the ice interpreter assembly.
56-857	Windshield defogging system deficiency.
56-861	Defective radome anti-icing valve.
56-895	Oil leaking from air cycle machine.
56-982	Failure of duct from jet pump.
56-1011	Release of anti-ice fluid while anti-ice switch is in "OFF" position.
56-1166	Cabin thermostat would not sense an overheat condition.
56-1207	Electrically actuated air valve stuck in full open position.
Hydraulics	
56-213	Brake relay valve leaking.
56-263	Damaged fitting in nose gear door restrictor valve installation.
56-333	Collapsed secondary hydraulic system suction hose.
56-363	Leaking rudder actuator cylinder.
56-591	Leaking nose gear actuator assembly.
56-742	Failed hydraulic reservoir.
56-787	Excessive hydraulic pump pressure.
56-807	Failed hydraulic accumulator.
56-889	Hydraulic pump leak.
56-906	Nose landing gear steering hose assemblies were chafing on landing gear strut.

SYSTEM STATION S/N	DESCRIPTION
56-930	Small hose in hydraulic pump housing caused oil to leak profusely.
56-944	Failure of secondary hydraulic system accumulator pressure gage.
56-1016	Momentary failure of hydaulic solenoid control valve for nose landing gear.
56-1028	Slow internal leak in hydraulic accumulator.
56-1043	Incorrectly identified hydrualic pumps.
56-1045	Quick disconnect line oil leak.
56-1055	Engine oil breather line too easily damaged.
56-1134	Failure of primary pump shaft seal.
Pneumatics	
56-224	Design deficiency in starter air bleed duct cap.
56-225	Design deficiency in starter air bleed duct cap.
56-226	Design deficiency in starter air bleed duct cap.
56-227	Design deficiency in starter air bleed duct cap.
56-238	Broken armament bay pressurization valve.
56-270	Leaking canopy selector valve.
56-329	Ruptured cooling air check valve.
56-330	Damaged cooling air check valve.
56-350	No provisions for in-flight recharging of high pressure pneumatic system.
56-401	Failed pneumatic control regulator.
56-432	Failed pneumatic pressure regulator.
56-435	Failed pneumatic pressure regulator.
56-488	Ram air regulator out of tolerance.
56-665	Failed pneumatic pressure regulator.
56-669	Cracked cooling air check valve.
56-698	Failed pneumatic pressure regulator.
56-720	D-C generator exhaust duct interference with Sundstrand drive.
56-765	Malfunctioning air purge valve assembly.
56-570	Excessive pressure on air regulator assembly.
56-781	Leaking pneumatic relief valve.
56-768	High pressure pneumatic tubing interference with hydraulic tubing.
56-788	Leaking bleed air manifold assembly.

SYSTEM STATION S/N	DESCRIPTION
56-793	Cut canopy actuating pneumatic cylinder hose.
56-811	Cracked cooling air check valve.
56-865	Air leak from vent ports of pneumatic pressure regulator.
56-926	Air leak in pressure regulator.
56-938	Pneumatic starter had defective seal.
56-959	Air leak in pressure regulator.
56-1017	Pressure regulator leaking air from vent ports in regulator body.
56-1020	Time consuming process to inspect air filter due to position of clamp.
56-1062	Air escaping from vent ports in pneumatic pressure regulator.
56-1162	Seal failure in launcher selector valve.
Flight Controls	
56-150	Broken drag chute cable assembly.
56-181	Frayed drag chute assembly.
56-193	Loose control stick boot.
56-198	Failed drag parachute guide assembly.
56-204	Lack of emergency deployment provisions in drag chute.
56-212	Loose rivets in elevon.
56-214	Lack of "D" rings with spare drag chute.
56-241	Failed drag chute cable assembly.
56-269	Loose rivets in speed brake.
56-282	Damaged speed brake seals.
56-311	Nose wheel steering engage switch unsatisfactory.
56-328	Loose rivets in elevon.
56-339	Damaged pilot chute for drag parachute.
56-343	Loose rivets in elevon.
56-362	Damaged drag chute jettison.
56-464	Failed elevator trim actuator.
56-490	Failed elevator trim actuator.
56-542	Failed fuel quantity indicator.
56-543	Defective vertical gyro.
56-559	Damaged drag chute cable guide.
56-567	Failed elevon rivets.
56-599	Stripped threads in elevon pick-off potentiometer.
56-613	Speed brakes are ineffective.
56-635	Defective yaw rate gyro.
56-658	Absence of elevon rig pin holes.
56-666	Lack of access to elevon servo valves.
56-737	Cracked speed brake door.
56-763	Drag chute lost in flight.
56-779	Yaw rate gyro erratic.
56-817	Failed aileron trim actuator.
56-836	Unsatisfactory lateral control action.
56-842	Faulty design of drag chute jettison pin.
56-886	Loss of drag chutes due to strap assembly malfunction.
56-893	Drag chute connector link screws loosen too readily and misplace from links.
56-898	Elevon servo actuators inducing oscillations in flight surfaces.
56-927	Drag chute control handle tends to release drag chute voluntarily.
56-962	Elevon servo actuators cause elevator flutter when yaw and pitch systems engaged.
56-963	Design deficiency of location of aileron trim actuator motors.
56-974	Chafing of elevon on fuselage during maximum g loading in flight.
56-978	Elevon servo actuators cause elevator flutter when yaw and pitch systems engaged.
56-984	Failure of hermetically sealed roll and pitch control gyros.
56-1125	Failure of drag chute pilot chute retaining cone.
56-1126	Chafing of drag chute center risers.
56-1128	Charring of antenna elevation wires in control stick grip assembly.
56-1139	Rudder servo actuator leaking hydraulic fluid.
56-1147	Potentiometer incorporated in elevator trim actuator not operating properly.
56-1148	Failure of rudder control hose assembly.
56-1164	Elevon flutter encountered on actuating pitch and yaw system.
56-1209	Improper installation of drag chute retaining cone release cable.
56-1217	Defective airspeed computer.

SYSTEM STATION S/N	DESCRIPTION
Fuel System	
56-73	Leaking fuel vent pilot valve.
56-158	Cracked refueling cross over tube.
56-194	Inoperative fuel tank boost pump.
56-243	Improperly wired low level fuel warning light.
56-307	Interference between body refuel tube and landing gear bellcrank.
56-309	Interference between body refuel tube and landing gear bellcrank.
56-441	Defective fuel tank probe.
56-450	Malfunctioning fuel flow equalizer.
56-451	Malfunctioning fuel flow equalizer.
56-470	Metallic contamination in fuel flow level warning switch.
56-473	Accessibility of fuel flow equalizer mounting bolts.
56-515	Inoperative fuel boost pump.
56-542	Failed fuel quantity indicator.
56-588	Malfunctioning fuel flow equalizer.
56-612	Lack of fuel shut-off valve position indicator.
56-653	Failed fuel flow equalizer control pressure line (AVICO flex hose).
56-738	Failed fuel flow equalizer diaphragm.
56-751	Unsatisfactory fuel drain tubing installation.
56-806	Failed fuel flow equalizer diaphragm.
56-837	Failure of fuel flow equalizer diaphragm.
56-847	Failure of fuel flow equalizer diaphragm.
56-848	Failure of fuel flow equalizer due to disengaged valve components.
56-864	Low level warning switch inoperative.
56-1033	Outer surfaces of equalizer pistons severely galled on fuel flow equalizer.
56-1042	Punctured diaphragm in fuel flow equalizer.
56-1086	Punctured diaphragm in fuel flow equalizer.
Radar, Fire Control and Armament	
56-294	Radar operational range limit.
56-295	Undersized cotter keys in missile and armament installations.
56-345	Mounting of MG-3 interconnect box is unsatisfactory.
56-348	Excessive air leak through connector plug.
56-403	Radar interlock switch installation is unsatisfactory.
56-550	Additional maintenance due to radar interlock switches.
56-572	Relationship between rage gate control and indicator.
56-576	Intermittent connection causing elliptical scope image.
56-648	Lack of detent for antenna elevation control knob.
56-679	Antenna cable too short.
56-700	Missile doors would not close due to malfunctioning timer.
56-752	Radar door lock shaft too large.
56-774	Leaking missile launcher selector valve.
56-783	Bus bar not installed.
56-863	Air leak from vent port of selector valve.
56-882	Air leak from vent port of selector valve.
56-893	Lack of emergency missile door operating system.
56-949	Adjusting clamp on launcher down switch severely worn.
56-952	Inboard mounting screws inaccessible on missile interconnection box.
56-970	Pin failure in radar tuner assembly.
56-1018	Arcing in magnetron sub-assembly.
56-1031	Pin holdback on missile launcher severely bent.
56-1058	Pin holdback on missile severely bent.
56-1061	Launcher in No. 4 position contained 18 non-standard 8-32 screws.
56-1087	Missiles failed to fire because the No. 1 rocket switch failed.
Radio and Electrical	
56-42	Location of AN-ARC-34 receiver-transmitter.
56-47	Failed circuit breaker in missile door circuit.
56-111	Excessive cord in cockpit thunderstorm lighting installation.

SYSTEM STATION S/N	DESCRIPTION
56-220	Defective a-c control panel.
56-250	Landing gear warning light sensitivity.
56-340	Abraded electrical wires.
56-348	Air leak through connector plug.
56-351	Location of AN-ARC-34 receiver-transmitter.
56-354	Disconnected vertical coaxial cable from jack to UHF radio.
56-425	Broken electrical wires.
56-544	Voltmeter out of calibration.
56-545	Scale graduations of a-c voltmeter.
56-589	Defective a-c control panel.
56-598	Defective amplifier in J-2 compass system
56-609	Landing lights are inadequate.
56-784	Cockpit landing gear light inoperative due to defective pulsitometer.
56-869	No provision for emergency cockpit lighting.
56-894	Cockpit lighting control system design deficiency.
56-908	Bearing failure caused a-c power system failure.
56-943	Electrical fire in connector assembly, cockpit floor. Result of water leakage in cockpit.
56-969	Hydraulic low pressure warning light flasher relay failure.
56-987	No neoprene cover for 28 V terminal strip incorporated in manufacture.
56-1052	Oil in exciter commuter causing failure of alternators.
56-1072	Two alternator failures due to oil leaks on exciter commutator.

Overhaul and Service

SYSTEM STATION S/N	DESCRIPTION
56-160	Incomplete aircraft exterior identification markings.
56-172	Design deficiencies in engine removal stand.
56-195	MA-1 gas turbine compressor adapter incompatible with aircraft fitting.
56-219	Lack of ground safety lock for ram air turbine door.
56-262	Length of aircraft nose wheel tow bar unsatisfactory.
56-279	MA-1 compressor coupling not compatible with aircraft fitting.
56-290	Armament bay door ground safety lock damaging armament bay door.
56-298	Canopy wedge support safety lock is unsatisfactory.
56-327	Waper pump pliers left in engine inlet.
56-331	Flat file left beneath canopy cinchdown rocket arm.
56-393	Engine compartment work stand design deficiency.
56-433	Lubrication equipment not compatible with aircraft lubrication fittings.
56-746	Dust and dirt in cabin air.
56-912	Lack of lightweight chain hoist on apparatus now used to remove and replace radar receiver-transmitter.

Handbooks

SYSTEM STATION S/N	DESCRIPTION
56-61	T.O. 1F-102A-6, "Inspection of Engine Starter Oil Lever."
56-159	Aircraft delivered with obsolete technical order compliance records.
56-312	T.O. 1F-102A-6, "No Inspection Requirement for Engine Pressure Ratio Indicator System."
56-324	T.O. 1F-102A-6, erroneously directs removal of afterburner for periodic inspection.
56-367	T.O. 1F-102A-4 does not include engine cooling shroud strap attaching nuts.
56-404	T.O. 1F-102A-4 does not include class codes for Section III.
56-444	T.O. 13A5-10-4, "Seat Catapult Tube Elbow" not referenced.
56-445	T.O. 1F-102A-4, "Engine Ground Run Procedures" not adequate.
56-459	T.O. 1F-102A-6, no inspection requirements on control column mechanism beneath cockpit floor.
56-697	T.O. 1F-102A-518, "Replacement of Main Landing Gear Struts."
56-723	T.O. 1F-102-547, inspection procedures for AVICA flexible hose assemblies not adequate.
56-770	T.O. 1F-102-527, "Reactivate Sideslip Feedback and Deactivate Pilot Assist."
56-775	T.O. 1F-102-2-4, typrographical error in Figure 3-4.

SYSTEM STATION S/N	DESCRIPTION
56-812	T.O. 1F-102-554, "Inspection and Replacement of Rudder Control Cylinder and Valve."
56-870	Miscalculation of inspection stated in Work Card 1F-102A-6-PE.
56-900	T.O. 1F-102A-7, "Installation of the Rudder Trim Actuator" does not indicate a necessary precaution.
56-934	T.O. 1F-102A-2-4, "Installation of the Oil Cooler Inlet Duct Assembly" inadequate.
56-1003	T.O. 1F-102A-2-4 makes no provision for use of anti-personnel screens during engine run-ups.
56-1009	T.O. 1F-102A, Section III, "Engine Failure During Flight". Air Start procedure does not instruct the pilot to energize the emergency a-c generator after flame-out.
56-1032	T.O. 1F-102A-4 fails to disclose any reference listing of the fuel flow equalizer...
56-1037	T.O. 1F-102A-2-12 check-out procedure not complete for checking rigging of launchers.
56-1124	T.O. 1F-102A-2-568 fails to include a specific part number.
56-1144	T.O. 1F-102A-2-4 contains inadequate instructions for removal, installation, and servicing of pneumatic starter.
56-1233	T.O. 1F-102A-2-4 contains insufficient information on torque values for afterburner mating flange bolts.

APPENDIX II

parts consumption

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
01-B	1ABH-C7932	Hose Assembly	1
	1ABH-C8372	Hose Assembly	2
	1ABH-C8422	Hose Assembly	1
	1ABH-PRP6230-10	Gasket	2
	1ABH-R306-162-125	Line Assembly	4
	1ABH-Q8H756-4	Washer	2
	1ABH-10006-3	Duct Assembly	2
	1ABH-10006-5	Duct Assembly	2
	1ABH-12882	Valve	6
	1ABH-22053	Mount Assembly	1
	1ABH-28108	Cover Assembly	1
	1ABH-28109	Cover Assembly	1
	1ABH-4105-5	Line	1
	1ABH-601000	Hose Assembly	2
	1ABH-8-02392	Tube	1
	1ABH-8-04330-1	Control	1
	1ABH-8-07317-11	Seal	1
	1ABH-8-10095	Duct	1
	1ABH-8-103080-1	Valve	2
	1ABH-8-17501-1	Fitting	1
	1ABH-8-17052-1	Fitting	1
	1ABH-8-17619-1	Door	1
	1ABH-8-17619-72	Fairing Assembly	1
	1ABH-8-22030	Bushing	2
	1ABH-8-22033-9	Bolt	2
	1ABH-8-22050-7	Ring	2
	1ABH-8-22051	Bolt	2
	1ABH-8-22054	Washer	2
	1ABH-8-22057	Barrel	1
	1ABH-8-22642	Barrel	2
	1ABH-8-22652-11	Eye	11
	1ABH-8-22652-13	Eye	7
	1ABH-8-22716	Valve	2
	1ABH-8-22905-17	Cable	1
	1ABH-8-22905-19	Cable	1
	1ABH-8-22914	Fitting	1
	1ABH-8-27013-7	Trunnion	2
	1ABH-8-27401-17	Tube	1
	1ABH-8-27401-45	Tube Assembly	1
	1ABH-8-27401-47	Tube Assembly	1
	1ABH-8-27416	Screen Assembly	2
	1ABH-8-27807	Nipple	1
	1ABH-8-27808	Cap	1
	1ABH-8-27952-7	Duct Assembly	1
	1ABH-8-38030-21	Seal	2
	1ABH-8-382022-001	Amplifier	1

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
01-B	1ABH-8-43108	Tube	1
	1ABH-8-44104	Bell Crank	1
	1ABH-8-44108	Guide	1
	1ABH-8-44115	Spring	1
	1ABH-8-44435	Guide	2
	1ABH-8-44436	Release	1
	1ABH-8-44454-9	Spacer	36
	1ABH-8-44456	"O" Ring	12
	1ABH-8-44456-9	Ring	24
	1ABH-8-44498	Pin	43
	1ABH-8-45359	Cable	39
	1ABH-8-45935-21	Door Assembly	6
	1ABH-8-45939	Shaft	7
	1ABH-8-49120-17	Pin	1
	1ABH-8-49120-19	Pin	2
	1ABH-8-49380-1	Spring Assembly	1
	1ABH-8-49934	Bell Crank	1
	1ABH-8-54469	Fitting	2
	1ABH-8-56446	Frame Assembly	1
	1ABH-8-56612-3	Door Assembly	1
	1ABH-8-64263-1	Knob	3
	1ABH-8-64493	Connector	1
	1ABH-8-65252	Bolt	3
	1ABH-8-65252-9	Bolt	2
	1ABH-8-65254	Spring	1
	1ABH-8-70600-49	Insert	1
	1ABH-8-70600-51	Insert	1
	1ABH-8-70649-11	Seal	2
	1ABH-8-70665	Handle Assembly	2
	1ABH-8-70962-7	Seal Assembly	6
	1ABH-8-71616-17	Fitting	1
	1ABH-8-71616-29	Fitting	1
	1ABH-8-71625-7	Seal	2
	1ABH-8-71625-9	Seal	2
	1ABH-8-71626-11	Bushing	1
	1ABH-8-71641-7	Seal	2
	1ABH-8-72601-41	Seal	2
	1ABH-8-73783-1	Support	1
	1ABH-8-73940	Seal	1
	1ABH-8-87059-7	Blanket	1
	1ABH-8-87059-9	Blanket	1
	1ABH-8-87126	Duct Assembly	1
	1ABH-8-87141	Manifold	1
	1ABH-8-87424	Cap Assembly	2
	1ABH-8-89231	Elbow	1
	1ABH-8-94930	Curtain Assembly	1
	1ABH-8-94613	Shield Assembly	5
	1ABH-18-21665-41	Blanket	3
02H1	0245-1577872	Tab Washer	3
	0245-183668	Seal-Finewell	181
	0245-185246	Seal	14
	0245-185248	Seal	14
	0245-186948	Gasket	15
	0245-190399-9	Fuel Control	1
	0245-193920	Tab Washer	64
	0245-194302	Filter, Afterburner Igniter Valve	2
	0245-195711	Tab Washer	330
	0245-196447	Seal	2

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
02H1	0245-196448	Seal, Front Accessory Case	4
	0245-199062	Seal, "O" Ring	42
	0245-199063	"O" Ring	6
	0245-199065	"O" Ring	7
	0245-199399-9	Fuel Control	1
	0245-199786	Seal, "O" Ring	19
	0245-202199	Gasket	2
	0245-203189	Nut	2
	0245-206047	Tab Washer	20
	0245-207011	Clip	1
	0245-209375	Seal	2
	0245-210301	Packing	1
	0245-213547	Seal	1
	0245-215776	Tube	1
	0245-215778	Tube	1
	0245-216042	Screen Assembly	1
	0245-218406	Seal	3
	0245-220037	Gasket	26
	0245-220038	Gasket	20
	0245-220532	Pump Assembly (Oil)	9
	0245-224466	Bolt	1
	0245-228118	Clip	20
	0245-229100	Cooler	2
	0245-229379	Bolt	8
	0245-230517	Oil Pressure Relief Valve	1
	0245-235201	Blade 1st Stage Turbine	113
	0245-237376	Chamber Weldments	14
	0245-237377	Chamber Weldments	17
	0245-237378	Chamber Weldments	6
	0245-237379	Chamber Weldments	4
	0245-242707	Nozzle Weldments	10
	0245-24604C	Afterburner Eyelid Segments	4
	0245-251-976	Igniter Valve	1
	0245-251976D	Bolt	4
	0245-25550	Control Assembly Comp. Bleed	2
	0245-256328	Screw	34
	0245-256943	Rivet 1st Stage Turbine	113
	0245-259444	Tube	1
	0245-26082G	Control	1
	0245-261936	Nozzle Segment Afterburner	1
	0245-271977-3	Seal	2
	0245-271977-15	Seal	2
	0245-275620	Fairing Welding Afterburner	20
	0245-279499	Afterburner Igniter Valve	2
	0245-279499-T	Valve Assembly	3
	0245-42342	Seal	2
	0245-589144-7	Tab Washer	2
	0245-589146-57	Seal	1
	0245-754-749-9827	Seal	1
	0245-754-749-9855	Seal	1
	1245-6-1253	Gasket	3
03-B	4103-148320M-STA	Brake Assembly	8
	4111-3-667M1	Wheel	2
	4138-13140-1	Valve	1
03-C	4200-A-12A	Generator	1
	4200-A12B	Generator	1

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
03C	4200-954-749-985	Oil Filter — "O" Ring	1
	420-224-54G10	Detector	3
	4211-B5325-24	Light	1
	4221-B5325	Lamp	3
	4246-A35A9-29	Panel	4
	4246-A359129-2	Panel	1
	4246-A35A9130-3	Panel Assembly	1
	4246-A40A1713	Regulator	1
	4246-A40A1735	Panel	3
	4246-A40A1735-2	Panel	1
	4246-A40A1750	Regulator	4
	4246-A40A2070	Base	1
	4246-A50J206	Generator	1
	4246-A50J207	Generator	1
	4248-T-173A	Flasher	4
	4248-T-196A	Flasher	1
	4248-T-198A	Flasher	3
	4248-24530148	Box Control	1
	4248-24530148	Box	6
	4248-430148	Box	1
	9446-383T100	Actuator	8
	9446-53TD100	Actuator	1
	9446-53TD100-401	Actuator	1
	9446-53TD100-451	Actuator	2
03-D	4305-317816	Seal	2
03-F	4502-T.O.13A1-1-508	Kit	1
	4502-52B3687-1	Hose	2
	4502-53J20940-2	Belt	1
	4502-54H19604	Belt	1
	4504-103080-1	Valve	1
	4519-59FH1032	Nut	100
	4519-10006-3	Duct Assembly	1
	4519-10095	Tube Assembly	2
	4577-12501-3	Refrigerator (Package)	4
	4577-12502-5	Turbine	2
	4577-12520-3	Duct Assembly	1
	4577-12523-2	Gasket	1
	4577-12528-3	Sensor	8
03-G	4600-10797	Cylinder	1
	4600-20500	Cylinder	1
	4606-A17394-3	Damper	2
	4626-538100-502	Strut Assembly	2
	4626-60012	Cylinder	2
	4646-13014DOC	Cylinder	2
	4646-60010-3	Damper	1
03-I	4800-2B16.75	Nose Assembly	1
	4800-251976D	Afterburner Igniter Valve	4
	4801-50808	Valve	2
	4804-AN6235-1A	Element	1
	4805-105242-300-1	Valve	7
	4805-4405-759-S	Clamp	4
	4806-10640	Valve	4
	4829-MC 1613	Valve	1
	4831-1356-512463	Accumulator	3
	4834-2151-1	Regulator	5
	4837-64-1049-3	Pump	4
	4839-59330	Starter	1
	4839-TV114800-1	Regulator	2
	4841-AA60658L	Pump Assembly	1

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
03-I	4841-AA60658LA	Pump	6
	4841-AA60658L-2	Pump Assembly	5
	4846-147-26001	Valve	1
	4846-247-26001	Valve	1
	4846-IN24-4	Control	1
	4846-10095	Tube Assembly	1
	4846-12810	Valve	1
	4846-459	Ring	20
	4846-4574	Screw	20
	4849-5152	Valve	1
	4851-36584-1	Filter	1
	4853-2-127-001	Valve	2
	4853-A40092	Reservoir	4
	4853-A40172	Regulator	6
	4853-A50099	Regulator	6
	4860-AV1B1122	Valve	1
	4868-15552-4	Coupling	1
	4868-15554-12	Coupling	1
	4876-M634810	Transmission	1
	4876-634810A	Gear Assembly	2
	4876-634828	Drive Assembly	2
	4876-634838	Cover	1
	4876-657140	Gear Assembly	3
	4879-55625-NL	Valve	2
	4880-9-6541	Valve	1
	4890-940472	Valve	2
	4890-870447	Valve	2
03-J	4939-59330	Starter	7
03-K	5509-AN6003-36A	Tube Assembly	6
	5542-2889-6A-A1	Regulator	5
	5559-870337	Cylinder Assembly	2
05-A	6234-7204-1A4A1	Indicator	1
	6240-664543	Transmitter	2
	6242-664543	Transmitter	1
	6258-673447	Gyro	2
05-C	6001-30400	Detector	1
	6001-804542	Transducer	2
	6003-AN5813-1	Tube	2
	6003-AN5816	Tube	1
	6003-AN5816-2	Tube	1
	6025-1846X-4-03	Altimeter	1
	6025-1952-4-03	Indicator	2
	6025-2206X4-03	Altimeter	1
	6034-1440-55C-1	Indicator	3
	6034-1440-55D-C55-1	Indicator	3
	6034-1636-6GB-1	Indicator	1
	6034-3419-5A-A1	Accelerometer	1
	6034-3911-1Q-A1-1	Indicator	5
	6053-A1050	Indicator	5
	6062-105914-01	Control	1
05-D	6125-1985-4-04	Indicator	1
	6136-31960	Gage	1
05-F	2206-1706C1	Actuator	4
	2206-1706C2	Actuator	6
	2206-1708D	Actuator	2
	2232-EG55F-1	Amplifier	2
	2232-FG55D1	Amplifier	1
	2232-GG13A-19	Gyro	4
	2232-GG13A-21	Gyro	5

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
05-F	2232-JG7044A16	Gyro	1
	2232-JG7044A39	Gyro	2
	2232-Q629B-1	Amplifier	2
	2240-653822	Amplifier	1
05-G	2343-AW1-14DD	Gage	5
	2343-SP103-47	Indicator	1
	2346-1337243	Voltmeter	1
	2357-381056A054	Transmitter	1
	2357-381056A0653	Transmitter	1
	2357-382022-001	Amplifier	2
	2357-383011-A191	Indicator	1
05-H	0501-SP103-57	Positioner	1
07	4300-026500	Cement	2 Qt.
	4300-402450	Paint	8 Qt.
	4300-780250	Primer	2 Gal.
	4300-899902	Thinner	2 Qt.
08-B	8800-PPM6397	Jar Battery	1
	8800-429220	Tape	9 rolls
08-C	8880-055000-5155	Clip	3
	8880-058000-5151	Clip	2
	8880-304000-2855	Connector	7
	8880-739000-5975	Terminal	500
	880-984542-3	Transformer	1
08-E	8850-100915	Clamp	20
	8850-184500-255	Fuse	7
	8850-337600-54	Connector	1
	8850-621280	Connector Receptical	1
	8850-874690	Jack	1
	8850-984542-3	Transformer	1
08-F	8860-404000-21	Cable	350 Ft.
	8860-898005-225	Wire	1500 Ft.
08-G	8870-112000-758	Fuse	4
	8870-136000-455	Fuse	6
08-G	8870-136000-535	Fuse	3
	8870-144000-525	Fuse	32
	8870-184500-255	Fuse	7
	8870-184500-325	Fuse	1
	8870-60054-1012	Brush Holders	6
	8870-728000-255	Lamp	5
	8870-852000-775	Lamp	20
	8870-87000-6155	Insulator, Stand Off	1
	8870-936000-555	Lamp	7
11-B	5200-00450 (Convair)	Intervalometer	1
	5200-008956-47	Amp. Elec., Control	17
	5200-008956-88	Control Transmitter	2
	5200-008957-55	Servo Roll	1
	5200-011483-5	Antenna	4
	5200-011583-1	Antenna	1
	5200-011583-5	Antenna	1
	5200-108801 (Mfg. No.)	Angle of Attack Positioner	1
	5200-203251-5	Comp., Air Navigation	3
	5200-203252	Comp., Alt., Diff.	2
	5200-215988	Control Assembly	2
	5200-22135	Controller	2
	5200-221352-3	Control Computer	1
	5200-221353-3	Missile Launcher	5
	5200-221380	Control, Roll-Pitch	2
	5200-221383-3	Control, Roll-Pitch	4
	5200-2213835	Control, Roll-Pitch	3

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
11-B	5200-222252-14	Directional Coupler	1
	5200-222253-252	Directional Coupler	1
	5200-275645	Control	1
	5200-374426	Flight Command Indicator	8
	5200-378027-5	Box Assembly	1
	5200-395010	Launcher Rail	10
	5200-401862 (Mfg. No.)	Book #5	1
	5200-403080-110	Flight Control Indicator	1
	5200-403621	Switch Assembly, Limit	1
	5200-408044 (Mfg. No.)	Terminal Board	1
	5200-408584 (Mfg. No.)	Glass Observation	1
	5200-409922	Clamshell Assembly	2
	5200-409923	Clamshell Assembly	2
	5200-411150 (Mfg. No.)	Controller	2
	5200-412843 (Mfg. No.)	Harness, Elec.	1
	5200-415664 (Mfg. No.)	Amplifier	2
	5200-423128-110	Comp. Air Navigation	1
	5200-423187-120	Mixer, Frequency	1
	5200-42922	Transducer	1
	5200-435-27-2001	Control, Roll-Pitch	1
	5200-463002-100 (Mfg. No.)	Brush	6
	5200-463054-113	Launcher, Rocket Airborne	7
	5200-463017-110 (Mfg. No.)	Resolver	1
	5200-464590	Mixer Frequency	1
	5200-465676	PFN Magnetron	4
	5200-47068	Network Pulse Forming	1
	5200-628630-52	Receiver Transmitter	2
	5200-632705	Voltage Regulator	9
	5200-632779-5	Voltage Regulator	1
	5200-633465	Relay, Assy, Armt. Cont.	4
	*5200-650419-5 (Mfg. No.)	Positioner, Mach	1
	*5200-650376 (Mfg. No.)	Switch	1
	*5200-650-378 (Mfg. No.)	Valve Isabaric	1
	5200-650419-6 (Mfg. No.)	Servo, True Air Speed	1
	5200-650-419-7 (Mfg. No.)	Actuator, Rotary	1
	5200-650419-8 (Mfg. No.)	Positioner, Angle of Attack	1
	5200-650419-9 (Mfg. No.)	Positioner, Jump Angle	1
	5200-650442 (Mfg. No.)	Waveguide Flexible	1
	5200-731760	Control, Antenna, Scan	1
	5200-804542 (Mfg. No.)	Arm, Relative Wind	1
	5200-867723	Synchronizer, Electronic	2
	5200-888375	Synchronizer, Aux.	2
	5200-923853	Test Set Comp.	1
	5200-924300	Gyro, Rate	1
	5200-940504-1 (Mfg. No.)	Motor	1
	5200-96092-16	Grip, Flt Control	1
16-A	1600-011270020	Control Radio	3
	1600-013424080	Receiver	2
16-0	3330-057253084	Capacitor	1
	3330-057253245	Capacitor, Fixed	1
16-P	3340-062450-340	Coil	2
	3340-603905220	Transformer	1
	3340-650272-3 (Mfg. No.)	Transformer	4
	3340-98454 (Mfg. No.)	Transformer	1
	3340-984542-3 (Mfg. No.)	Transformer	4
	3340-987520-1 (Mfg. No.)	Transformer	2
16-Q	3350-057251554	Capacitor	2
	3350-057253245	Resistor, Fixed	1
	3350-098000-22	Resistor, Fixed	1

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
16-Q	3350-098000-513	Resistor, Fixed	1
	3350-098000-2271	Resistor	1
	3350-098000-3331	Resistor, Fixed	3
	3350-098000-5451	Resistor, Fixed	1
	3350-128000-2215	Resistor	1
	3350-128000-3515	Resistor	3
	3350-128000-3575	Resistor	6
	3350-128000-7115	Resistor	2
	3350-146000-1857	Resistor	8
	3350-230100-211	Resistor	1
	3350-460500-6 (Mfg. No.)	Resistor	1
	3350-793000-4651	Resistor	3
	3350-840010-1311	Resistor	1
	3350-954007-1 (Mfg. No.)	Resistor	1
	3350-960902-16 (Mfg. No.)	Resistor	1
16-R	3360-A6-22	Switch Assembly	1
	3360-F7890	Switch	1
	3360-H-6-21	Switch	1
	3360-H-6-22	Switch	1
	3360-H-6-24	Switch	1
	3360-H-6-27	Switch	1
	3360-SG 12A2	Switch	1
	3360-0705000519	Circuit	2
	3360-073414180	Switch	1
	3360-074032030	Switch	1
	3360-10821-A	Switch	1
	3360-11075A9-7	Switch	2
	3360-18420-1	Switch	1
	3360-25423409	Thermostat	2
	3360-33500-1	Switch	1
	3360-4151-12B39	Switch	1
	3360-417A8DBL-94	Switch	1
	3360-575-112	Switch Assembly	1
16-R	3360-650-4171	Switch	2
	3360-650419-7 (Mfg. No.)	Actuator, Rotary	2
	3360-96092-16 (Mfg. No.)	R-2 Hand Control	1
	3360-973503-36 (Mfg. No.)	Switch	1
	3360-973514 (Mfg. No.)	Rotary Switch	1
	3360-974000-2 (Mfg. No.)	Switch	2
16-S	3370-212000-2315	Crystal	14
	3370-212000-2317	Crystal	1
	3370-259000-6135	Electron Tube	7
	3370-262000-6	Tube	1
	3370-262000-6275	Vacuum Tube	5
	3370-265000-5814	Tube	2
	3370-265000-5815	Electron Tube	3
	3370-265000-5816	Electron Tube	2
	3370-265000-5860	Vacuum Tube	1
	3370-274000-2245	Tube	5
	3370-286000-5645	Tube	8
	3370-298000-6475	Electron Tube	12
	3370-299000-5685	Tube	1
	3370-316000-655	Tube	1
	3370-316000-1155	Vacuum Tube	5
	3370-316000-1367	Vacuum Tube	12
	3370-316000-5677	Resistor	5
	3370-316000-5717	Vacuum Tube	4
	3370-316000-5825	Electron Tube	4
	3370-316000-5827	Vacuum Tube	7

CLASS	STOCK NUMBER	NOMENCLATURE	QUANTITY
16-S	3370-316000-6655	Tube	1
	3370-331000-1523	Electron Tube	1
	3370-331000-6175	Electron Tube	3
	3370-421500-6249	Electron Tube	2
	3370-436000-7125	Tube	3
	3370-4480000-5135	Tube	2
	3370-451000-6145	Tube	3
	3370-6873000-3222	Crystal	3
	3370-689000-1331	Crystal	5
	3370-87000-6155		1
	3370-90292-1 (Mfg. No.)	Tube	1
	3370-903158 (Mfg. No.)	Tube	1
	3370-90386-3 (Mfg. No.)	Tube	2
	3370-92006-2 (Mfg. No.)	Cathode Ray Tube	3
	3370-928006-1	Electron Tube	4
	3370-928006-2	Electron Tube	2
	3370-928008-1	Vacuum Tube	4
	3370-949102 (Mfg. No.)	Crystal	2
	3370-949102-14 (Mfg. No.)	Crystal	1
	3370-949102-17 (Mfg. No.)	Crystal	12
	3370-949107-7 (Mfg. No.)	Crystal	1
	3370-949114-1	Vacuum Tube	3
16-T	3380-CPL2SL2	Relay	4
	3380-MAX-4004	Relay	2
	3380-RY416B	Relay	1
	3380-T-2025	Relay	1
	3380-415661 (Mfg. No.)	Relay	1
	3380-511120-3995	Relay	2
	3380-512325-4353	Relay	1
	3380-5770C-513	Relay	2
	3380-575225-2385	Relay Assembly	6
	3380-8300-300	Relay	2
	3380-950012-2 (Mfg. No.)	Relay	1
16-T	3380-950021-1 (Mfg. No.)	Relay Assembly	22
	3380-951600-10 (Mfg. No.)	Switch	1
	3380-974000-4 (Mfg. No.)	Switch	2
19-A	8200-SE0893	Adapter	4
20-B	2010-B4135-1	Bridles Assembly	10
	2010-C4121	Streamer Assembly	24
	2010-C8017	Drag Chute	43
	2010-D2016	Riser Assembly	28
	2010-E2137	Drag Chute	8
	2010-E8016	Bag	20
23-A	6800-185000	Wire	2 lb.
	6800-238200	Wire	1 lb.
	6800-29600	Wire	1 lb.
	6800-388000	Wire	2 lb.
	6800-785000	Wire	1 lb.
	6800-786000	Wire	1 lb.
	6800-787000	Wire	1 lb.
	6800-797000	Wire	1 lb.
	6800-906569	Wire	6 lb.
24	8500-283000	Cartridge	24
29	6700-PDL-4265-156D	Clamp Assembly	1
	6700-015519	Coupling	1
	6700-119900	Screw	1
	6700-17189-200C	Gasket	12
	6700-191613	Coupling	2
	6700-194361-734	Clamp	12

	STOCK NUMBER	NOMENCLATURE	QUANTITY
29	6700-194380	Clamp Assembly	1
	6700-197000	Paper	1
	6700-256942	Rivet	85
	6700-256943	Rivet	113
	6700-256944	Rivet	79
	6700-264550	Gasket	12
	6700-613925-319	Screw	25
	6700-625606-628	Screw	10
	6700-625606-924	Screw	10
	6700-634375-43216	Screw	1
	6700-634375-5612	Screw	1
	6700-6362273-27	Screw	6
	6700-636385-324	Screw	16
	6700-636385-5614	Screw	10
	6700-636385-55912	Screw	50
	6700-636385-55913	Screw	45
	6700-636385-55914	Screw	155
	6700-667695	Screw	4
	6700-667950	Screw	14
	6700-673350	Screw	3
	6700-708150	Screw	10
	6700-708550	Screw	14
	6700-708560	Screw	24
	6700-785542-24	Screw	100
	6700-840647	Coupling	4
	6700-840659	Coupling	24
51-I	6ZAA-624000	Pump	1
Federal	2620-269-7635	Casing	13
	2620-269-7651	Casing	6
	3110-14204316	Bearing	1
	4720-C7693B	Hose Assembly	8
	4720-C7761	Hose Assembly	1
	4720-98045	Hose Assembly	1
	4760-R3806-16Z-12.5	Hose Assembly	3
	6135-120-1020	Battery	91
	6220-025-7722A	Light Assembly	2
	9150-223-4003	Grease	1 lb.
	9150-223-4134	Hydraulic Oil	74 Gals.
	9150-252-6383	Hydraulic Oil	6 Gals.
	9150-257-5361-3	Grease	2 lbs.
	9150-261-8298	Grease	3 lbs.
	9150-263-3490	Hydraulic Oil	1 Qt.

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Air Force Flight Test Center
Flight Test Engineering Division
Edwards Air Force Base, California
(Title Unclassified) F-102A PHASE
VI FUNCTIONAL DEVELOPMENT REPORT,
by R. L. Stephens, Major, USAF,
R. D. Cousins, Captain, USAF, R. M.
White, Captain, USAF. January 1957.
88p. AFFTC-TR-56-34

The F-102A aircraft has the potential to furnish the Air Defense Command with an excellent weapon system. The handling characteristics of the aircraft are exceptional, but demonstrated deficiencies such as engine compressor surge, afterburner blowout and engine over-temperature at high altitude, and low kill probability of the armament system limit the operational capability of the aircraft.

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F-102A

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PHASE IV STABILITY TEST

J. REVERDY ALLENDER
Project Engineer

ROBERT M. WHITE
Captain, USAF
Project Pilot

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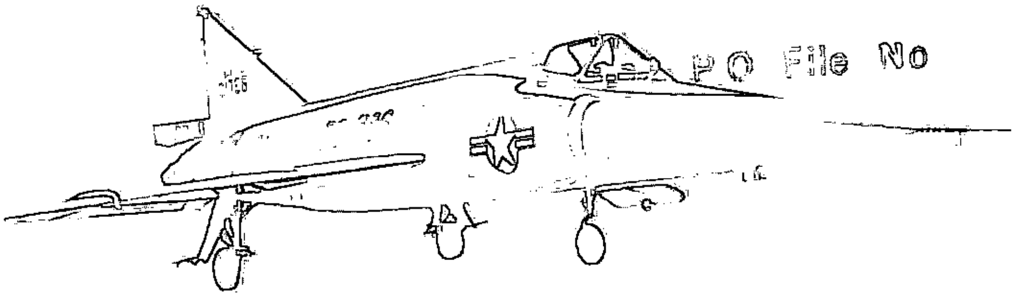

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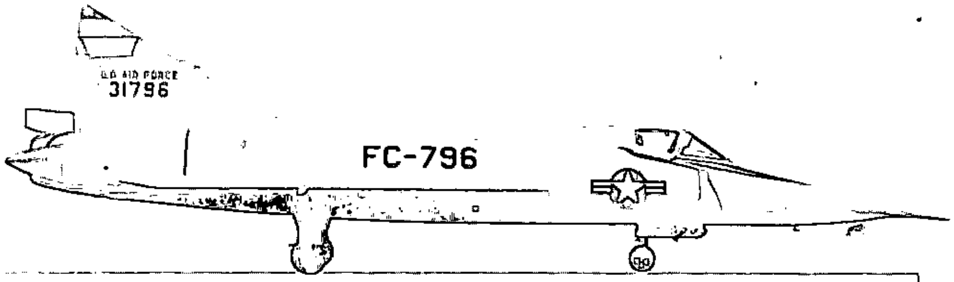


F-102A
PHASE IV
STABILITY TEST

AFFTC-TR-57-5 ■ April 1957

ASTIA Document No. AD 118708

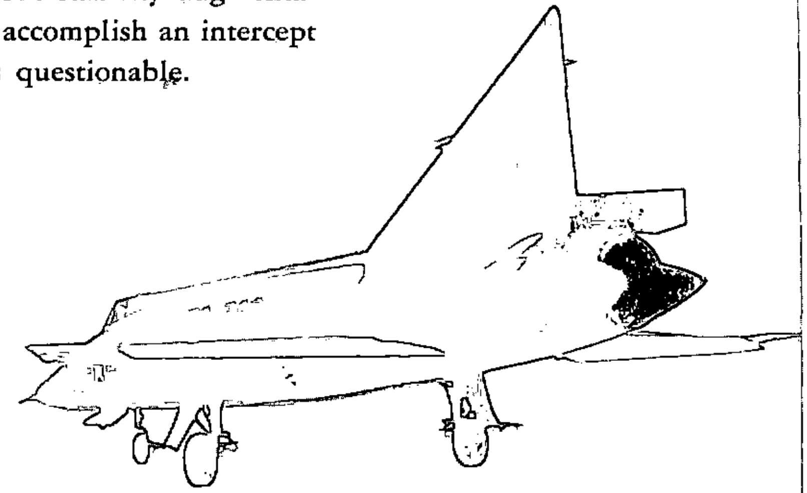
J. Reverdy Allender, Project Engineer

Robert M. White, Captain, USAF, Project Pilot

147 Copies, F.M.S. Studios, Pasadena, Calif. 8 April 1957

ABSTRACT

The F-102A has no stability and control deficiencies which seriously affect its all-weather intercept capability as long as the stability augmentors are operating. The aircraft can be flown safely without stability augmentation, but its ability to successfully accomplish an intercept mission under these conditions is questionable.



This report has been reviewed and approved/**19 MARCH 1957**

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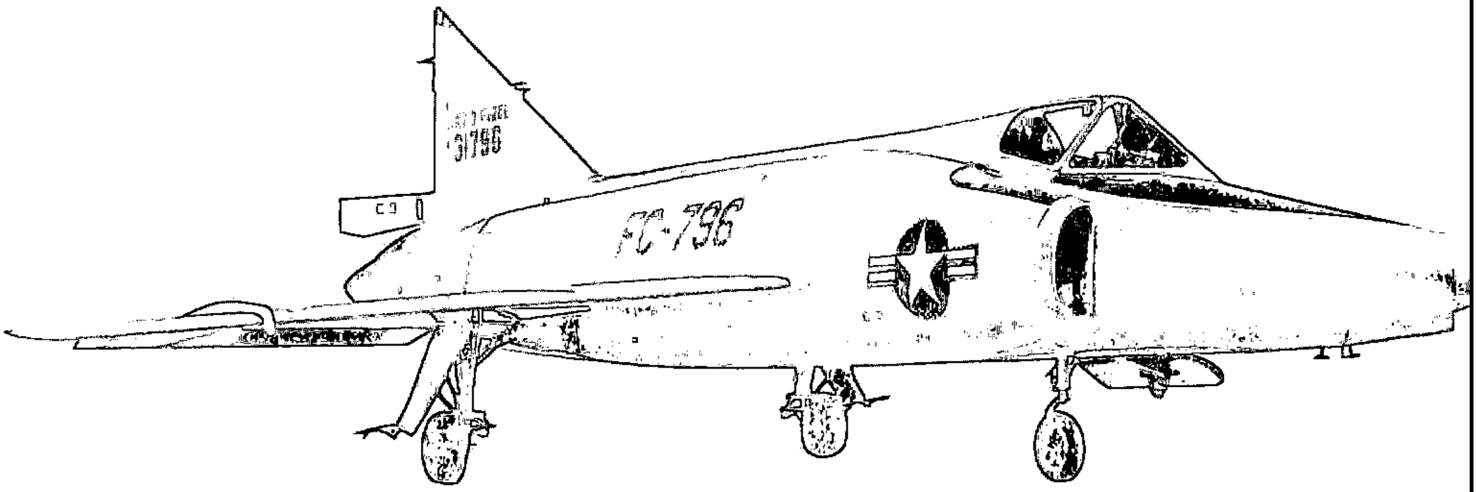
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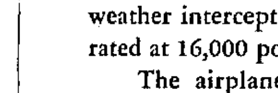
INTRODUCTION

Phase IV stability and control flight tests were conducted on F-102A USAF S/N 53-1796 to obtain data for the Pilot's Handbook of Operating Instructions and to determine compliance with the flying requirements of Military Specification MIL-F-8785 (ASG).



The tests were performed at the Air Force Flight Test Center, Edwards Air Force Base, between 19 April and 14 September 1956. The program consisted of 41 flights, requiring 38 hours and 55 minutes.

The F-102A is a single-place, high altitude, all weather interceptor powered by a J57-P-23 jet engine rated at 16,000 pounds static thrust with afterburning.

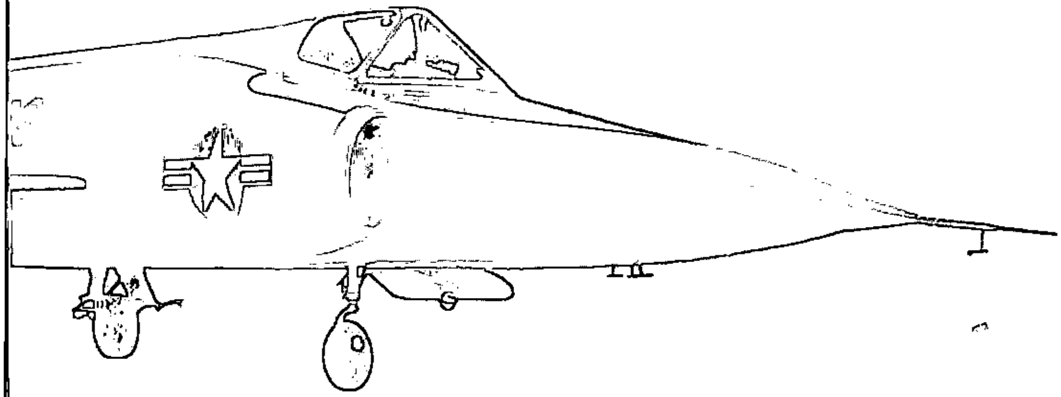
The airplane is equipped with yaw and pitch dampers, a turn coordinator, and a sideslip feedback. The sideslip feedback is provided only on the "small tail" airplanes until such time as they are retrofitted with a vertical tail of 40 percent larger area. (The first 62 production aircraft will initially have the small tail.) On the test aircraft provision was made to control the yaw and pitch dampers, turn coordinator, and sideslip feedback individually. On the standard aircraft one switch would control all these functions.

The dual head "yoke" stick, with radar control in the left grip, was installed during the tests.

The control system employs two separate irreversible hydraulic systems with a bungee system to provide feel. In addition to the bungee system there is a "q" (dynamic pressure) sensing device in the directional and longitudinal systems to vary control force with airspeed, and an automatic trim servo in the longitudinal system to compensate for unstable stick force in the transonic region. Control is provided by elevons located on the trailing edge of the wing and by a conventional rudder.

The airplane is characterized by a thin, 60 degree delta wing and half delta vertical tail, a sharp "V" shaped canopy, waisted fuselage, two air intake ducts on the sides of the fuselage, and a fully enclosed armament bay. The main gear retracts inboard into the wing and fuselage, and the nose gear retracts forward into the nose section of the fuselage. A clamshell type speed brake is installed above the tail cone. The speed brakes also act as doors for the drag chute compartment.

TEST RESULTS



☐ cockpit evaluation

Entrance to the cockpit is made by climbing a ladder attached to the left engine duct. It is difficult to climb the ladder when wearing full flight equipment, since it hangs past the vertical. The ladder should be modified so that the pilot moves forward toward the airplane as he climbs up. Emergency ground exit can be made by hanging from the engine duct and dropping to the ground.

The canopy hold-open rod consists of a tubular metal bar attached to the canopy, which rests on a receptacle inside the canopy sill. The rod is unsatisfactory for taxi operation, since winds above 30 knots will cause the canopy to fly up with possible damage to the airplane, and vibration from taxi operation may cause the hold-open rod to slip, allowing the canopy to close with possible injury to the pilot. Since taxi operation with the canopy closed in hot weather causes excessive pilot discomfort, a means should be provided to positively lock the canopy in any position.

The "T" on the canopy locking handle is oriented vertically. The pilot rubs his knuckles against the side of the cockpit when moving the handle, and the handle partially obstructs view of the first three or four warning lights on the warning light panel when it is in the forward position. Both of these discrepancies can be eliminated by orienting the "T" horizontally.

When the canopy is almost shut, high pressure air is relieved from the counter balance cylinder to permit the locking claws to be engaged easily. On several occasions the air was not released, and two or three attempts and up to 10 minutes were required to get the canopy closed. Aside from anoyance and pilot effort involved, scramble take-off procedures cannot tolerate the time lost in this simple operation. Canopy cinch down must be made more reliable.

The control handles for the drag chute, emergency gear extension, and canopy locking should be identified by placarding directly on the handles.

The rudder trim switch, located on the instrument panel pedestal behind the control stick, is difficult to locate and operate. Since the rudder trim is used frequently, the switch should be made more accessible by moving it to the left console as near the throttle quadrant as possible.

Labeling of the yaw and pitch damper switch is ambiguous. It is called a "FLIGHT MODE" switch and its two positions are labeled "DIRECT MANUAL" and "MANUAL." The switch should be labeled "STABILITY AUGMENTORS" and the two positions labeled "ON" and "OFF."

The anti-ice switch is three position: "AUTOMATIC" in the up position, "OFF" at center, and "MANUAL" in the down position. When the switch is moved from "AUTOMATIC" to "OFF," the pilot can easily overshoot to the "MANUAL" position and deplete his anti-ice fluid reservoir. The switch should be replaced by one similar to the cabin pressure switch in this aircraft, which would eliminate the possibility of inadvertently moving the switch to the "MANUAL" position.

Five rheostat switches are presently used to control cockpit lighting: 1. flight instrument group edge lighting and standby compass; 2. instrument panel (other than flight instruments) edge lighting; 3. console edge lighting; 4. instrument panel floodlighting; and 5. console floodlighting. The number of controls should be reduced to three to simplify the problems of control identification and operation. One control should be provided for all 115 volt AC edge lighting, one control for floodlighting, and the third for the instrument flight group lighting (28 volt DC).

The MA-4 lap belt cannot be pulled tight enough to hold a small pilot securely. A seat belt which can be adjusted to fit all pilots is considered mandatory from a safety standpoint.

The oxygen hose provided as part of the ship's system is not long enough for proper use with the MC-1 or MC-3 altitude suit. The length of the oxygen hose should be increased at least 4 but not more than 6 inches. It is believed that a hose longer than 36 inches (6 inches longer than that presently installed) may interfere with operation of cockpit controls.

The present fuel quantity indication allows a check of total fuel on board and fuel remaining in the No. 3 tank in each wing, but does not permit a check of the total fuel remaining in each wing. In the case of asymmetric fuel flow such as with a flow divider failure, it is mandatory for the pilot to be able to monitor the total fuel remaining in each wing. The system should be modified to allow the pilot to read both the total fuel and the quantity in each wing.

There is no immediate and positive means by which the pilot can determine if the fuel shut-off valves have operated properly in accordance with the position of the fuel shut-off valve selector control. Small indicator lights should be installed on the fuel selector panel to show open positions of the shut-off valves.

The windshield anti-reflection curtain sags, stretches, and it is easily torn, resulting in interference with the pilot's vision. It should be replaced with a durable fixed divider.

The high pressure pneumatic system incorporates a priority valve to provide a reserve of 1500 psi air pressure for emergency gear extension, ram air turbine displacement, drag chute operation, and wheel brakes. A cockpit warning light indicates that the entire system has depleted to 1500 psi, but does not show the priority system air pressure. If the air pressure in this system were low the pilot could make a landing unaware of an emergency that would leave him without one or both wheel brakes, and without a drag chute. A priority system pressure gage should be provided in the cockpit to give the pilot an indication of priority system pressure.

The landing gear warning light, located in the landing gear control handle, is difficult to see and should be replaced with a red light located on the instrument panel. In the present installation, the light comes on when the gear is not down and locked at altitudes below 10,000 feet and at airspeeds below 275-300 knots IAS. When flying adverse weather holding patterns, the light is on during the preparatory phases to landing. Besides being annoying to the pilot, the early actuation of the light reduces its value as a warning device. Sensing should be changed so that the light is actuated at 200 knots IAS and below.

The emergency landing gear extension control must usually be held in the aft (extend) position. As a consequence, the pilot's left hand cannot be used for operation of the power lever. If the emergency landing gear control slips forward it could result in partial retraction of the landing gear or possible rupture of the secondary hydraulic system, depending on the normal gear control handle position. Action should be taken to insure that all F-102 aircraft are provided with a positive catch to hold the emergency landing gear control in the aft position after operation by the pilot.

Unintentional jettisoning of the drag chute has occurred because of lack of a sufficiently positive detent or catch to hold the drag chute control handle in the "DEPLOY" position. The drag chute control handle mechanism should be modified to insure positive engagement and retention in the aft position.

Arrangement of the instrument panel is considered satisfactory and the layout appears optimized for the limited space available. There are several instrument deficiencies which should be corrected, however.

The turn and slip indicator gives erroneous indication when the aircraft is being rotated in a roll maneuver, or when rolling into or out of a turn. The indications are often opposite to the direction of the turn being initiated. The indications are correct only after a steady state turn has been established and is being maintained. The installation should be changed so that the instrument is oriented along the flight path axis of the aircraft rather than being placed so that its face is parallel to the instrument panel.

A plainly visible accelerometer should be installed in all F-102 aircraft.

The primary heading reference of the F-102A is a system composed of a J-2 gyro magnetic compass and the radio magnetic indicator. Indications are extremely poor because of error at the beginning, throughout, and upon completion of the turn. Errors as large as 20 degrees have been noted and it is always difficult to arrive at an exact heading.

The present altimeter has been misread by as much as 10,000 feet during high performance descents. This should be alleviated by installing an altimeter which provides a "barber pole" indication of aircraft altitude below 15,000 feet.

The present selsyn indicator must be read to determine afterburner nozzle position, since the position indications look alike. The indicator could be used more rapidly if the "OPEN" indication were black letters on a white background, and "CLOSE" were white letters on a black background.

The K4B-MM-2 remote attitude indicator has an "OFF" flag which indicates complete AC or DC power failure. Electrical failures in pitch or roll components will give erroneous readings on the instrument but no warning is provided. The attitude indicating system should be modified so that loss of either roll pitch signals will cause the "OFF" flag to appear.

☐ taxiing and ground handling

The taxiing and ground handling characteristics of the F-102A are generally satisfactory. Visibility to the front and sides is satisfactory, but the area aft of the wing tips cannot be seen due to the canopy design. Idle or near idle power is adequate for all taxi operations except initial movement from stand still. Nose wheel steering response is rapid and smooth, except that the high rudder force causes the aircraft to straighten out more quickly than desired during recovery from a short radius turn. The configuration of the nose wheel steering switch and the force required to keep the switch in the "ENGAGE" position result in hand fatigue. A flat-top switch similar to the ignition switch with minimum spring tension would be a better installation.

Braking action during taxi operation is unsatisfactory in that there is little relationship between brake pedal force and braking obtained. Large pedal forces are required to initiate braking action, and a slight increase in force will lock the wheel.

☐ take-off

Initial run-up and take-off are accompanied by a surge of ventilation air, which blows dust in the pilot's face. Since this impairs vision, it is a definte hazard to safe flight and should be corrected as soon as possible.

The controls are trimmed automatically prior to take-off by actuating a push button in the cockpit. This sets the rudder and aileron trim at zero degrees and the elevator to 5 degrees up. The aircraft is aligned with the runway using nose wheel steering. After the engine check at military power is completed the brakes are released with the nose wheel steering engaged. After the first 1 or 2 seconds of acceleration are completed and it is assured that directional control is being maintained, the afterburner is lighted. A distinct increase in acceleration will indicate a successful afterburner light.

It is recommended that nose wheel steering be engaged during the afterburner light. A severe directional change when the afterburner is lit indicates a malfunction of the afterburner eyelids and requires the afterburner to be shut off, power reduced to idle, and the take-off aborted.

Nose wheel steering should be discontinued when the rudder becomes effective at approximately 70 knots IAS. Differential braking should not be required but could be used cautiously if necessary. The aircraft is easily rotated to take-off attitude at 125 knots IAS using light stick forces with either a forward or aft center of gravity, and becomes airborne at 150 knots.

The elevons should be left near the trim position until 125 knots IAS is obtained, then sufficient force should be applied to rotate the aircraft to the desired attitude. No advantage is obtained by applying up elevon at low speeds. Figure 1, Appendix I shows a time history of a take-off wherein pull forces were maintained throughout most of the take-off roll. Figures 2 and 3 illustrate a take-off where back pressure was applied at the optimum nose wheel lift-off speed. In each instance, the aircraft responded at nearly the same speed. With an aft cg, full up elevon could cause rates of rotation at nose wheel lift-off that would be difficult to control.

The rapid acceleration after becoming airborne requires that a climb be maintained so that the landing gear may be fully retracted prior to reaching the gear limit speed of 220 knots IAS. Trim changes with gear retraction are minor.

Take-off using military power is similar, except for the longer take-off roll, shallower climb angle, and slower acceleration.

A time lag of as much as 2 seconds occurs between release of the pedals and actual release of the brakes. This sometimes results in non-simultaneous release of the brakes, impairing directional control in the take-off roll. The time lag between release of the brake pedals and the actual release of the brakes should be eliminated.

■ dynamic longitudinal stability

With the pitch damper off the aircraft does not meet MIL-F-8785(ASG) Specifications, which require that oscillations damp to 0.1 amplitude in one cycle. The damping is lowest at supersonic speeds, where oscillations require two to three cycles to damp to 0.1 amplitude and a small residual oscillation persists for a considerable length of time. (Figure 7, Appendix I.) Small oscillations which damp very slowly are easily induced in normal flying. Damping is much improved at subsonic Mach numbers but is still not considered satisfactory.

With the pitch damper operating longitudinal oscillations damp to 0.1 amplitude in one cycle and residual oscillations are eliminated. The pitch damper is necessary for high speed radar tracking.

A sharp movement of the longitudinal control causes a momentary change in normal acceleration opposite to the expected direction. This is caused by a change in lift coefficient due to elevon deflection before the aircraft responds to the pitching moment created.

Time histories of dynamic longitudinal oscillations presented in Figures 4 through 17 of Appendix I are summarized in the following table.

DYNAMIC LONGITUDINAL STABILITY

Fig.	Altitude	Mach No.	Conf.	Pitch Damper	Yaw Damper	No. Cycles to Damp to 1/10 Amp.	Residual Oscillation
4	45,000	1.07	Combat	on	on	1.5	no
5	45,000	0.9	Cruise	on	on	1.0	no
6	35,000	1.1	Combat	on	on	1.0	no
7	35,000	1.1	Combat	off	on	2.5	yes
8	35,000	1.1	Combat	on	off	1.0	no
9	35,000	1.1	Combat	off	off	2.5	yes
10	35,000	0.9	Cruise	on	on	1.0	no
11	35,000	0.9	Cruise	off	off	1.5	no
12	35,000	0.7	Cruise	on	on	1.0	no
13	35,000	0.7	Cruise	off	off	2.0	no
14	10,000	0.7	Cruise	on	on	0.5	no
15	10,000	0.5	Cruise	on	on	0.5	no
16	10,000	0.33	Pwr Approach	on	on	0.5	no
17	10,000	0.33	Pwr Approach	off	on	1.0	no

☐ static longitudinal stability

The static longitudinal characteristics of the F-102A are similar to those of other transonic and supersonic aircraft. Stability is positive at all altitudes in the low subsonic speed region, but is nearly neutral between 0.75 and 0.9 Mach number, as shown in Figure 24. The trim change associated with transonic flight begins at approximately 0.75 Mach number at low altitudes, and at 0.85 to 0.9 Mach number at high altitudes. A pull force of approximately 8 pounds is required to maintain level flight while accelerating from 0.9 to 1.1 Mach number at 35,000 feet with a center of gravity of 26.9 percent MAC. With an aft center of gravity (30.3 percent MAC), the pull force is decreased to 5 pounds, as shown in Figure 22, Appendix I.

The aircraft is equipped with an automatic trim servo to compensate for the trim change in the transonic and supersonic region. The trim servo automatically retrims the elevons up from 0.85 to 1.1 Mach number, which completely eliminates the stick force reversal in this speed region. As shown in Figures 19 and 21, a push force of approximately 12 pounds is required to accelerate from 0.9 to 1.1 Mach number with the trim servo operating.

Although trim servo operation is satisfactory, one minor deficiency was noted. When the trim servo system is engaged supersonically and the aircraft is slowed to minimum flying speed without manually retrimming, the pull forces required decrease at speeds below 0.85 Mach number, (Figure 21, Appendix I). This is not an unsatisfactory condition since the minimum pull forces are in excess of 10 pounds, and in ordinary operation the aircraft would be manually retrimmed. It should also be noted that the operating limits of the trim servo had not been finalized at the time of the Phase IV tests.

The low speed stick-fixed neutral points were found at lift coefficients of 0.20, 0.30, and 0.40. The neutral point was found to vary from 32.8 percent MAC at a lift coefficient of 0.40 to 37.6 percent MAC at a lift coefficient of 0.20. The neutral point at high lift coefficients or low speed appears from extrapolated data to be approximately 2.5 percent MAC aft of the present flight limit of 30.5 percent MAC.

☐ trim changes

The longitudinal trim changes resulting from speed brake operation, afterburner operation, and landing gear extension are small, and the forces required are generally within the control friction band. Lighting the afterburner causes a dynamic oscillation, which damps out quickly without pilot effort. (Figure 26, Appendix I.) No trim change is evident after the oscillation. This oscillation could be caused by non-simultaneous opening of the afterburner eyelids.

Trim changes were determined by trimming the aircraft for level flight, introducing the variable, and maintaining level flight. The change in forces and positions are summarized in the following table.

LONGITUDINAL TRIM CHANGES

Calibrated Airspeed Knots	Alt-Ft.	Landing Gear	Speed Brake	Variable	Elevon Defl.	Long. Stick Force Change
325	35,000	up	In	Sp Brk out	0.7 dn	Negligible
225	20,000	up	In	Gear dn	1.3 dn	5 Lbs. Forward
240	35,000	up	In	A/B on	None	Negligible

☐ maneuvering flight

The maneuvering force gradients are unsatisfactory at the extreme limits of the operational flight envelope.

The force gradient is excessive above 0.9 Mach number at 50,000 feet. As shown in Figure 27, the gradients of 30 pounds per g at 26.9 percent MAC and 25 pounds per g at 30.1 percent MAC obtained at 0.97 Mach number greatly exceed the 9 pounds per g maximum permitted by Air Force Specifications. The gradient at supersonic speeds at 35,000 feet is also excessive as may be seen in Figure 29.

The force gradient of only 2.5 pounds per g obtained at 0.72 Mach number and 35,000 feet, as shown in Figure 32, is less than the 3.5 pounds per g minimum allowed by the specification. Maneuvering gradients are satisfatcory throughout the remainder of the subsonic flight regime, with the exception of the power approach configuration with a forward cg where the force gradient is excessive.

Force gradient characteristics during sudden pull-ups are inconsistent and undesirable under certain conditions. If the stick is displaced and returned to neutral in from 1 to 2 seconds, the stick force per g is sometimes less than that required for steady maneuvering flight, which is contrary to MIL-F-8785 (ASG) requirements. However, if the stick is displaced and returned to neutral in 1 second or less, stick forces exceed steady maneuvering flight gradients. This characteristic is not objectionable. Pull-up and return rates of 3 seconds or over usually result in gradients similar to those obtained in steady accelerations.

A stick-free pitch-up is encountered when making decelerating turns from supersonic trim speeds. Decreasing back forces are required above 2.5g and the aircraft will maintain 3g with the stick free. This condition is encountered with the trim servo on or off, as may be seen in Figures 37 and 38, Appendix I, and results from the combined effect of decelerating through the trim change and the bob weight effect of the yoke stick.

▣ dynamic lateral-directional stability

The airplane meets Air Force requirements in dynamic lateral-directional damping, since oscillations damp to 0.10 amplitude or less in 1 cycle with the yaw damper operating. No significant difference in the damping is noted with addition of either the turn coordinator or sideslip feedback. A barely perceptible high frequency oscillation accompanied by light buffet persists in the speed region from .96 to 1.0 true Mach number with the yaw damper operating. This is annoying but is not considered unsatisfactory since it occurs in a seldom-used region of flight.

Little damping occurs with the yaw damper off. As shown in Figures 46 and 50, the aircraft oscillates at a frequency of approximately ½ cycle per second, requiring 1½ cycles to damp to ½ amplitude.

The production yoke stick produces an undesirable effect on the dynamic lateral-directional characteristics. A sharp lateral disturbance of the control stick causes a chatter in the control system, and an elevon oscillation of approximately 3 cycles per second. This persists for 3 to 4 cycles, and is followed by a slower oscillation in the natural frequency of the aircraft. This is caused by the bob weight effect of the stick coupled with the dynamics of the control system and the airplane. At 1.1 Mach number the mass of the stick is sufficiently affected by the lateral loads to produce a continuous lateral-directional oscillation with the stick free. This effect may be seen in Figure 43, Appendix I.

At Mach numbers above 0.9 the aircraft exhibits a small amplitude directional oscillation in response to a longitudinal pulse. As shown on the dynamic longitudinal time histories, Figures 4 through 17, Appendix I, these oscillations are eliminated by the yaw damper.

☐ static directional stability

Although the aircraft exhibits positive static directional stability in all configurations and all altitudes, directional control does not meet the specification requirements since the rudder force gradient becomes neutral and even negative at large deflections such as that required for 10 degrees of sideslip in the power approach configuration. This is illustrated in Figure 65, Appendix I. Control is satisfactory to the pilot; however, since positive gradients are encountered in all realistic realms of flight.

The rudder force required for a given rudder deflection rises rapidly with Mach number due to a Mach sensing device in the rudder feel system.

The aircraft meets the Air Force Specification under all conditions with the sideslip feedback operating. Nearly twice the force is required for a given rudder deflection with the feedback operating as would be required without it. (Figure 54.)

□ lateral control

The lateral control system is unsatisfactory in that wings-level flight is difficult to maintain. With the aircraft trimmed for level flight a single maneuver requiring lateral control will cause the aircraft to be out of trim upon completion of the maneuver. This problem is caused by a combination of high sensitivity of the control near trim and high breakout forces from trim. The breakout friction and sensitivity at trim should be reduced, and in addition, positive centering must be provided for the yoke stick.

The roll rate with maximum aileron deflection meets Air Force Specifications in the cruise configuration. Roll rates above 200 degrees per second are attainable at 10,000 feet with a helix angle of 0.086.

The full lateral control capabilities of the aircraft cannot easily be realized, however, because the control stick hits the pilot's knees at three-fourths deflection (10.5 to 12 degrees). This reduces the roll capability at 10,000 feet to only 160 degrees per second with a helix angle of 0.066. At 45,000 feet the helix angle is further reduced to .06. The aircraft is still capable of meeting the requirement calling for 100 degrees of bank change in 1 second after control initiation at altitudes below 20,000 feet.

At speeds below 300 knots, adverse yaw tends to slow the roll, and at speeds above 300 knots complimentary yaw accelerates the maneuver. The sideslip feedback and turn coordinator decrease the yaw in all cases. The yaw damper, with the other stability augmentation devices off, initially tends to create adverse yaw in all rolls. The roll capabilities shown in Figures 66 through 69 are those obtainable within 120 degrees of bank change with turn coordinator and sideslip feedback operating. Rolls through 360 degrees with half aileron showed no increase in rate beyond 120 degrees of bank change.

The lateral control does not meet Air Force Specifications in the power approach configuration. The helix angle of .03 obtainable is 40 percent below the required .05, although the peak roll rates of 35 degrees per second commenserate with the .03 helix angle obtainable at 180 knots IAS are adequate under adverse landing conditions. The principle reason for the low helix angle is a 0.3 second delay between control initiation and response of the aircraft. Only about 12 degrees of aileron travel, rather than the full 14 degree capability of the control system, is available because of insufficient room for full stick throw.

The forces required to roll the aircraft are acceptable in all configurations. The bungee system used in the lateral control system is unaffected by inflight air loads and produces a force gradient of approximately 1 pound per degree of total differential elevon.

□ inertial coupling

Limited tests were performed to determine the inertial coupling tendencies within the limits specified by the manufacturer. The aircraft was rolled through 360 degrees at 80 degrees per second with the sideslip feedback off and 120 degrees per second with the feedback on.

At airspeeds above 305 knots CAS and at all supersonic speeds the aircraft exhibits complimentary yaw during elevator fixed rolls. The complimentary yaw is increased by down elevator. An up elevator input causes decreased complimentary yaw, or in some cases, adverse yaw. At speeds below 305 knots, the aircraft exhibits adverse yaw when rolled without elevator input. With a positive elevator input, the adverse yaw is increased; with a negative elevator input it is decreased. In no case were the yaw angles encountered dangerous. The roll characteristics at all speeds with the sideslip feedback operating are similar to those above 305 knots with the feedback off.

Further discussion is included in AFFTC-TN-55-24, "Roll Evaluation of the F-102A" and AFFTC-TN-56-15, "Roll Evaluation of the F-102A with an Enlarged Vertical Stabilizer."

□ trim rates and trim control

The longitudinal and directional trim rates and response are satisfactory but the effectiveness of the lateral trim control is considered unsatisfactory because of the excessive delay between initiation of lateral trimming action and the accomplishment of the desired amount of trim. This problem is amplified by the high sensitivity to lateral control movement. The steady state lateral trim rate in flight is approximately 40 percent higher than either the directional or longitudinal trim rate.

The trim rates were determined on the ground and inflight by trimming the aircraft through the maximum range available. In every case, the values shown are average values over the entire range. Both the longitudinal and directional controls have feel force systems that are functions of airspeed. Since they affect the forces being trimmed out, they also affect the trim rate on these controls. The lateral control system feel force is obtained from a bungee system only and is not affected by airspeed.

Control	Rate of Trim-deg/sec		In-flight Airspeeds Knots, IAS
	Ground	In-flight	
Directional	**1.55**	**0.72**	**200**
Lateral	**1.45**	**1.05**	**300**
Longitudinal	**1.32**	**0.85 & 0.75**	**200 & 265**

stalls and low speed handling characteristics

The low speed handling characteristics are good. The aircraft exhibits buffet at speeds well above the point where dangerous sink rates develop. As the angle of attack is increased, directional control is gradually lost as the rudder is blanked out by the wing and fuselage. Continuous use of rudder and aileron is required to maintain straight and level flight below 120-125 knots IAS. Longitudinal control remains good at speeds near stall and recovery is easily effected by reducing the back pressure on the stick. The aircraft was not flown at speeds below 110 knots since spin tests had not been demonstrated.

approach and landing

The recommended initial approach speed is 300 to 350 knots IAS. The speed brake is extended and power reduced during the break turn to arrive on the downwind leg at 220 knots. Landing gear is lowered prior to turning on the base leg. After the gear is lowered the speed brake may be closed to reduce buffeting. A speed of 200 knots is maintained on the base leg, and the turn to final is started at 190 knots and completed at 180 knots. The optimum approach speed is 160 knots with power being used to maintain airspeed, the over-the-fence speed is 150 knots, and the touchdown is made at 135 knots.

The drag chute is deployed immediately after touchdown while the nose wheel is still off the ground. The drag chute is in line with the cg so that no pitching moment is created by deployment. Directional control remains good upon deployment, and yaw due to crosswind is slight. The nose wheel is lowered at 100 to 110 knots IAS by use of the longitudinal control.

Braking action during landing is unsatisfactory. Caution must be used when braking due to the heavy pedal forces and lack of feel in the brake system. Heavy braking will cause skidding, and with the excessive lag in brake release a blown tire will usually result before the skid can be stopped. Positive and effective braking action requires that anti-skid devices be incorporated in the brake system.

Drag chute deployment is not reliable. Fifteen failures occurred in 100 deployments accomplished during Phase IV and Phase VI testing. The drag chute deployment time is satisfactory, but an additional second is required for deployment if the landing is made with the speed brakes closed.

engine operation

Continuous engine stalls occur with a combination of low indicated airspeed and low power setting at high altitude. During one flight at 45,000 feet altitude the throttle was slowly retarded to idle, causing steady compressor stalls. Throttle movement had no effect upon recovery and the aircraft was dived at 0.9 Mach number to prevent the exhaust temperature from exceeding the maximum limits. Dive recovery was made just below 30,000 feet and, as airspeed decreased, slow throttle advancement permitted engine acceleration through the stall region. This incident is similar to several encountered during the Phase VI F-102A program. One of the Phase VI aircraft was provided with a modified bleed valve schedule that eliminated this difficulty. All aircraft should be equipped with an optimum bleed valve schedule as soon as possible.

On several high final approach landings the throttle was kept in idle. When attempting to add power for the last part of the final approach several compressor stalls were experienced which delayed engine acceleration approximately 5 seconds. This was accompanied by a loss in airspeed of 5 to 10 knots. The need for stall-free engine operation at low altitude, low airspeed and low power setting is mandatory from a safety standpoint.

The test aircraft did not have the extended ramp on the inlet duct, and very annoying duct rumble and buzz were experienced above 1.2 Mach number.

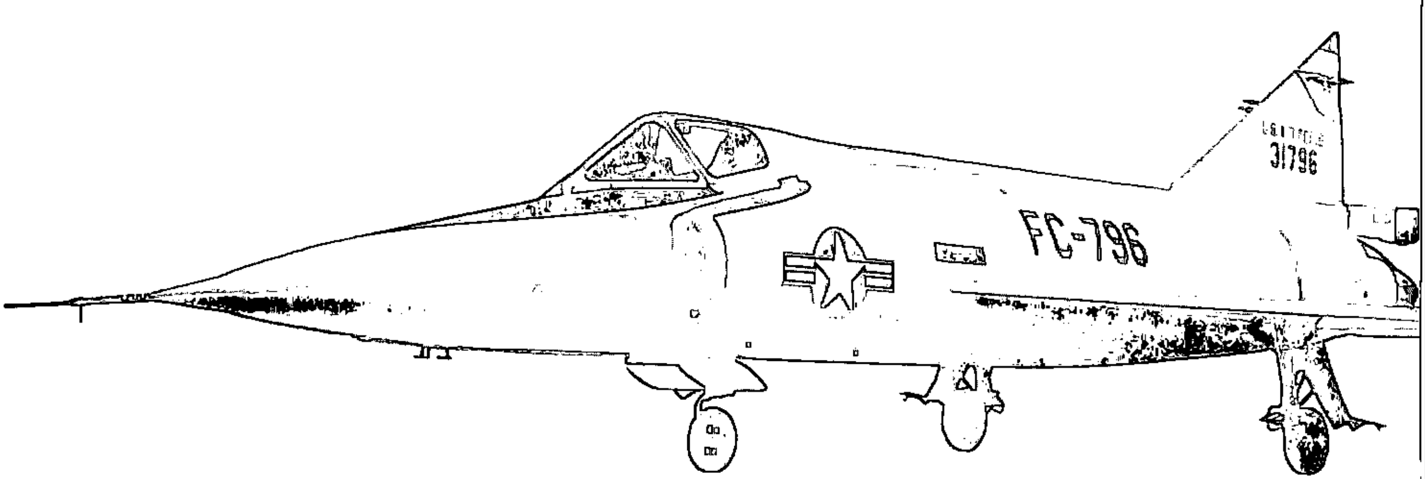
control friction

The aileron and rudder break-out forces are excessive and exceed the requirements of Air Force Specification MIL-F-8785 (ASG). The static break-out friction for the lateral control system at the trim position exceeds the specification by 1 pound, or 50 percent. Both lateral and directional break-out friction are much less at other than trim positions.

The static break-out forces as measured after flight No. 7 are as follows:

Control	Break-out Force-lbs.	Maximum Alowable Break-out Force MIL-F-8785 (ASG)
Longitudinal	3	3
Lateral	3	2
Directional	11	7

The static friction calibrations were obtained by moving the controls in slow, short movements so that the break-out force was measured with each movement (see Figures 75 through 77). No in flight control friction data was obtained.



CONCLUSIONS

The stability and control characteristics of the F-102A with stability augmentors operating are considered satisfactory for an all-weather interceptor. Without the stability augmentors operating the aircraft could be flown safely, but would probably be unable to successfully accomplish an intercept mission.

Several deficiencies which are annoying but do not seriously decrease the combat effectiveness were encountered.

1. The yoke stick prevents effective use of full aileron.
2. The bob weight effect of the yoke stick produces undesirable dynamic effects.
3. Lateral and directional control break-out friction is excessive and lateral control sensitivity near trim is too high.
4. Longitudinal control forces are excessive above 50,000 feet at Mach numbers greater than 0.9 and in the power approach configuration with a forward cg, but forces are too light below 35,000 feet at Mach numbers below 0.7 in the cruise configuration.
5. The maneuvering force gradients are unsatisfactory at the extreme limits of the operational flight envelope.
6. Forces required to obtain a given value of normal acceleration during abrupt pull-ups vary with the speed of the control input and return to neutral.

RECOMMENDATIONS

It is recommended that the following items be corrected as soon as possible:

1. The throw of the lateral control system be modified to allow the pilot to obtain full aileron.
2. The rudder trim switch be relocated to the left console for easier operation.
3. The belt length or its installation be changed to insure sufficient adjustment range for all pilots.
4. A priority system pneumatic pressure gage be installed in the cockpit.
5. The lateral trim control be modified to give less lag between control initiation and aircraft response.
6. The lateral break-out friction and control sensitivity near trim be reduced.
7. A positive detent or catch be provided to hold the emergency gear extension handle in place when activated.
8. Canopy cinch down operation be made more reliable.
9. The fuel quantity indication be modified to allow the pilot to read both the total fuel and the quantity in each wing.
10. The time delay between release of the brake pedal and actual release of the brakes be eliminated.
11. The present altimeter be replaced with an altimeter which provides a "barber pole" indication of aircraft altitude below 15,000 feet.
12. Steps be taken to eliminate engine compressor stalls.
13. The extended ramp on the inlet duct be installed on all aircraft.
14. An anti-skid device be installed in the brake system.
15. The attitude indicating system be modified so that loss of either roll or pitch signals will cause the "OFF" flag to appear.

The following items are less critical but should be accomplished as soon as feasible.

1. A bob weight be added to the longitudinal control system to offset the weight of the yoke stick.
2. The oxygen hose length be increased 6 inches.

3. The drag chute control mechanism should be modified to insure positive engagement and retention in the aft "DEPLOY" position.

4. A plainly visible accelerometer should be installed in all F-102 aircraft.

5. The surge of ventilation air on initial engine run-up and take-off should be eliminated.

6. The angle at which the ladder for cockpit entrance hangs be modified so that the pilot moves forward toward the airplane.

7. A means be provided to positively lock the canopy in any position during taxi operations.

8. The "T" on the canopy locking handle be oriented horizontally.

9. The control handles for the drag chute, emergency gear extension and canopy locking be identified by placarding directly on the handles.

10. The flight mode switch be called "STABILITY AUGMENTATORS" and the labeling be changed from "DIRECT MANUAL" and "MANUAL" to "ON" and "OFF."

11. The anti-ice switch be replaced by one similar to the cabin pressure switch in this aircraft.

12. The number of rheostat switches used to control cockpit lighting be reduced from five to three.

13. Small indicator lights should be installed on the fuel selector panel to show open positions of the fuel shut-off valves.

14. The windshield anti-reflection curtain should be replaced with a durable fixed divider.

15. The landing gear warning light should be relocated with a red light located on the instrument panel.

16. Sensing should be changed on the landing gear warning light so that it operates at altitudes below 10,000 feet and airspeeds below 200 knots IAS.

17. The present primary heading reference should be replaced with a system which provides accurate heading indications during turns.

18. The turn and slip indicator should be aligned so the instrument longitudinal axis is parallel to the aircraft flight path longitudinal axis.

19. The afterburner selsyn indicator be changed so the "OPEN" indication will employ black letters on a white background and the "CLOSE" position white letters on a black background.

20. The present nose wheel steering switch be replaced with a flat-top switch similar to the ignition switch.

21. The rudder break-out friction be reduced.

APPENDIX I

The elevon data presented in these plots may be in error by as much as plus or minus one-half degree when compared with data from a different flight. These errors result from zero shifts, which were impossible to detect due to the differential action of the elevons. The data on any single maneuver should have an accuracy of approximately plus or minus one-tenth degree.

Elevator angles are taken as the average of the two elevon positions. The terms elevator and average elevon are used interchangeably throughout the report.

Aileron angles are taken as the difference between the two elevon positions. The terms aileron and differential elevon are used interchangeably throughout the report.

stability curves

Figure

1 TAKE-OFF TIME HISTORY 16
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75 LONGITUDINAL BREAK-OUT FRICTION 95
76 LATERAL BREAK-OUT FRICTION 96
77 RUDDER BREAK-OUT FRICTION 97

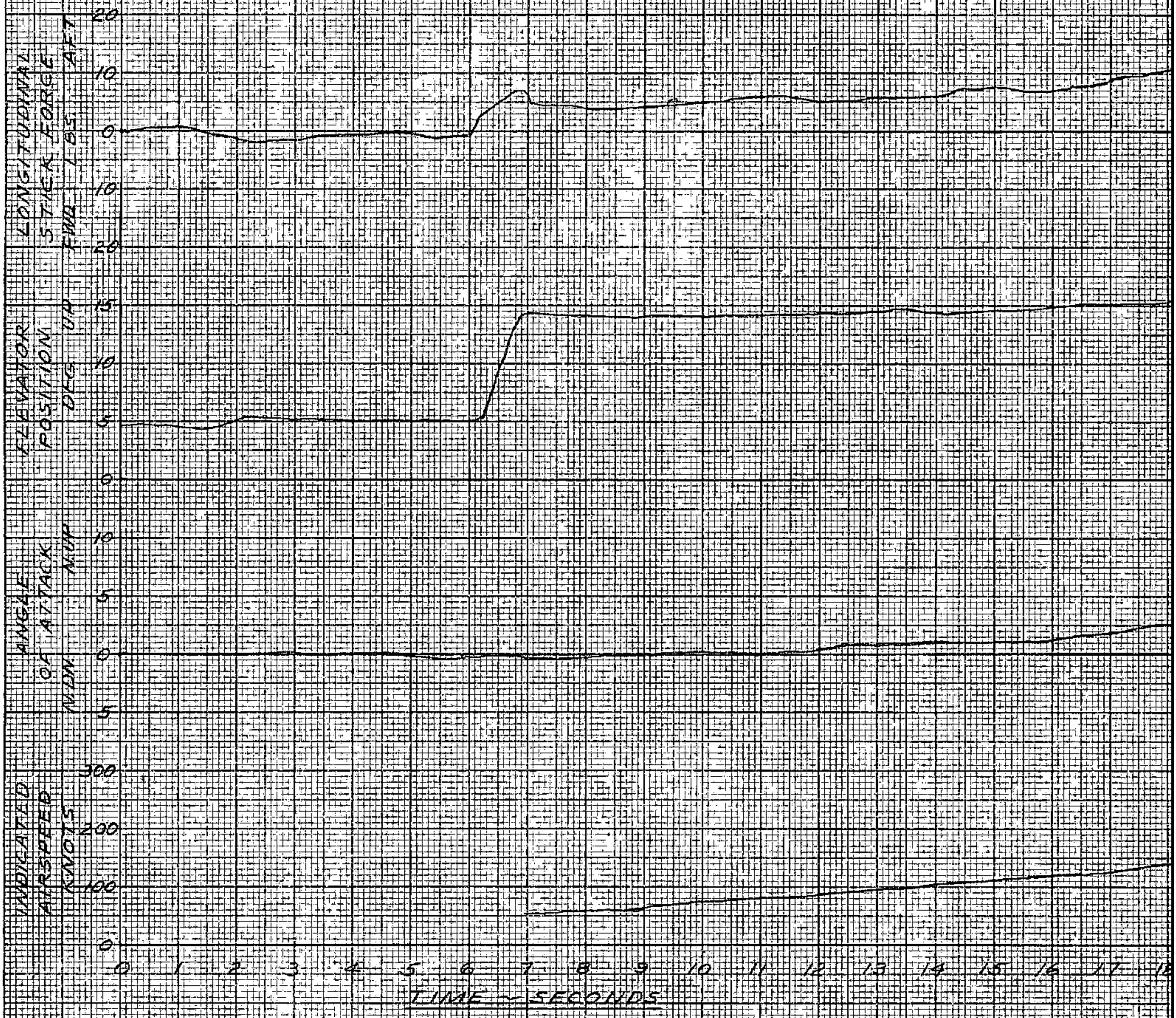
FIGURE NO. 1

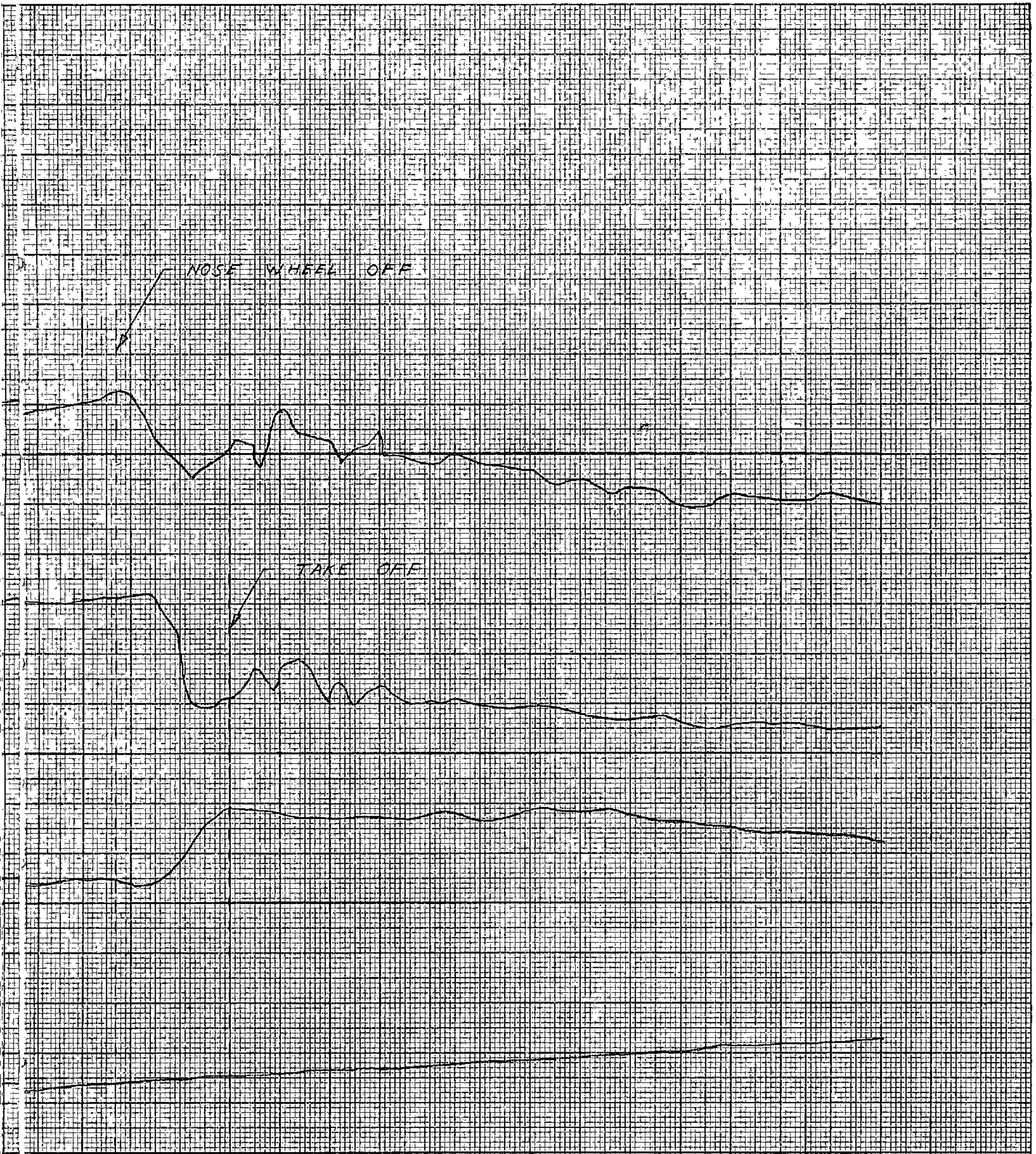
TAKE-OFF TIME HISTORY

F-102A USAF NO. 53-1796

WEIGHT 27000 LBS.

C.G. 28.5 PERCENT MAC




NOSE WHEEL OFF
TAKE OFF
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34
TIME ~ SECONDS

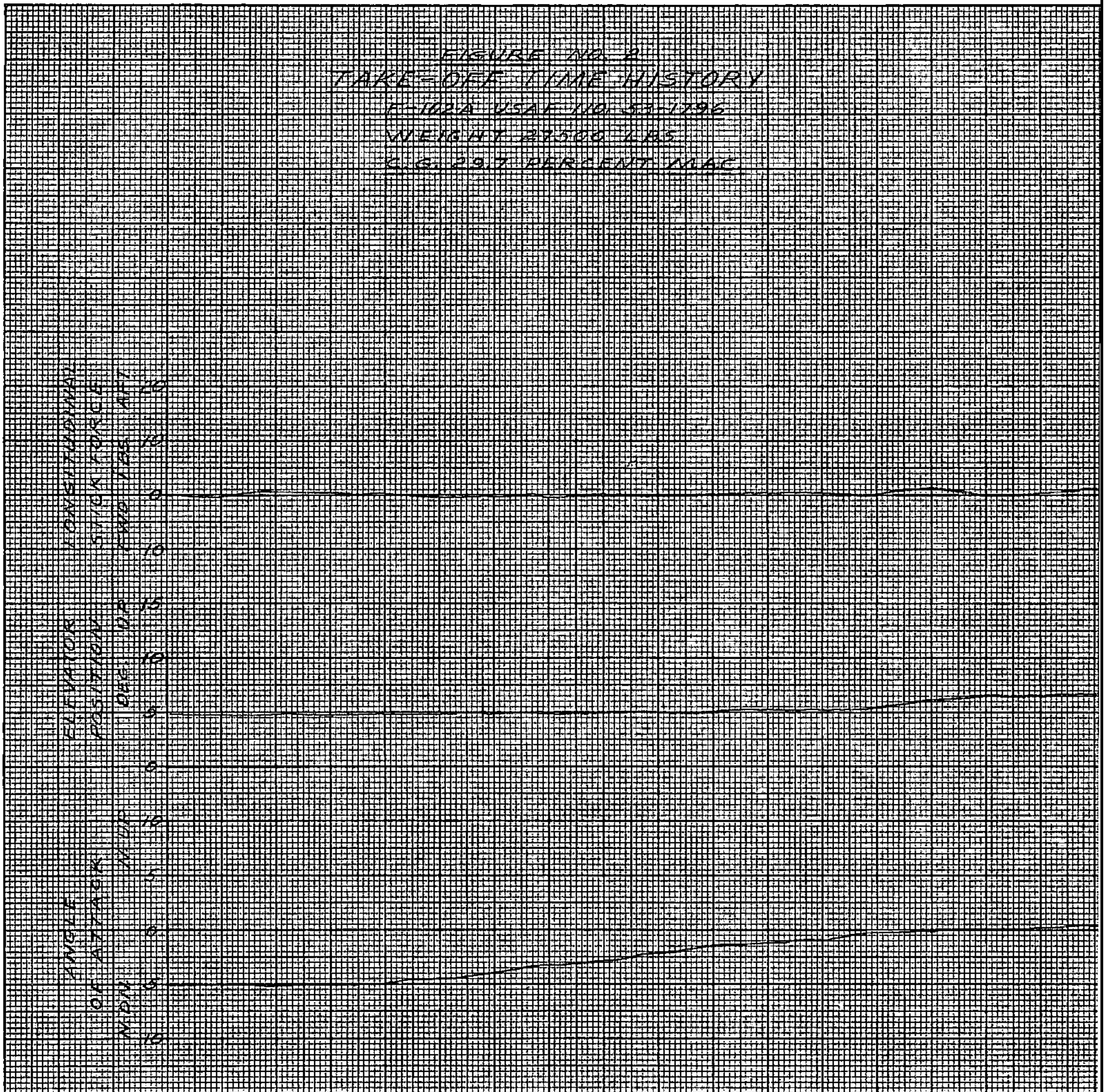
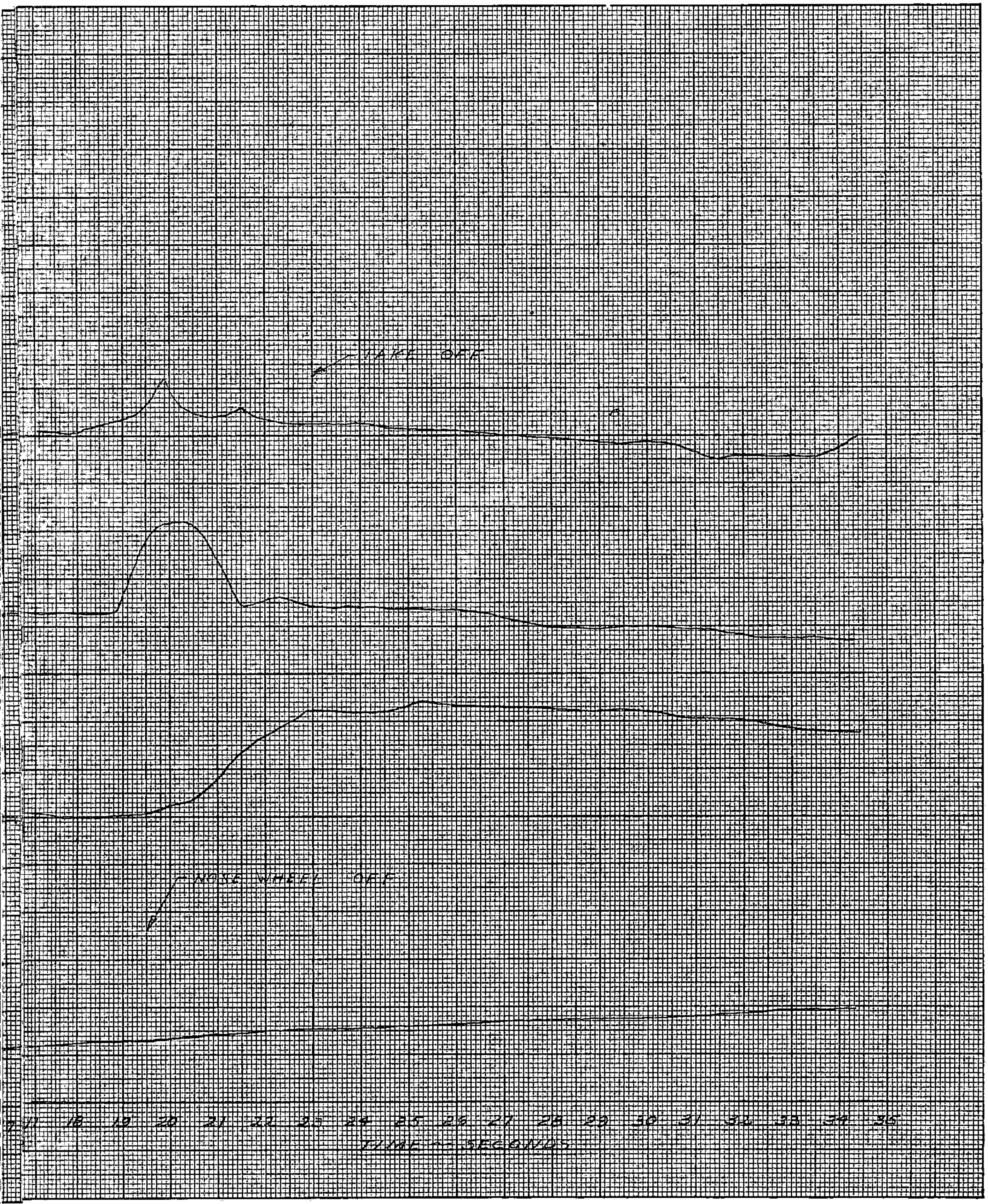


FIGURE NO. 2
TAKE-OFF TIME HISTORY
F-102A USAF NO. 53-1296
WEIGHT 27500 LBS.
C.G. 29.7 PERCENT MAC.


TAKE OFF
NOSE WHEEL OFF
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18
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30
31
32
33
34
35
TIME — SECONDS

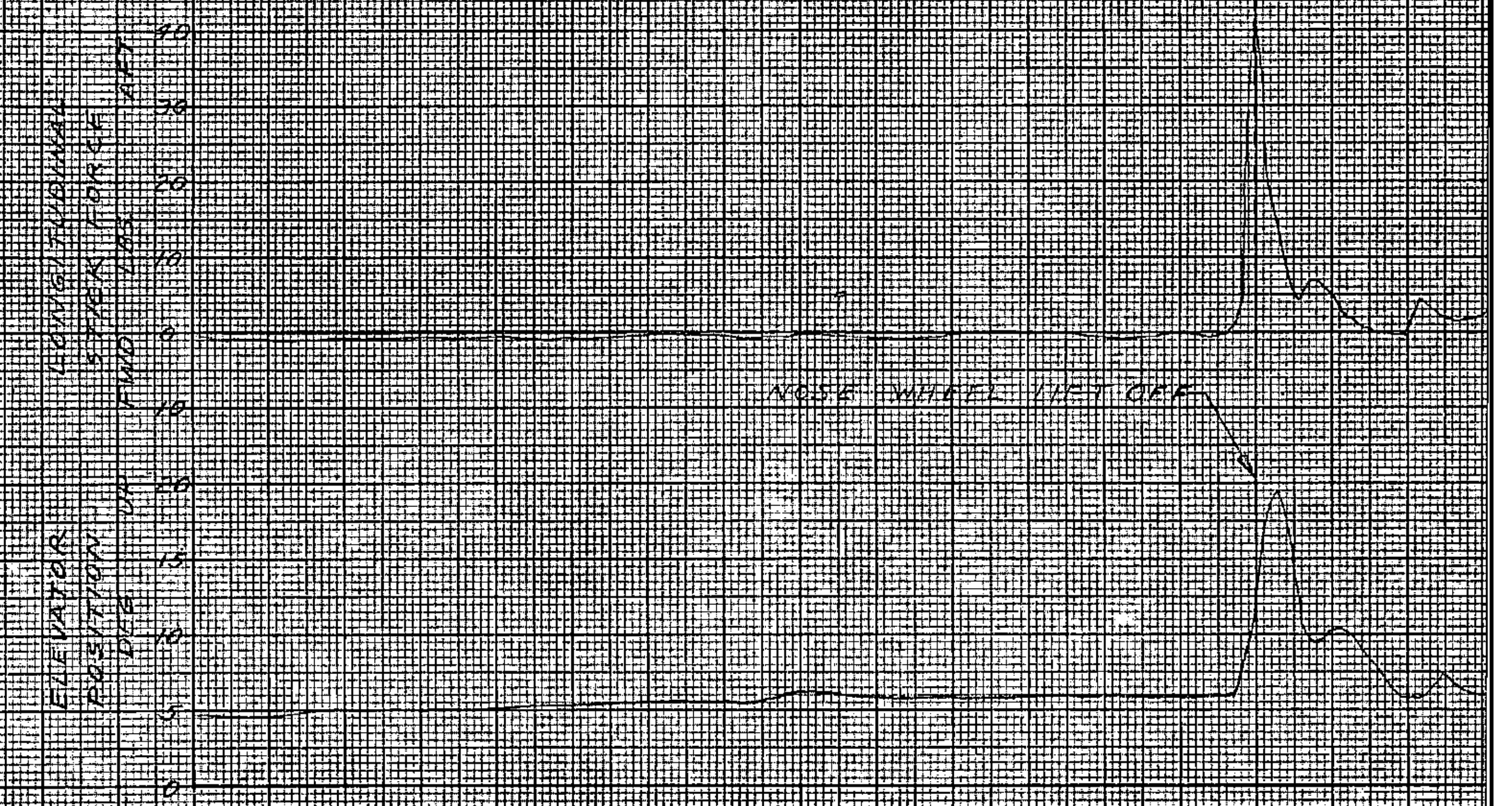
FIGURE NO. 5

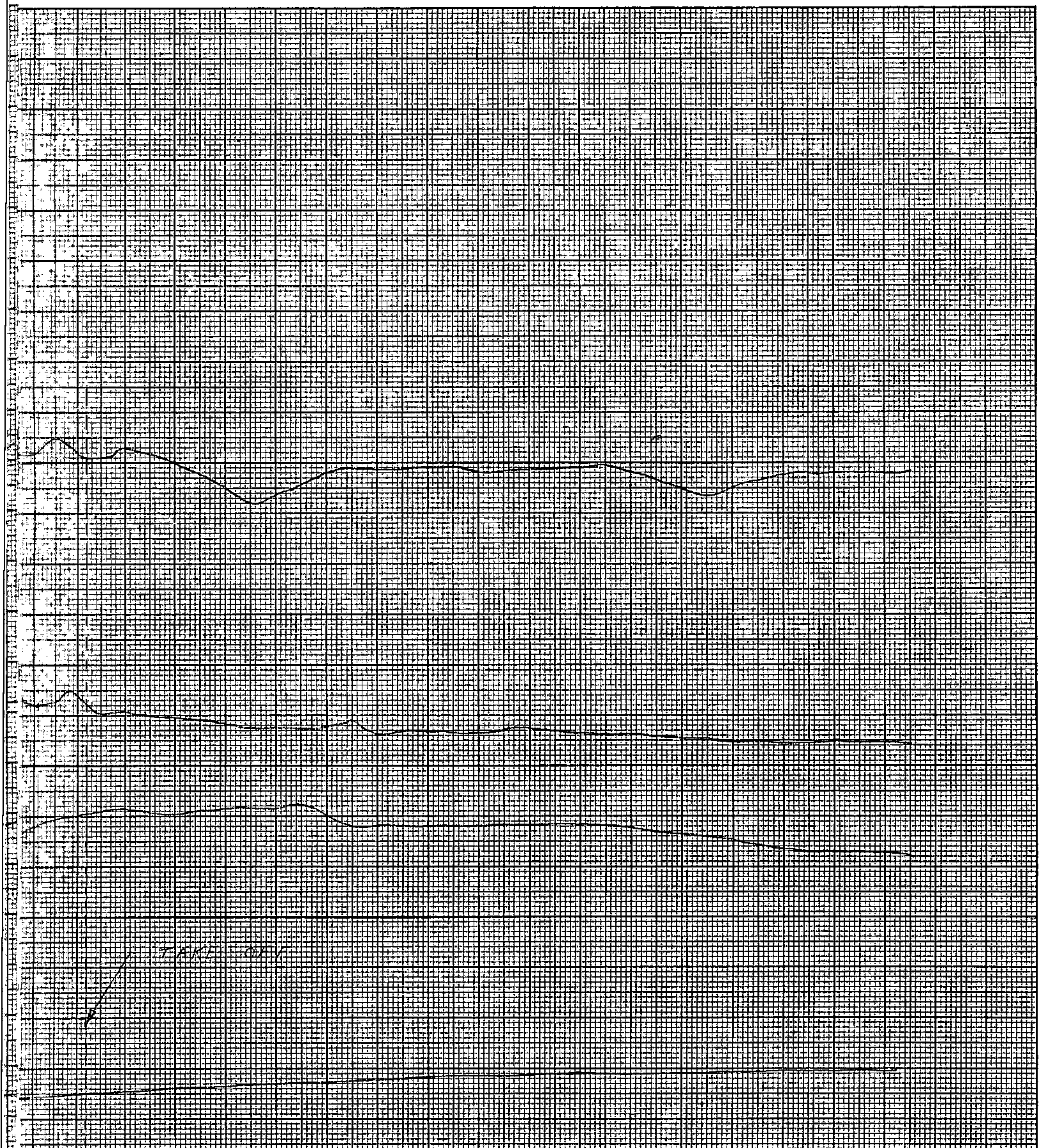
TAKE-OFF TIME HISTORY

F-102A USAF NO. 53-1796

WEIGHT 26200 LBS.

C.G. 30.5 PERCENT MAC.




TAKE OFF
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41
42
TIME - SECONDS

Figure No. 4

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
45,000 Altitude
24200 Lbs. Weight

Pitch Damper On
Mach No. 1.07
Yaw Damper On

C. G. 27.0 percent MAC

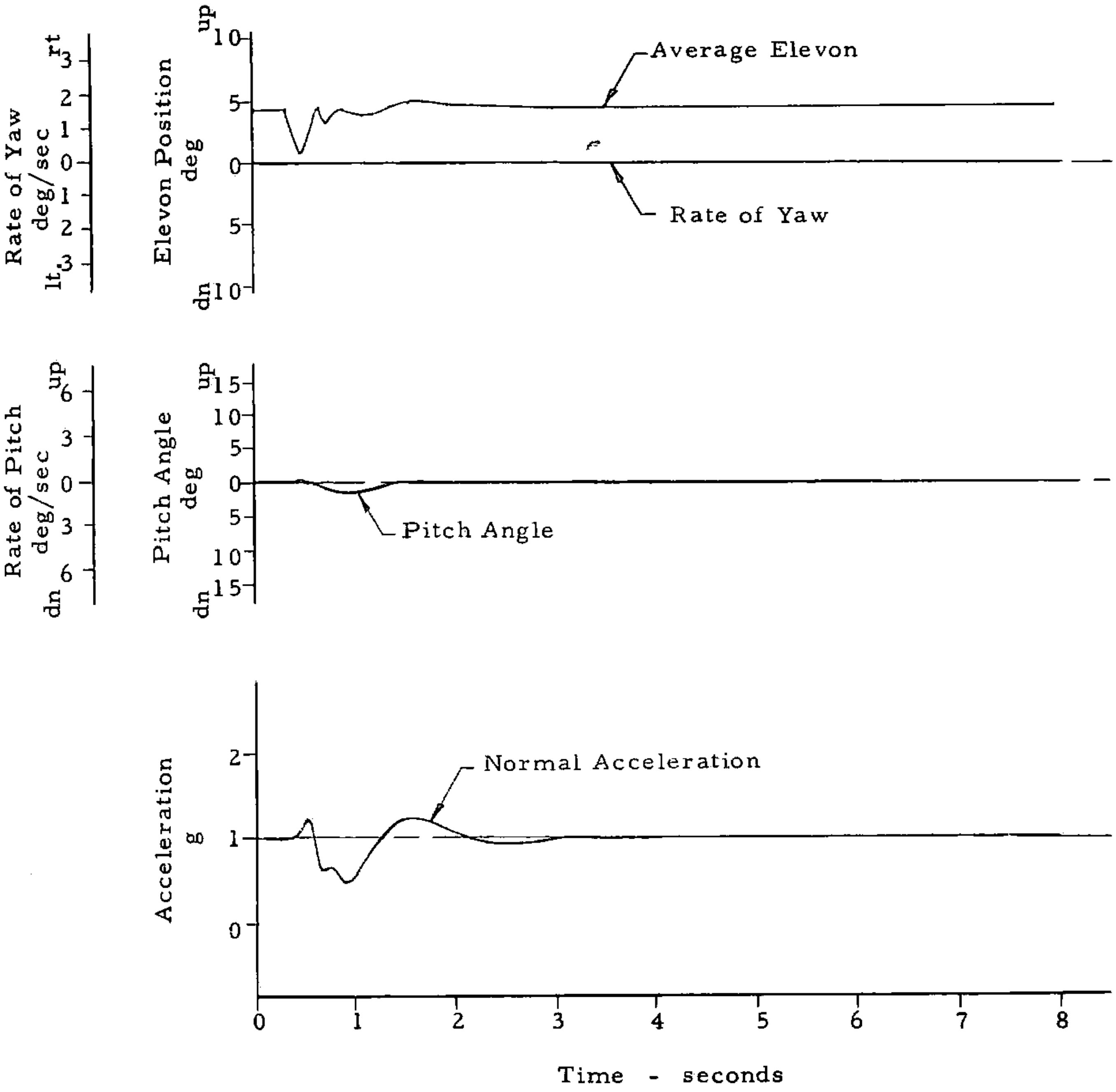


Figure No. 5

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration Pitch Damper On

45,000 Altitude Mach No. 0.9

23600 Lbs. Weight Yaw Damper On

C. G. 27.1 percent MAC

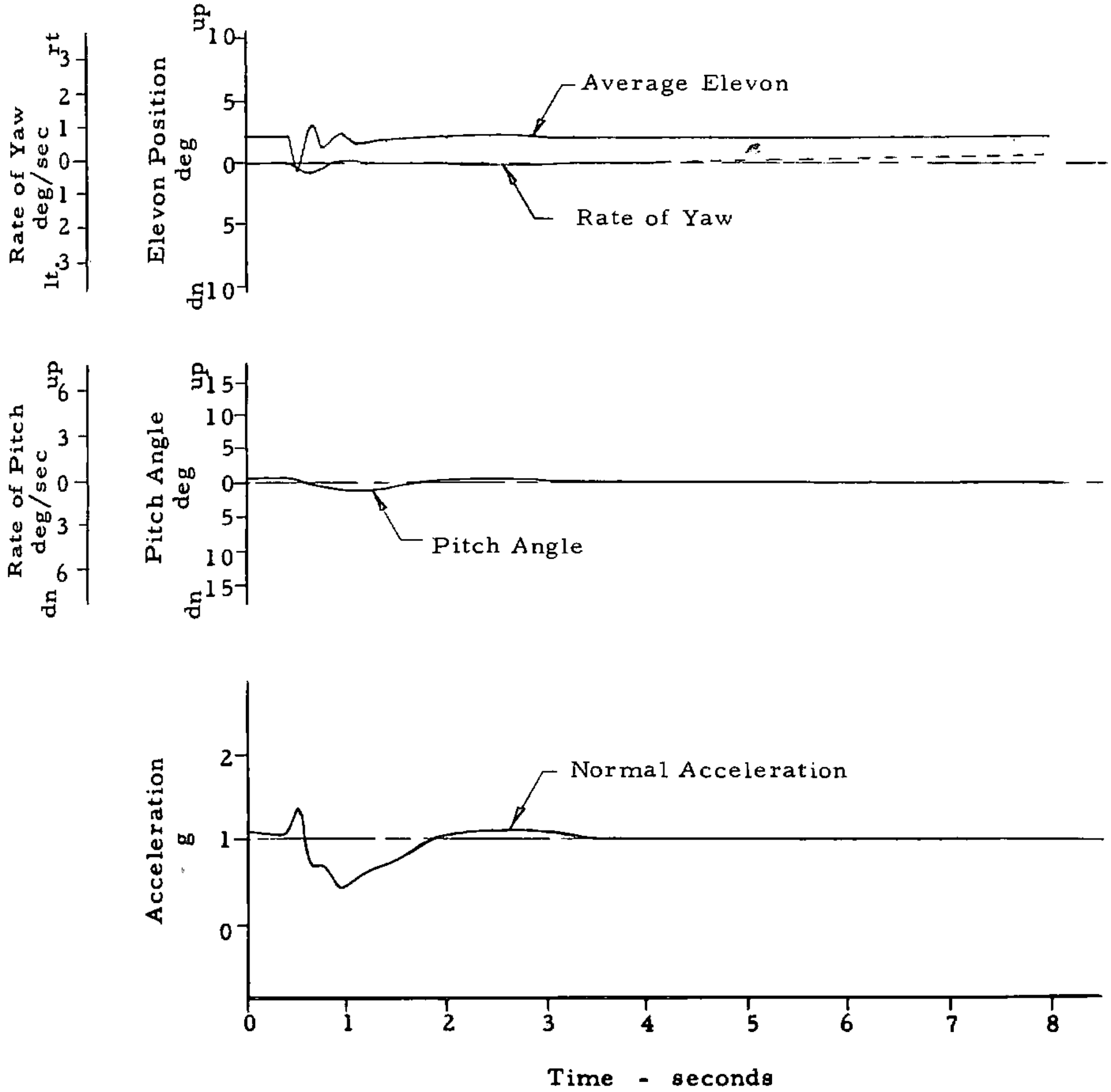


Figure No. 6

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration

35, 000 Altitude

23400 Lbs. Weight

Pitch Damper On

Mach No. 1.1

Yaw Damper On

C.G. 27.1 percent MAC

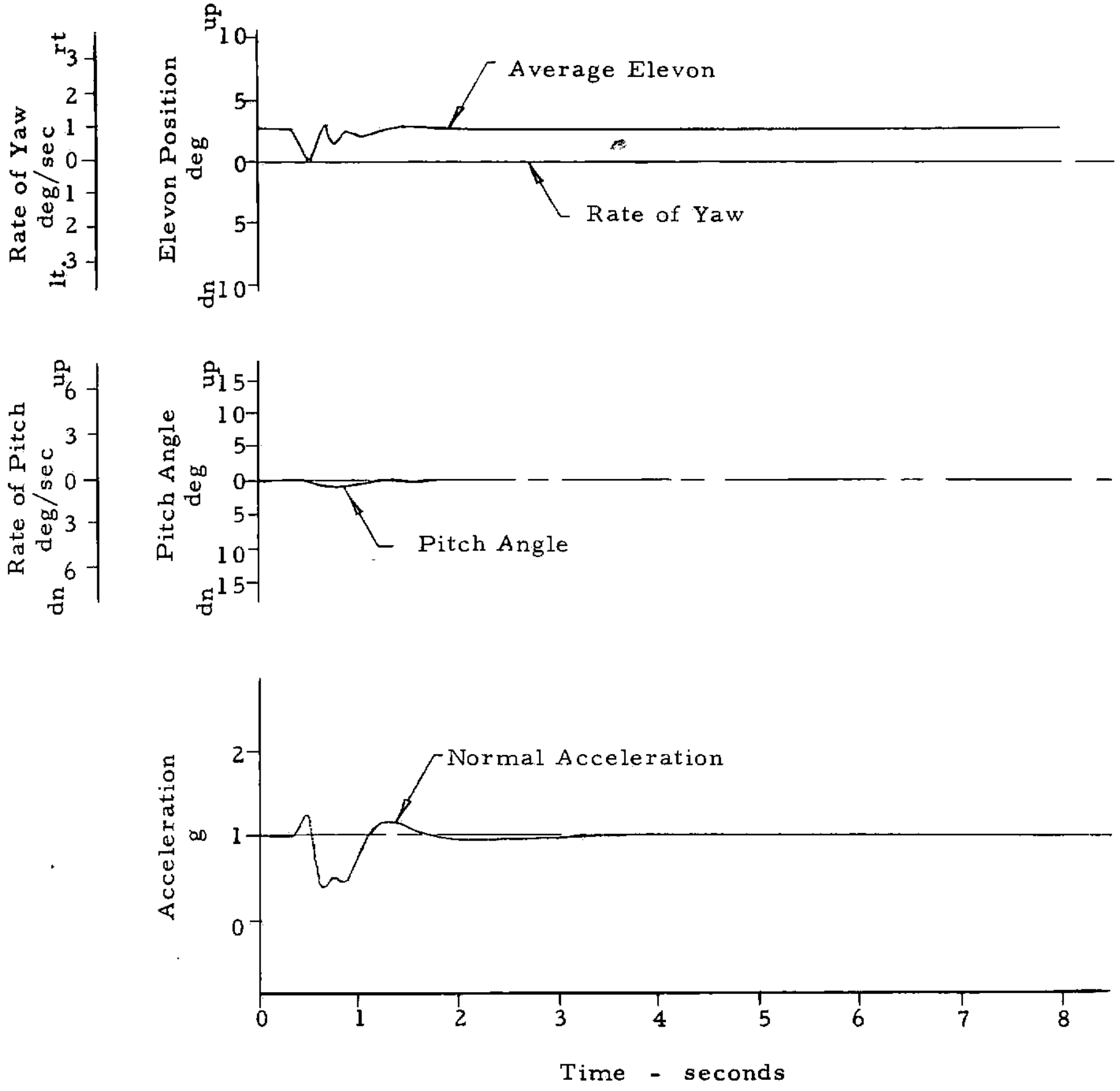


Figure No. 7

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
23400 Lbs. Weight
C.G. 27.1 percent MAC

Pitch Damper Off
Mach No. 1.1
Yaw Damper On

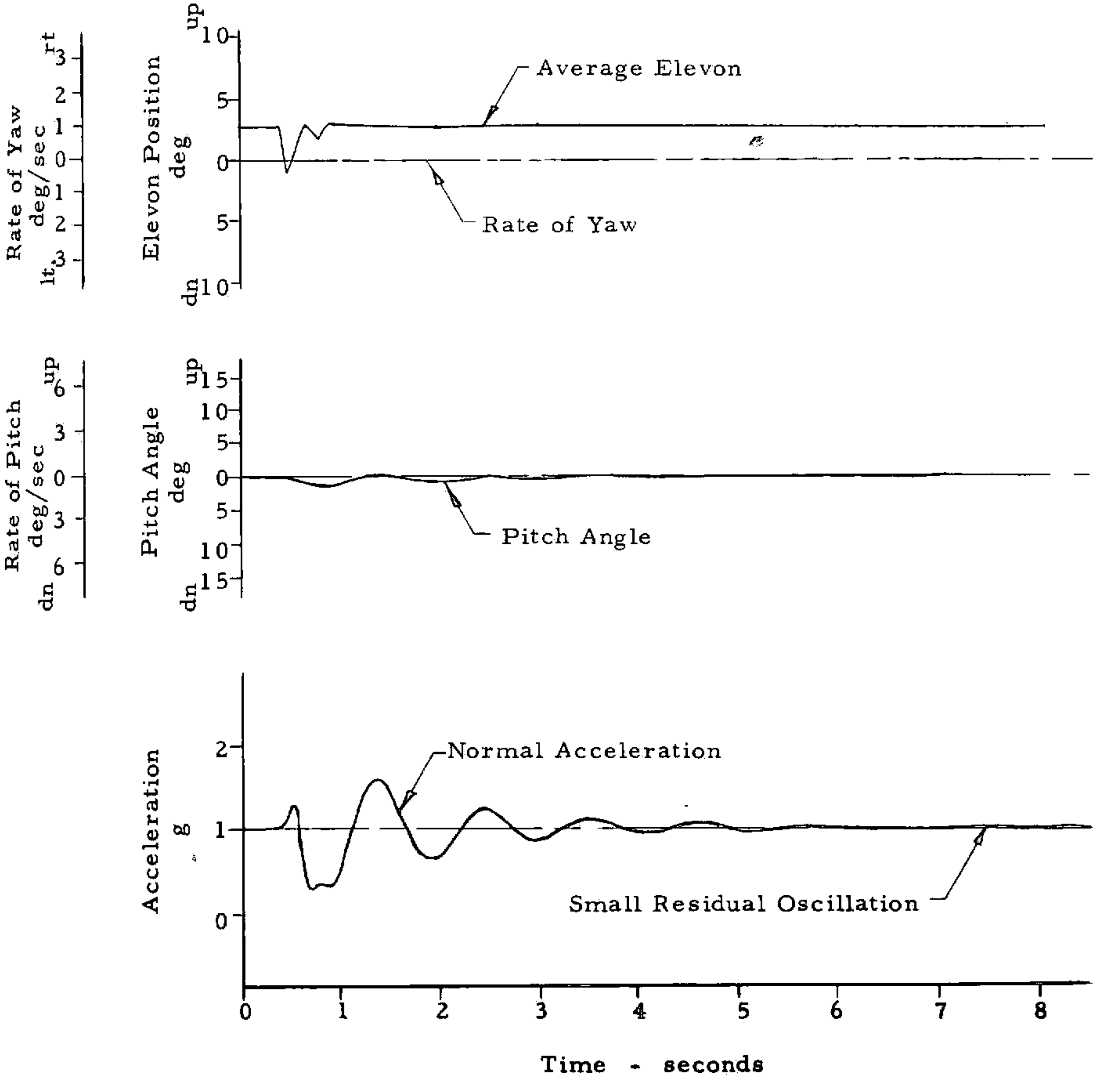


Figure No. 8

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration Pitch Damper On

35,000 Altitude Mach No. 1.1

24600 Lbs. Weight Yaw Damper Off

C.G. 26.9 percent MAC

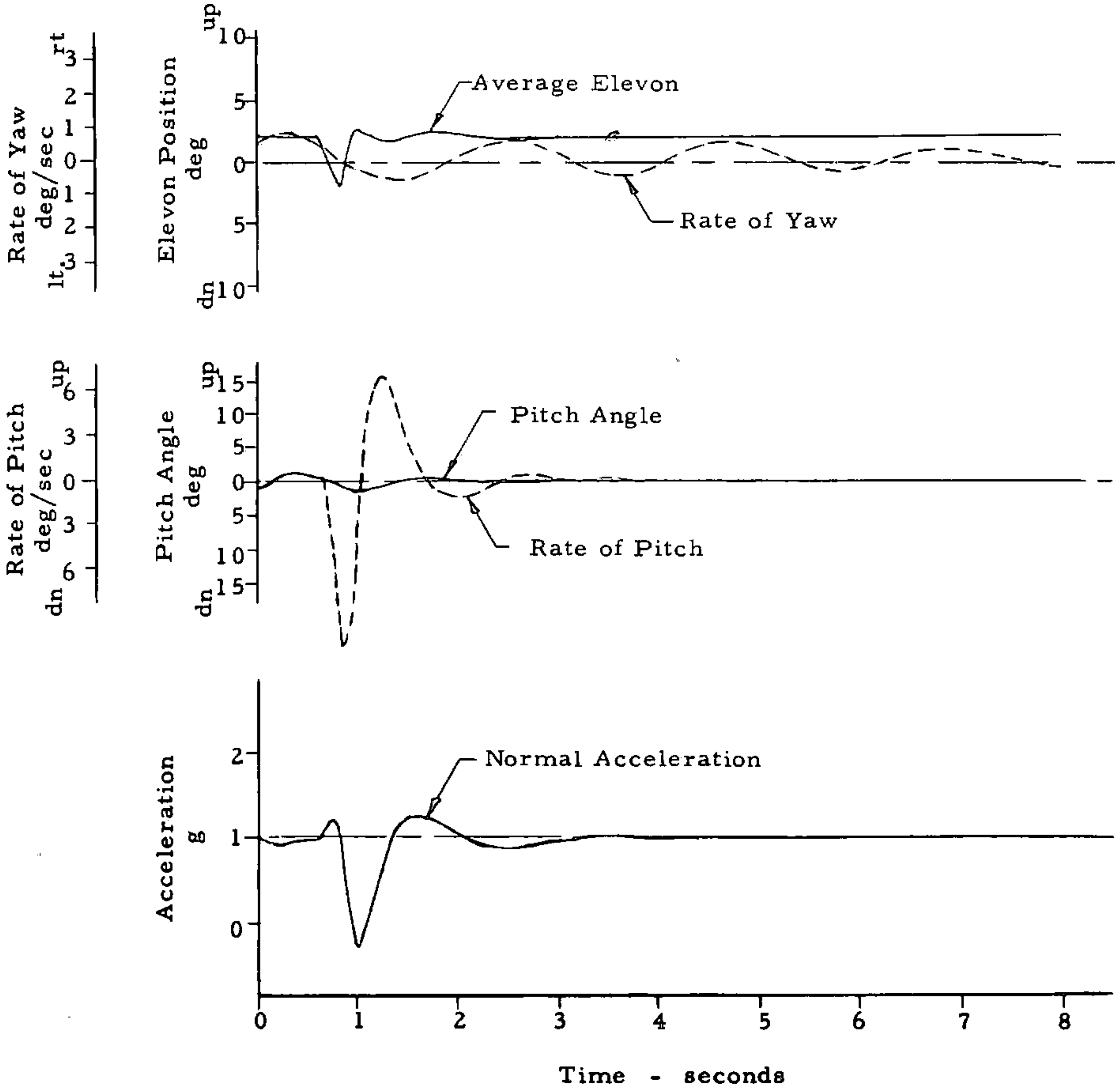


Figure No. 9

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
24600 Lbs. Weight

Pitch Damper Off
Mach No. 1.1
Yaw Damper Off

C.G. 26.9 percent MAC

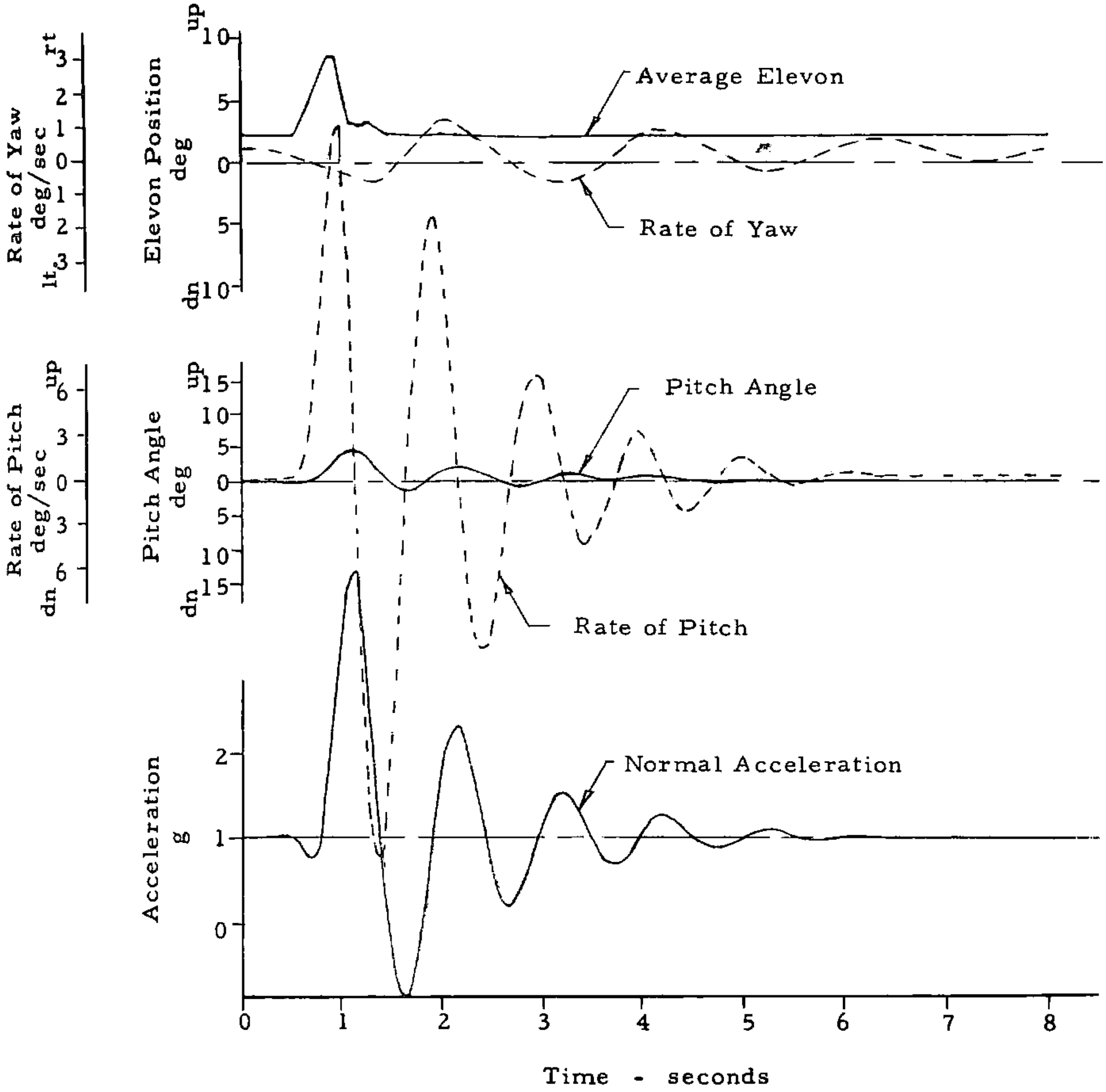


Figure No. 10

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
24000 Lbs. Weight

Pitch Damper On
Mach No. 0.9
Yaw Damper On

C.G. 27.0 percent MAC

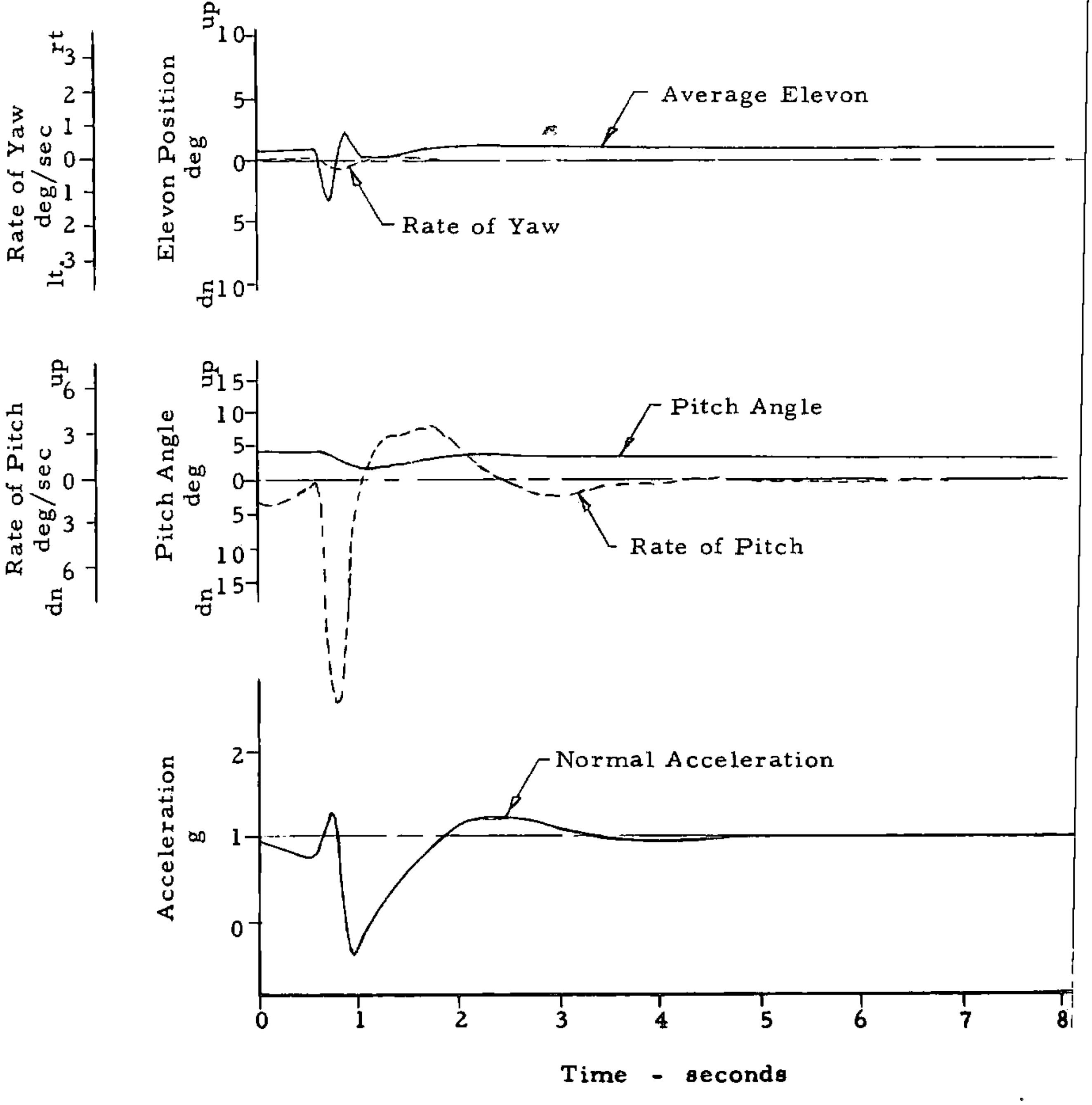


Figure No. 11

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
24200 Lbs. Weight
Pitch Damper Off
Mach No. 0.9
Yaw Damper Off

C.G. 27.0 percent MAC

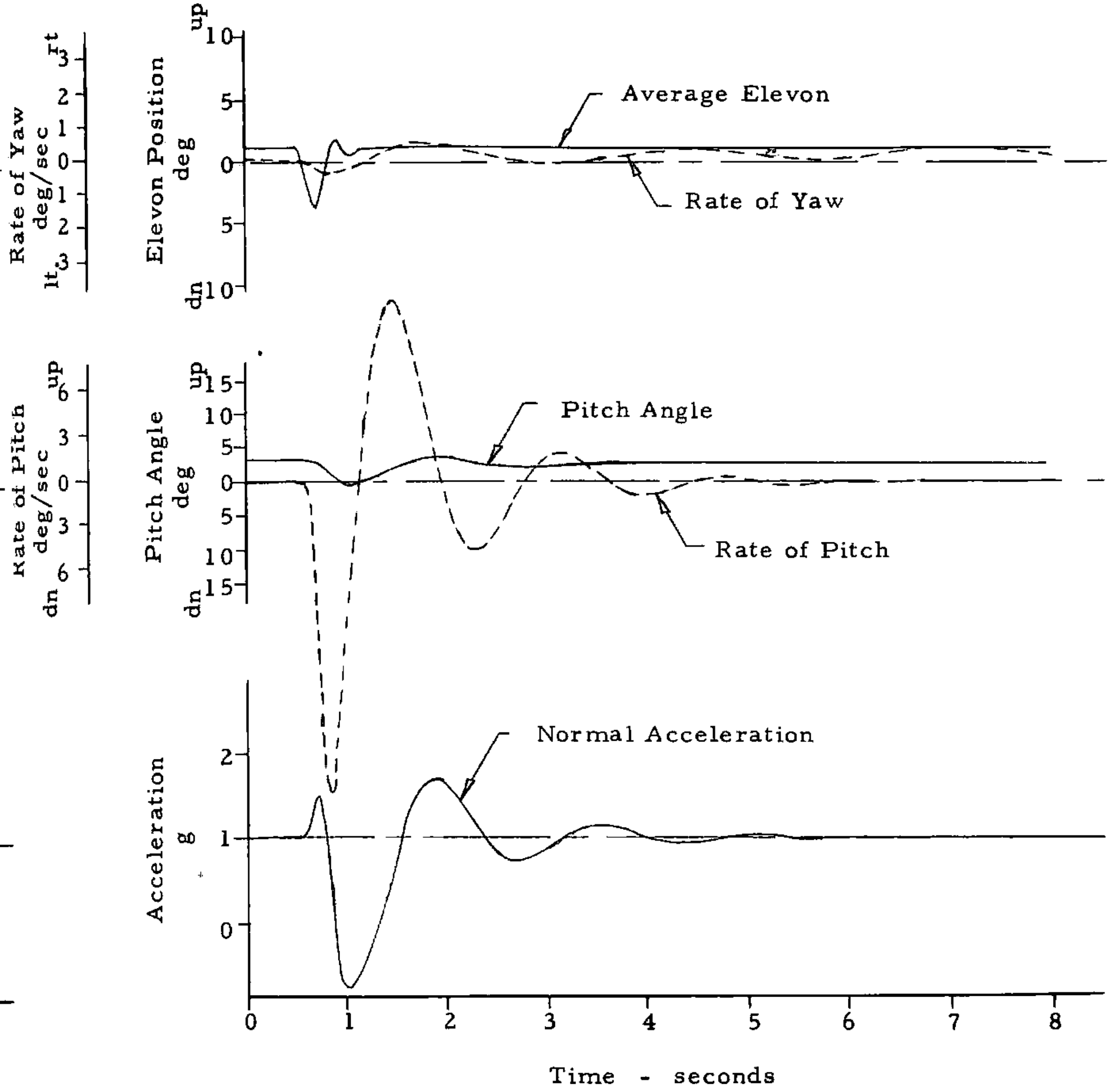


Figure No. 12

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
Pitch Damper On
35,000 Altitude
Mach No. 0.7
24000 Lbs. Weight
Yaw Damper On
C.G. 27.0 percent MAC

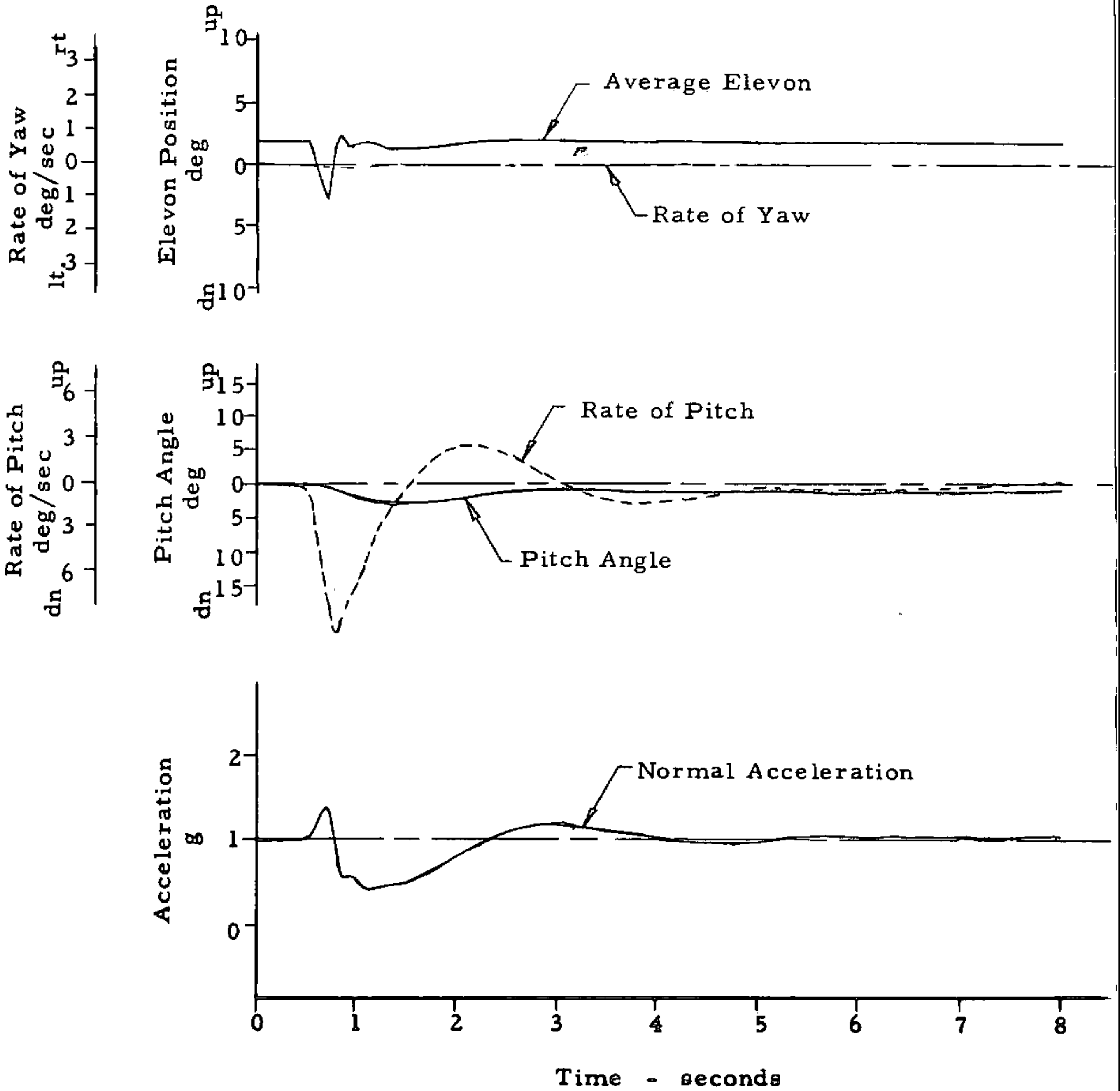


Figure No. 13

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration — Pitch Damper Off

35,000 Altitude — Mach No. 0.7

24000 Lbs. Weight — Yaw Damper Off

C.G. 27.0 percent MAC

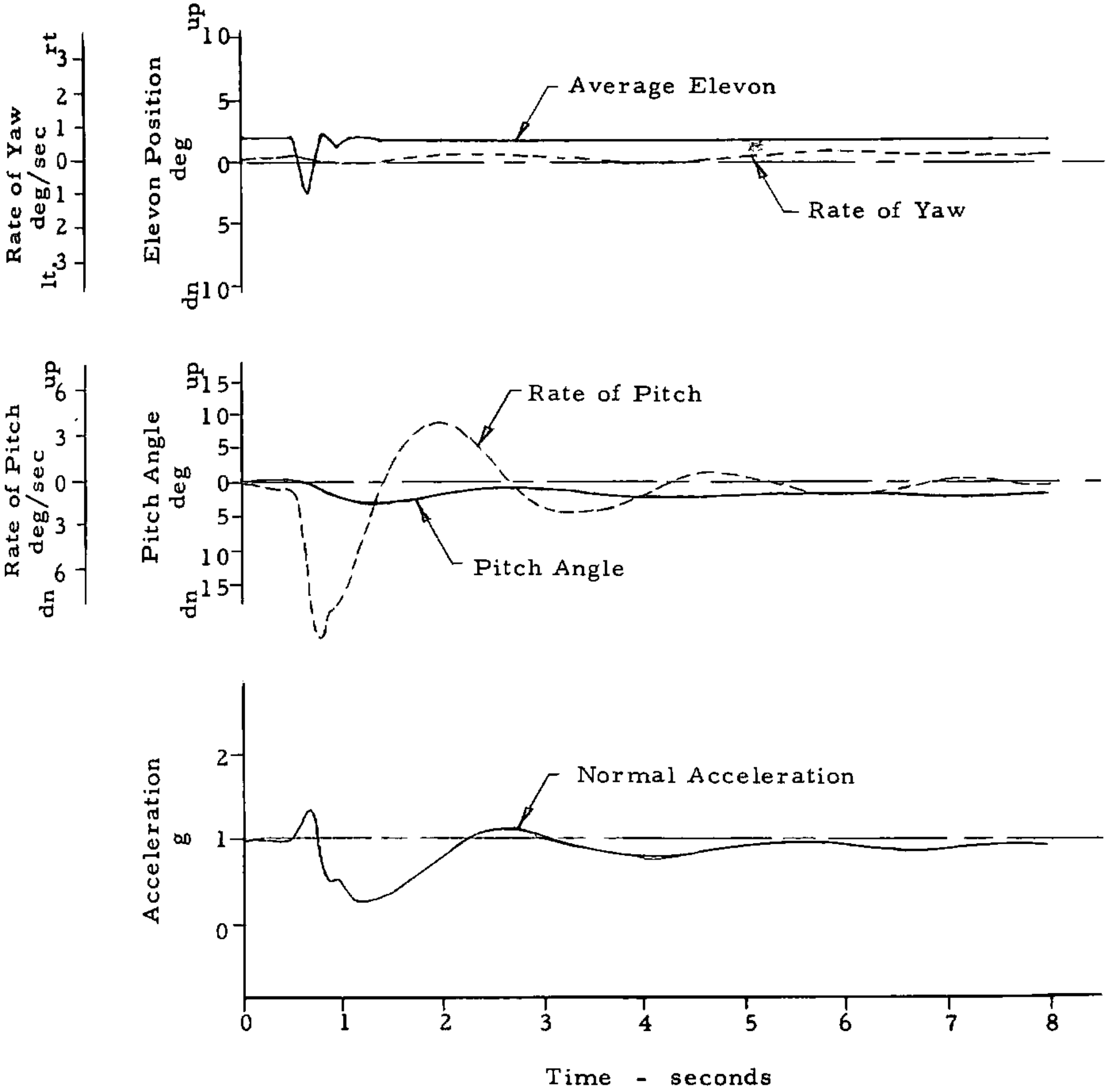
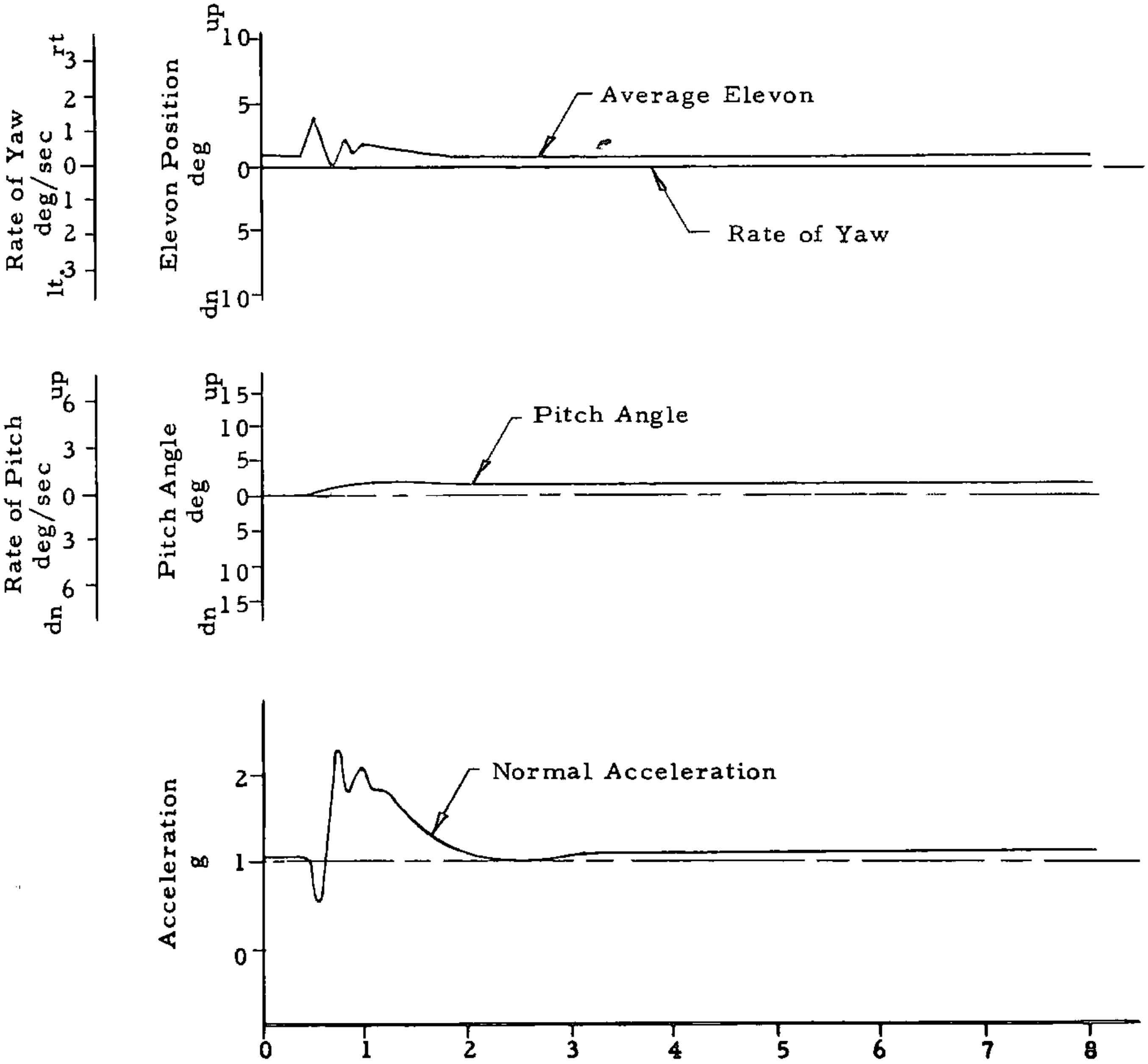


Figure No. 14

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise	Configuration	Pitch Damper	On
10,000	Altitude	Mach No.	0.70
22800	Lbs. Weight	Yaw Damper	On

C.G. 27.0 percent MAC



Time - seconds

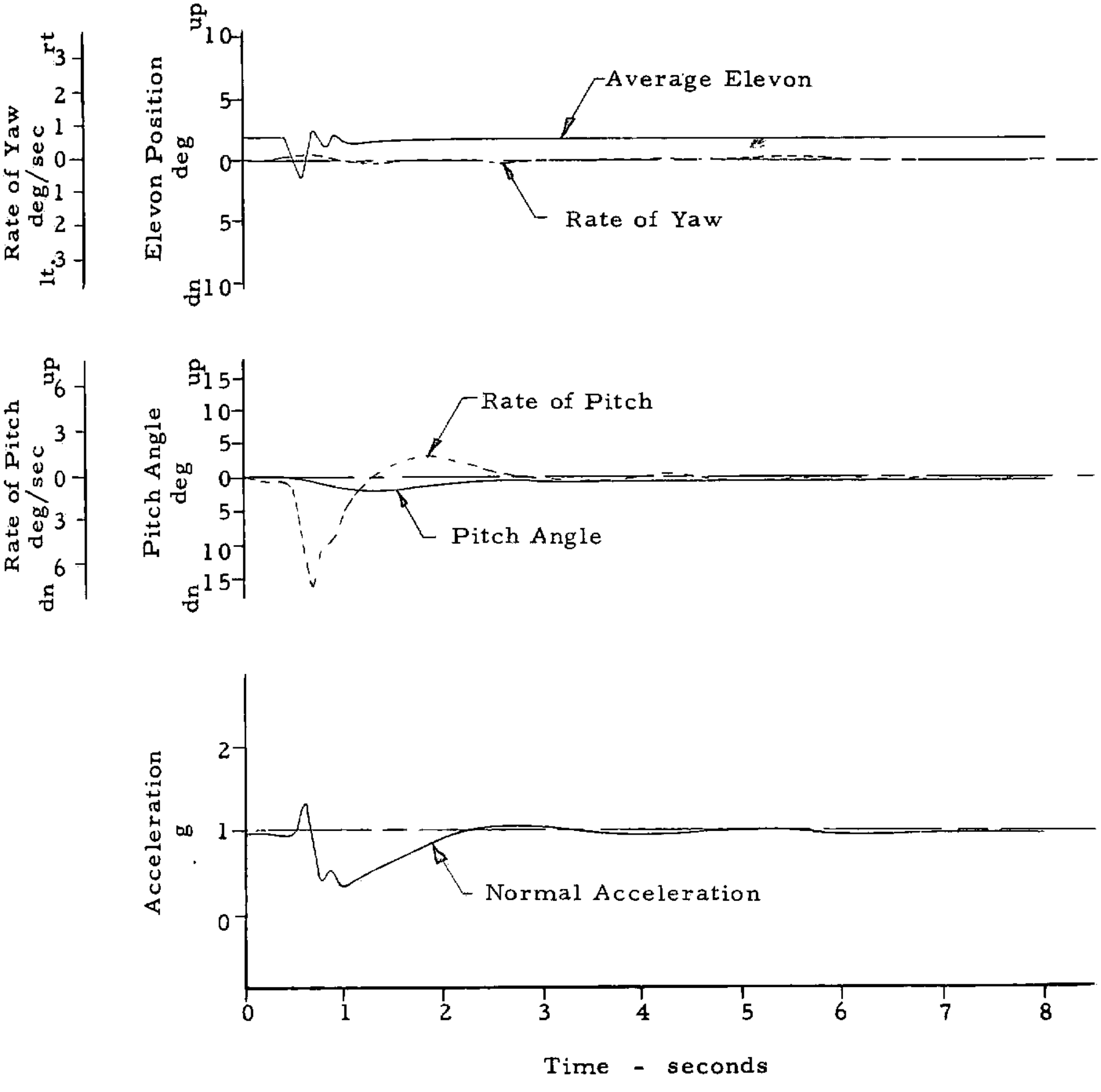
APPENDIX I

Figure No. 15

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration	Pitch Damper On
10,000 Altitude	Mach No. 0.5
22700 Lbs. Weight	Yaw Damper On

C.G. 27.0 percent MAC



APPENDIX I

Figure No. 16

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Power Approach Configuration
10,000 Altitude
21700 Lbs. Weight

Pitch Damper On
Mach No. 0.327
Yaw Damper On

C.G. 25.7 percent MAC

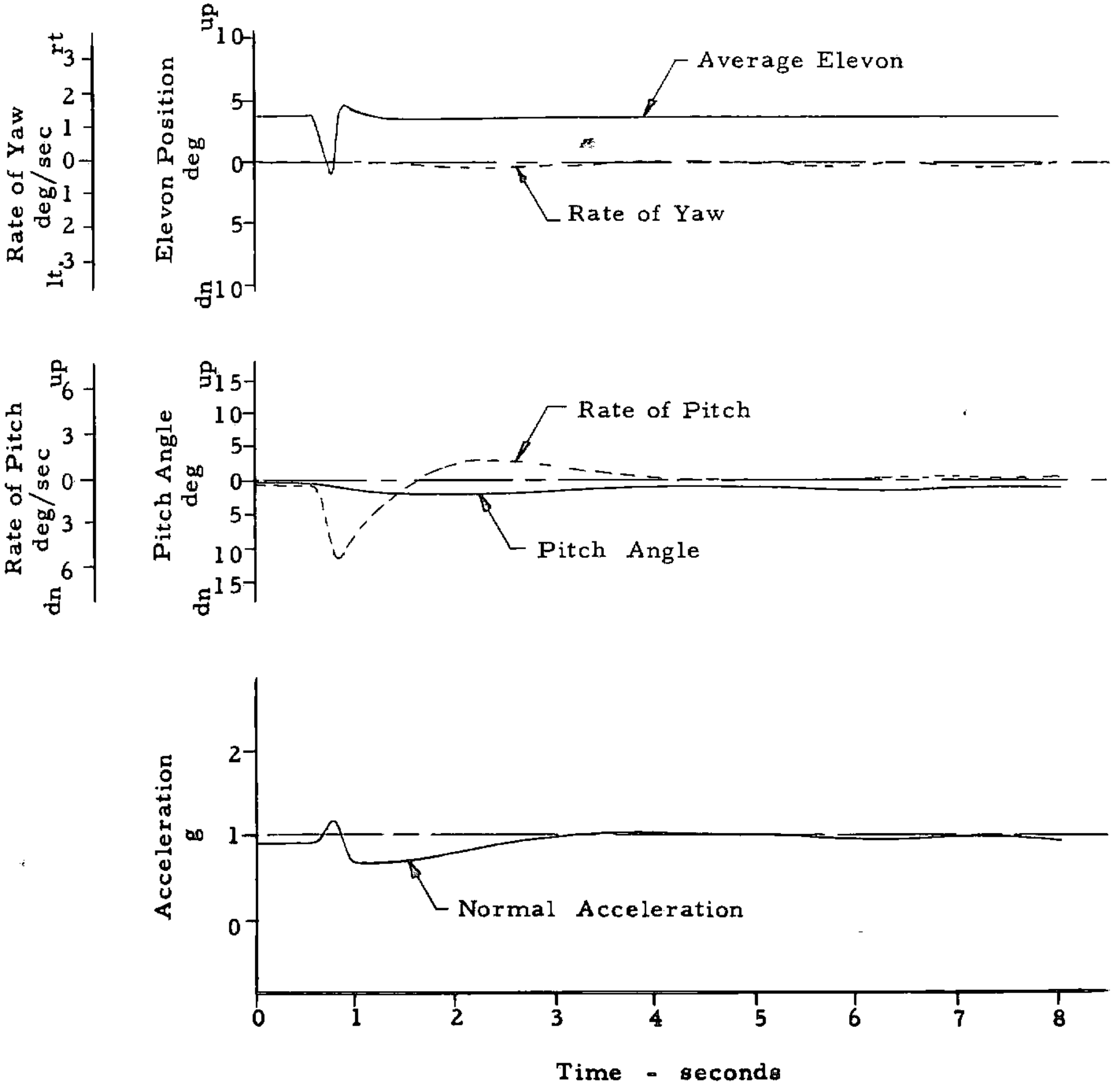


Figure No. 17

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Power Approach Configuration
10,000 Altitude
21700 Lbs. Weight

Pitch Damper Off
Mach No. 0.327
Yaw Damper On

C.G. 25.7 percent MAC

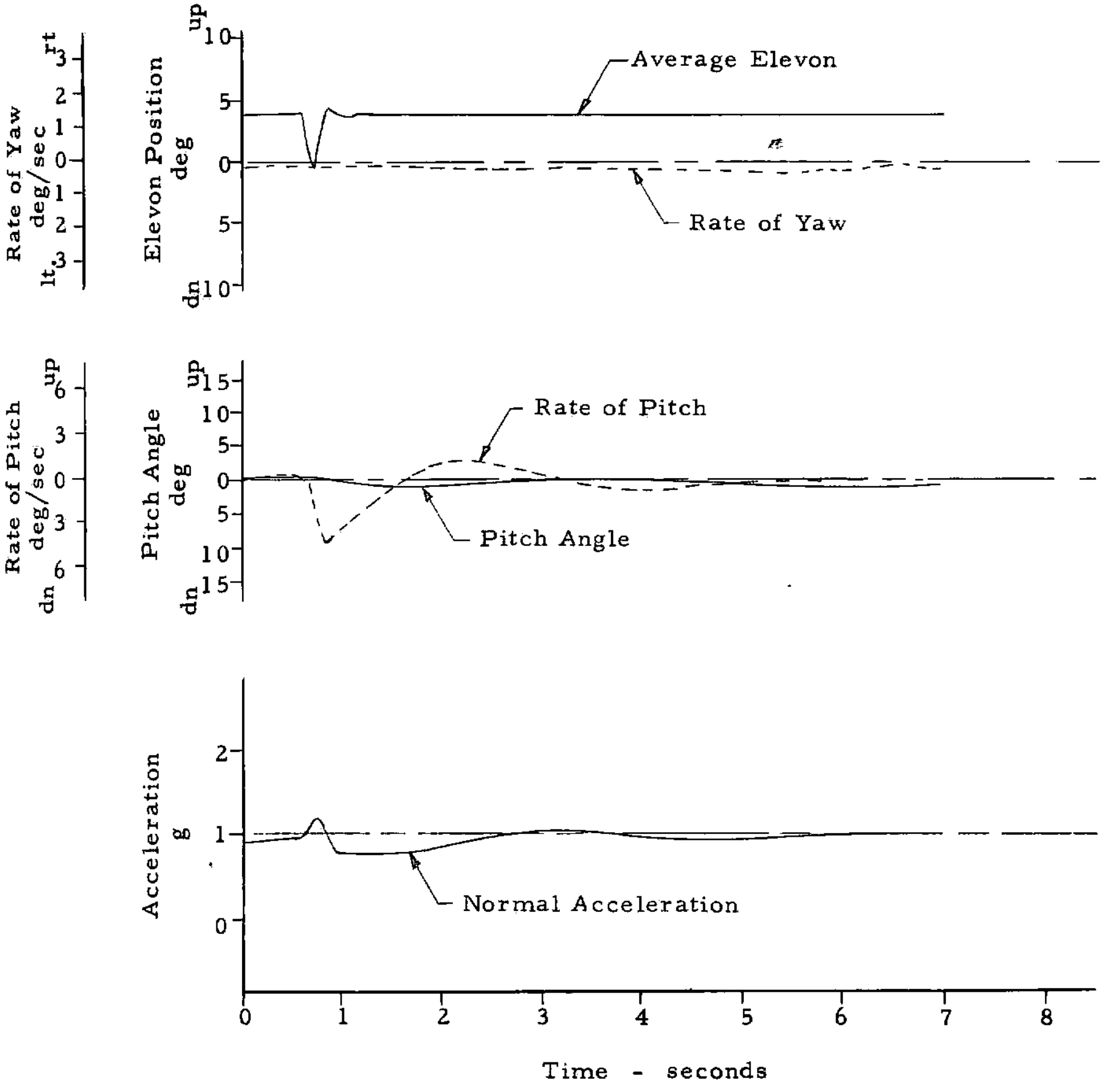


FIGURE NO. 18

STATIC LONGITUDINAL STABILITY

F-102A USAF NO. 53-1796

35000 FT ALTITUDE

TRIM SERVO ON AND OFF

CRUISE CONFIGURATION

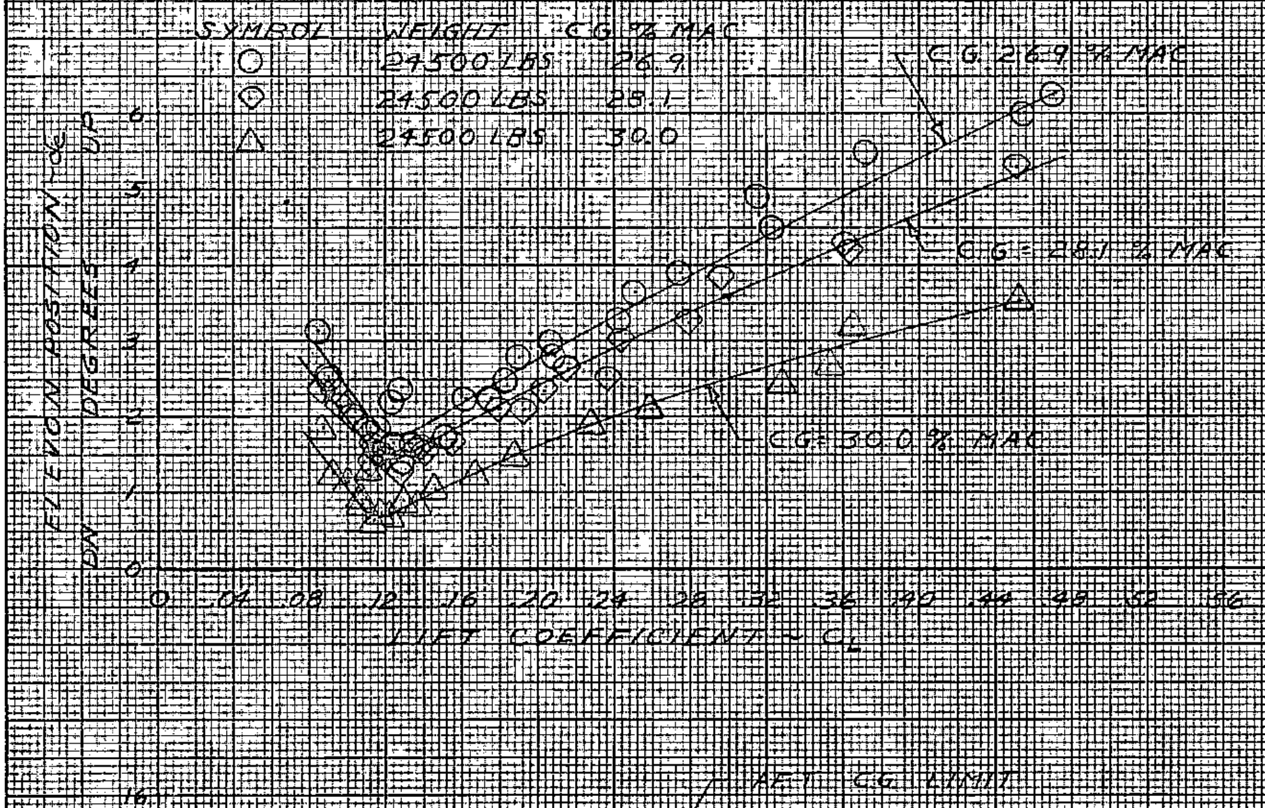


FIGURE NO. 19

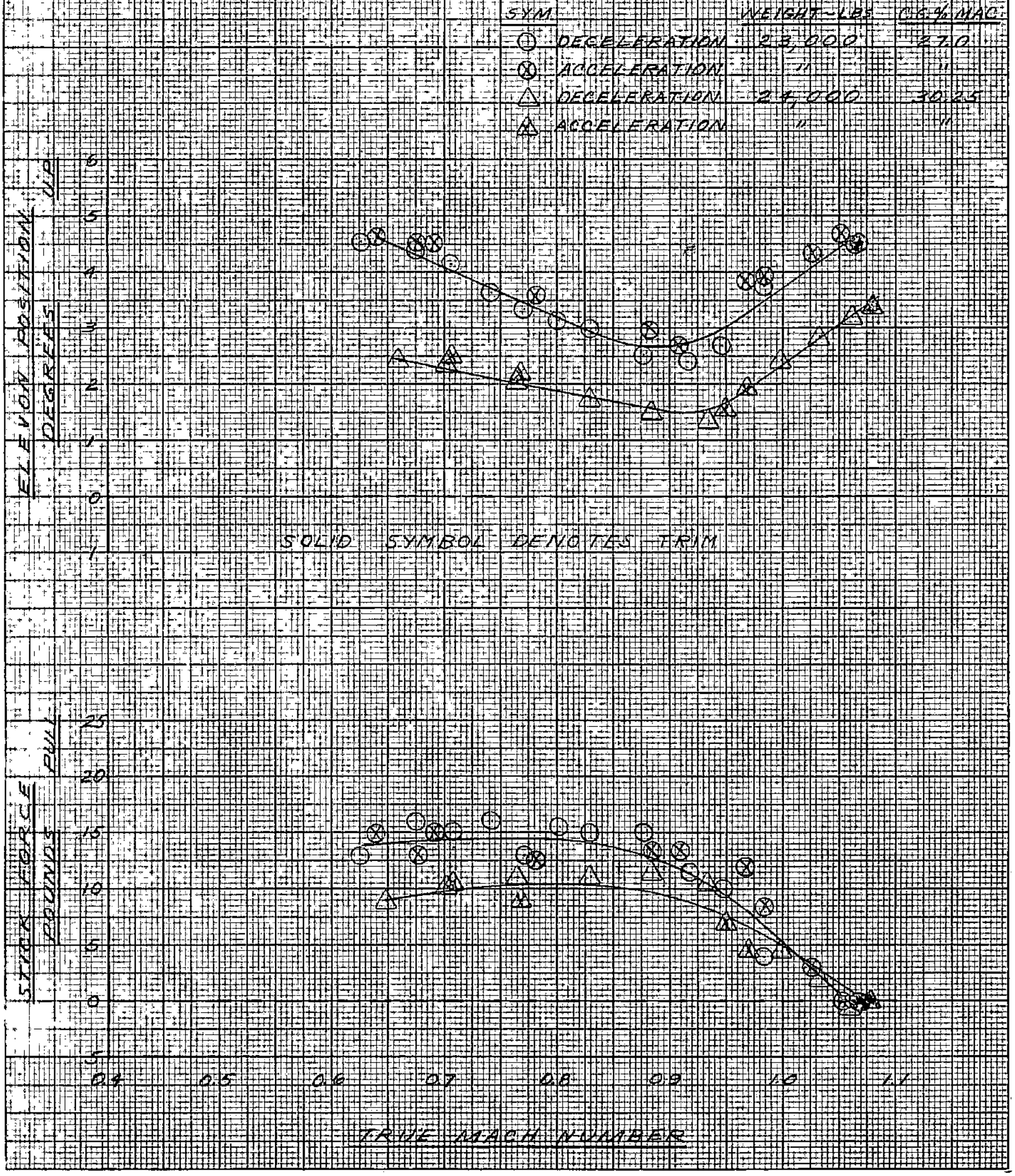
STATIC LONGITUDINAL STABILITY

F-102A USAF NO. 53-1796

45,000 FT. ALTITUDE

TRIM SERVO ON

CRUISE CONFIGURATION



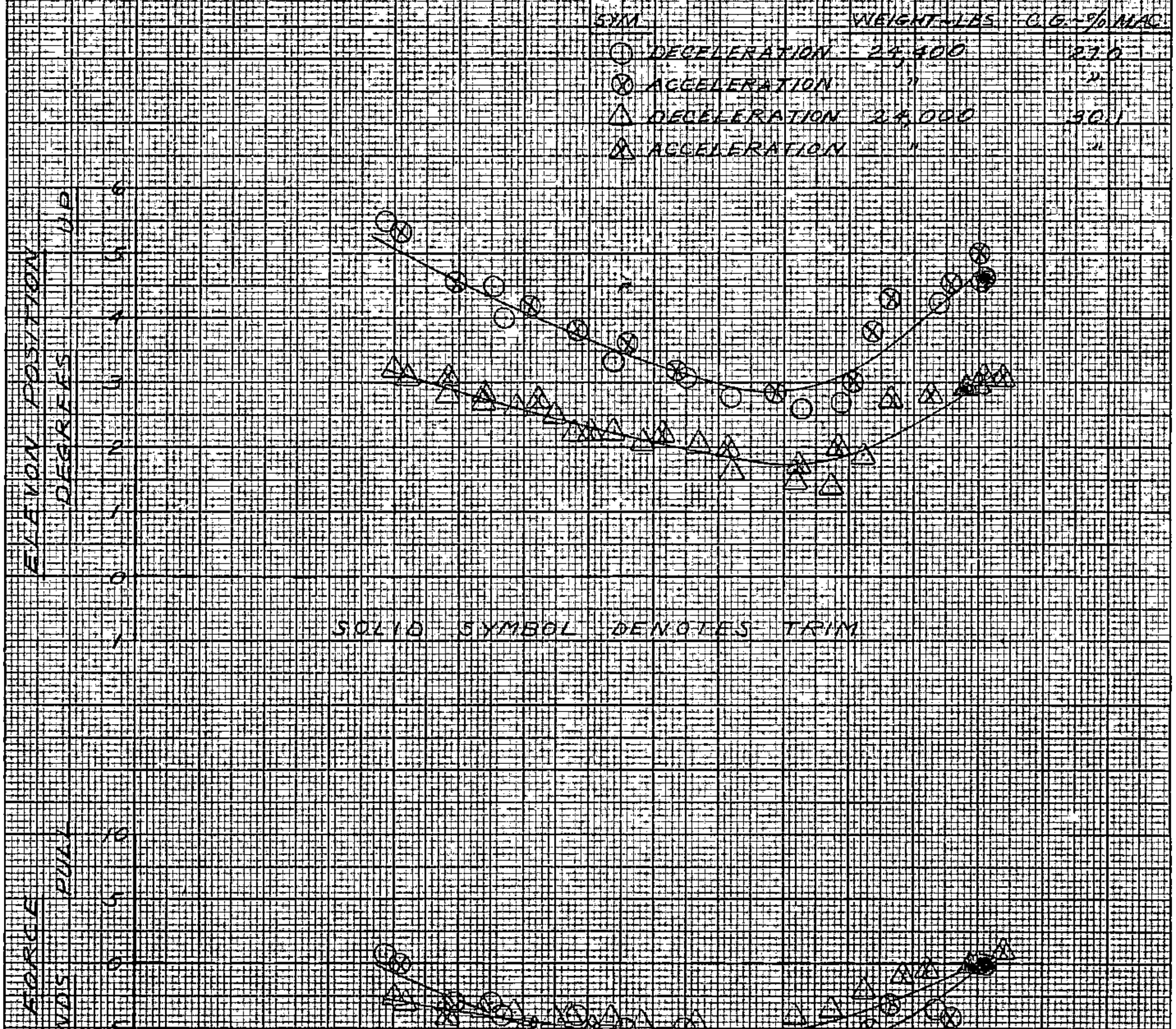
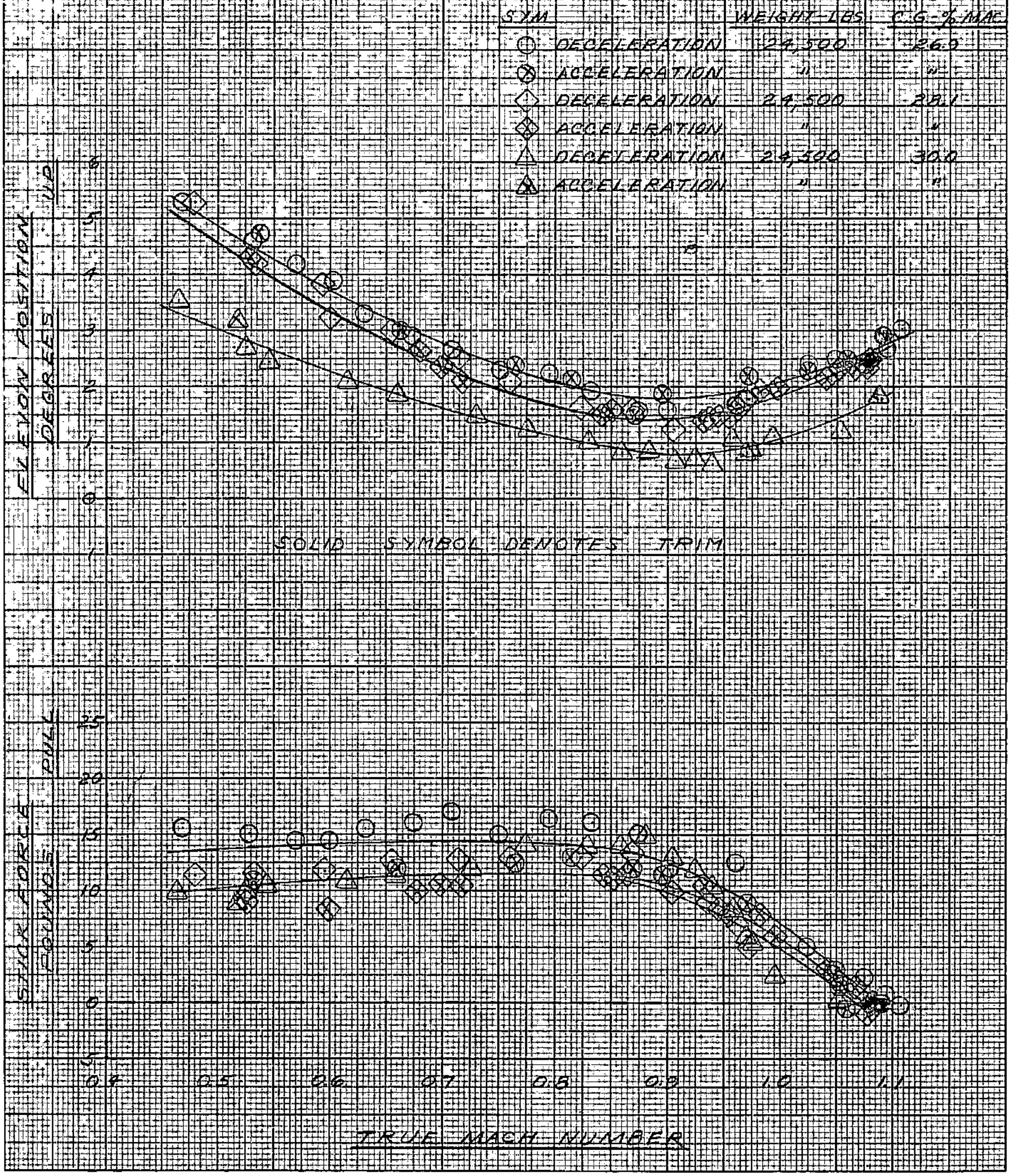


FIGURE NO 20
STATIC LONGITUDINAL STABILITY
F-102A USAF NO 53-1796
45,000 FT ALTITUDE
TRIM SERVO OFF
CRUISE CONFIGURATION

FIGURE NO. 21
STATIC LONGITUDINAL STABILITY
F-102A USAF NO. 53-1796
35,000 FT. ALTITUDE
TRIM SERVO ON
CRUISE CONFIGURATION



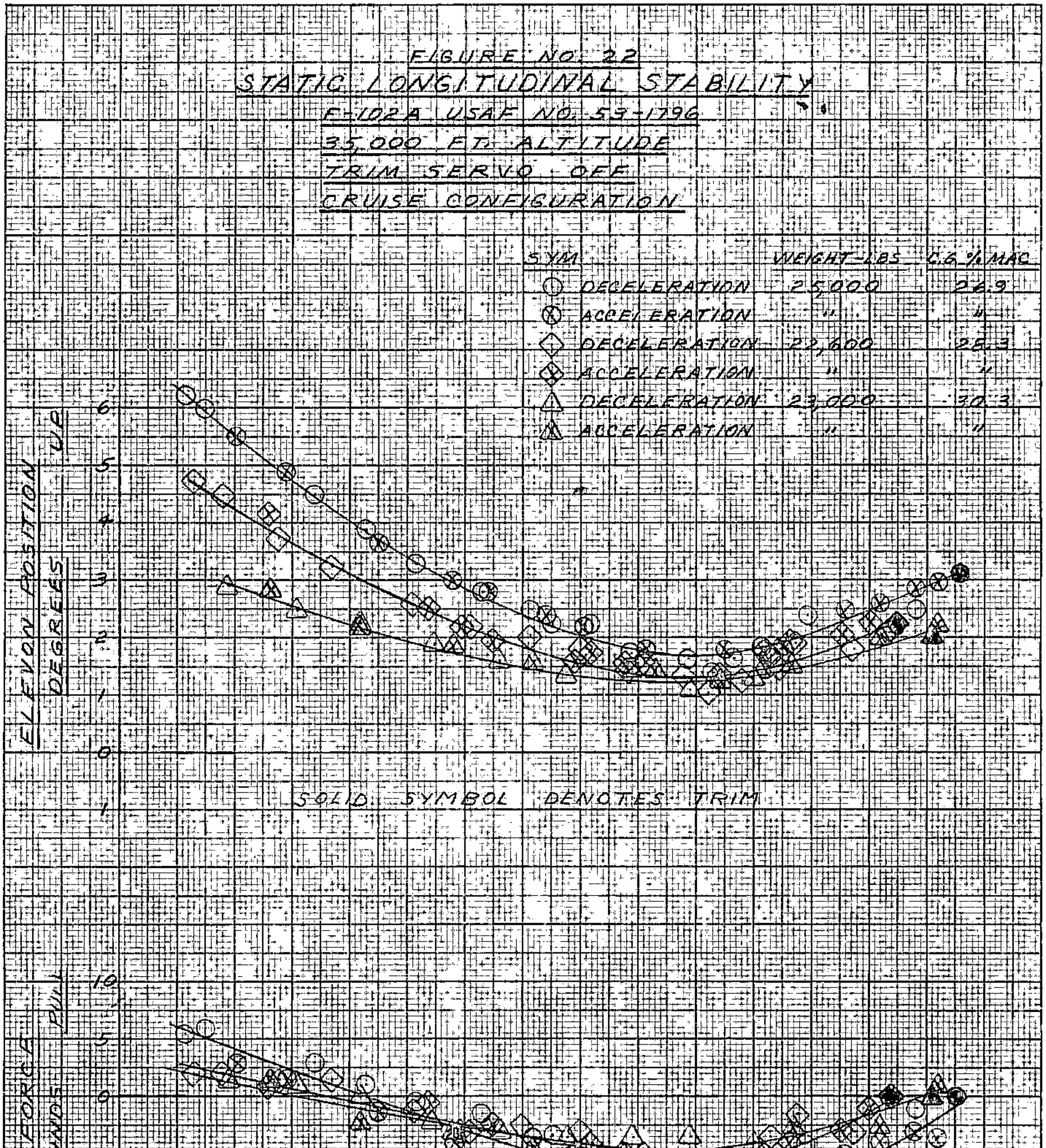


FIGURE NO. 22

STATIC LONGITUDINAL STABILITY

F-102A USAF NO. 53-1796

35,000 FT. ALTITUDE

TRIM SERVO OFF

CRUISE CONFIGURATION

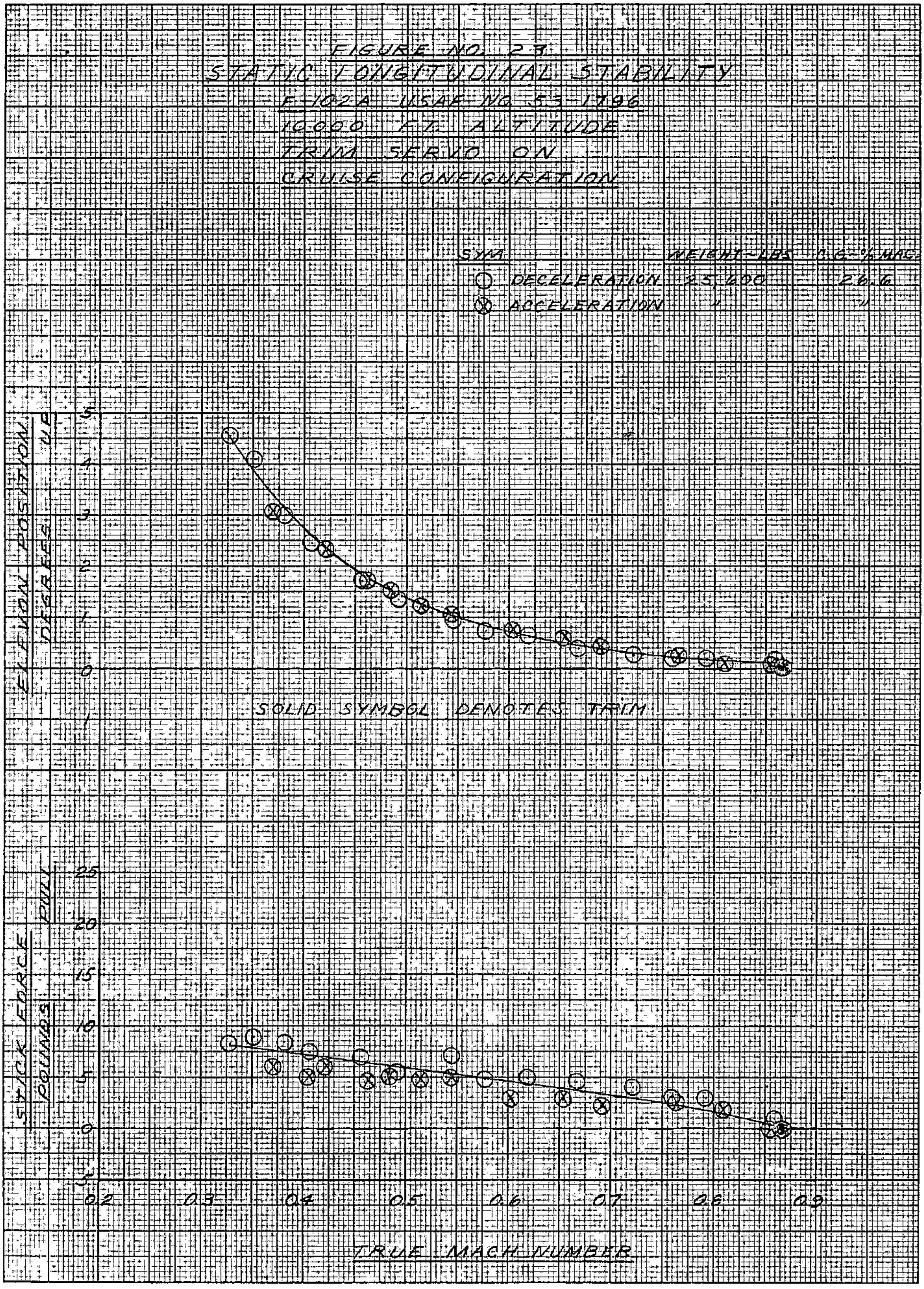


FIGURE NO. 23
STATIC LONGITUDINAL STABILITY
F-102A USAF NO. 53-1796
10 000 FT. ALTITUDE
TRIM SERVO ON
CRUISE CONFIGURATION

SYM		WEIGHT-LBS	C.G.-% MAC.
○	DECELERATION	25,400	26.6
⊗	ACCELERATION	"	"

SOLID SYMBOL DENOTES TRIM

FIGURE NO. 24
STATIC LONGITUDINAL STABILITY
F-102A USAF NO. 53-1796
10,000 FT. ALTITUDE
TRIM SERVO OFF
CRUISE CONFIGURATION

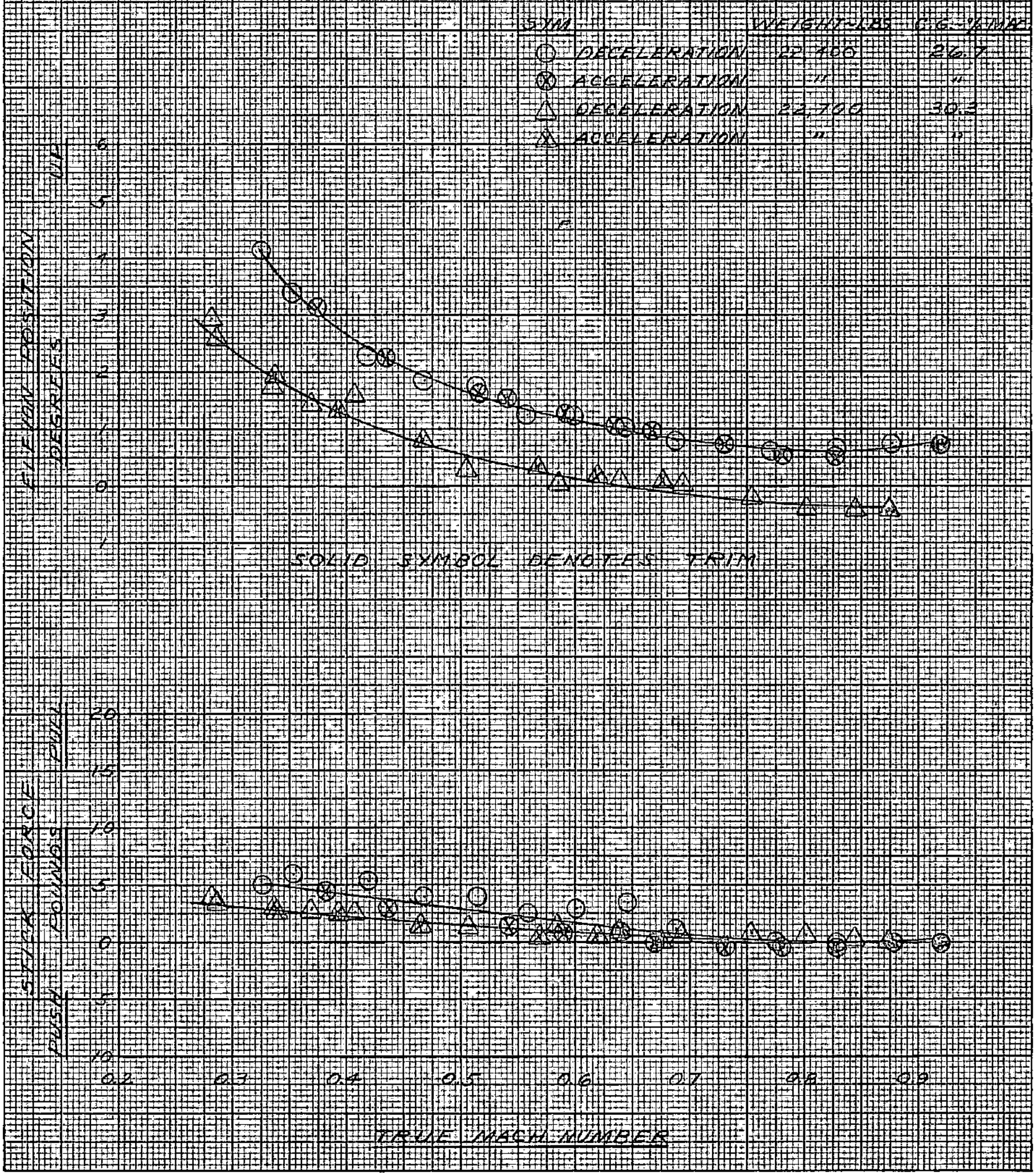


FIGURE NO. 25

STATIC LONGITUDINAL STABILITY

F-102A USAF NO. 53-1796
10,000 FT. ALTITUDE
TRIM SERVO OFF
POWER APPROACH CONFIGURATION
LANDING GEAR EXTENDED
SPEED BRAKES IN

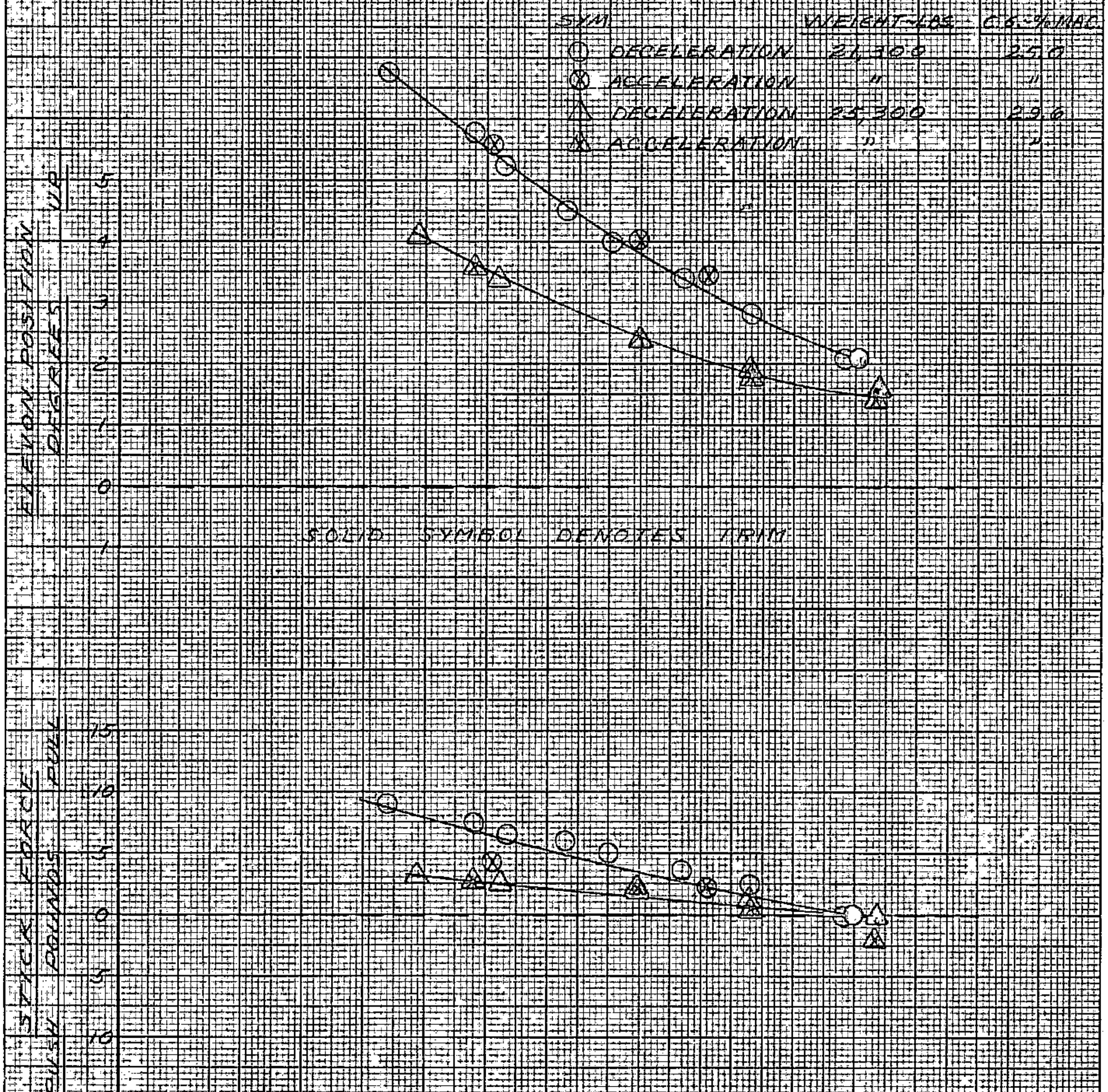


Figure No. 26

TIME HISTORY OF AFTERBURNER LIGHT
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
24,500 Weight

Pitch Damper On
Mach No. .72
C.G. 28.1 %MAC

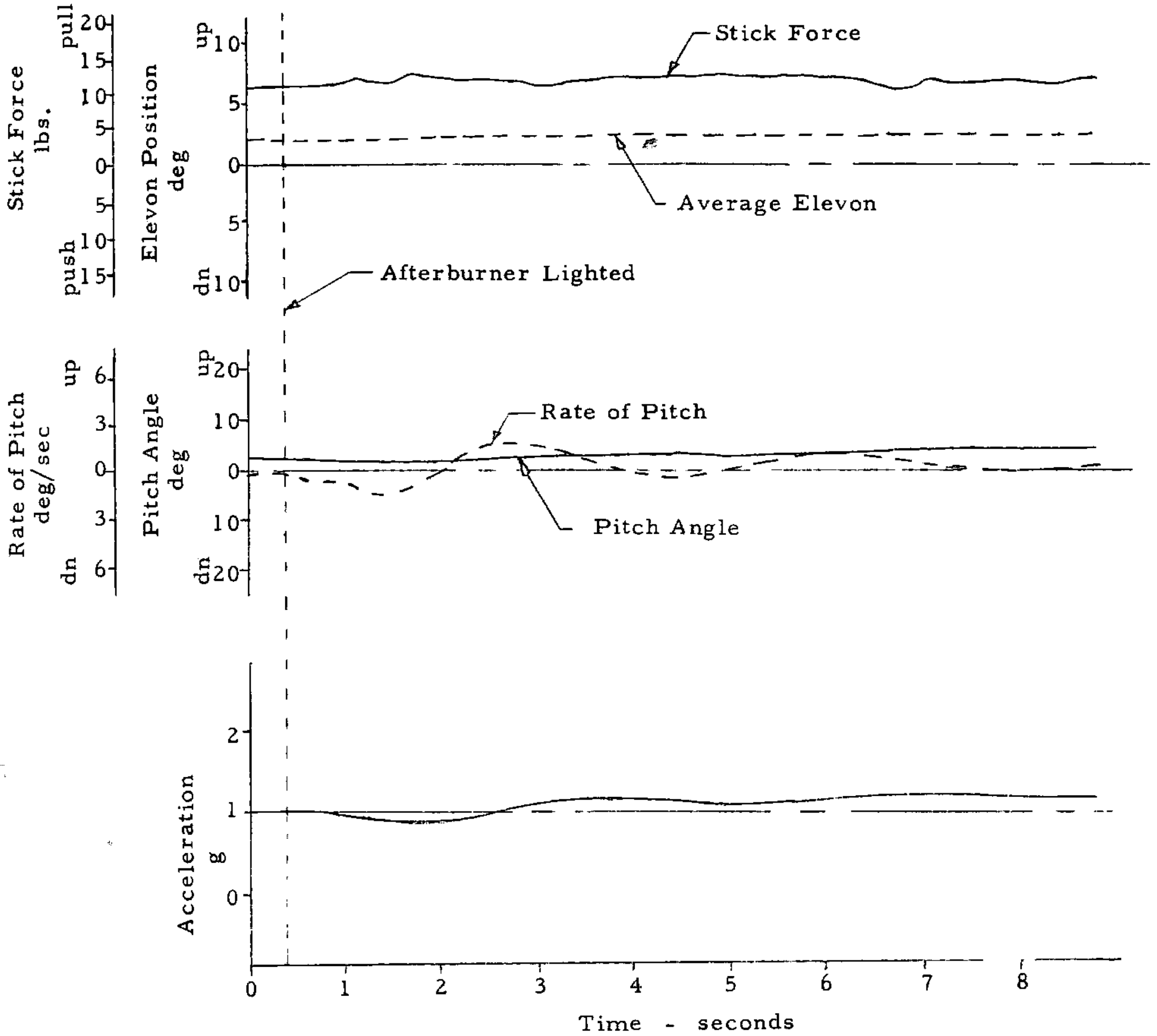


FIGURE NO. 27
MANEUVERING FLIGHT
F-102A USAF NO 53-1796
50500 FT. ALTITUDE
CRUISE CONFIGURATION
0.970 MACH NUMBER

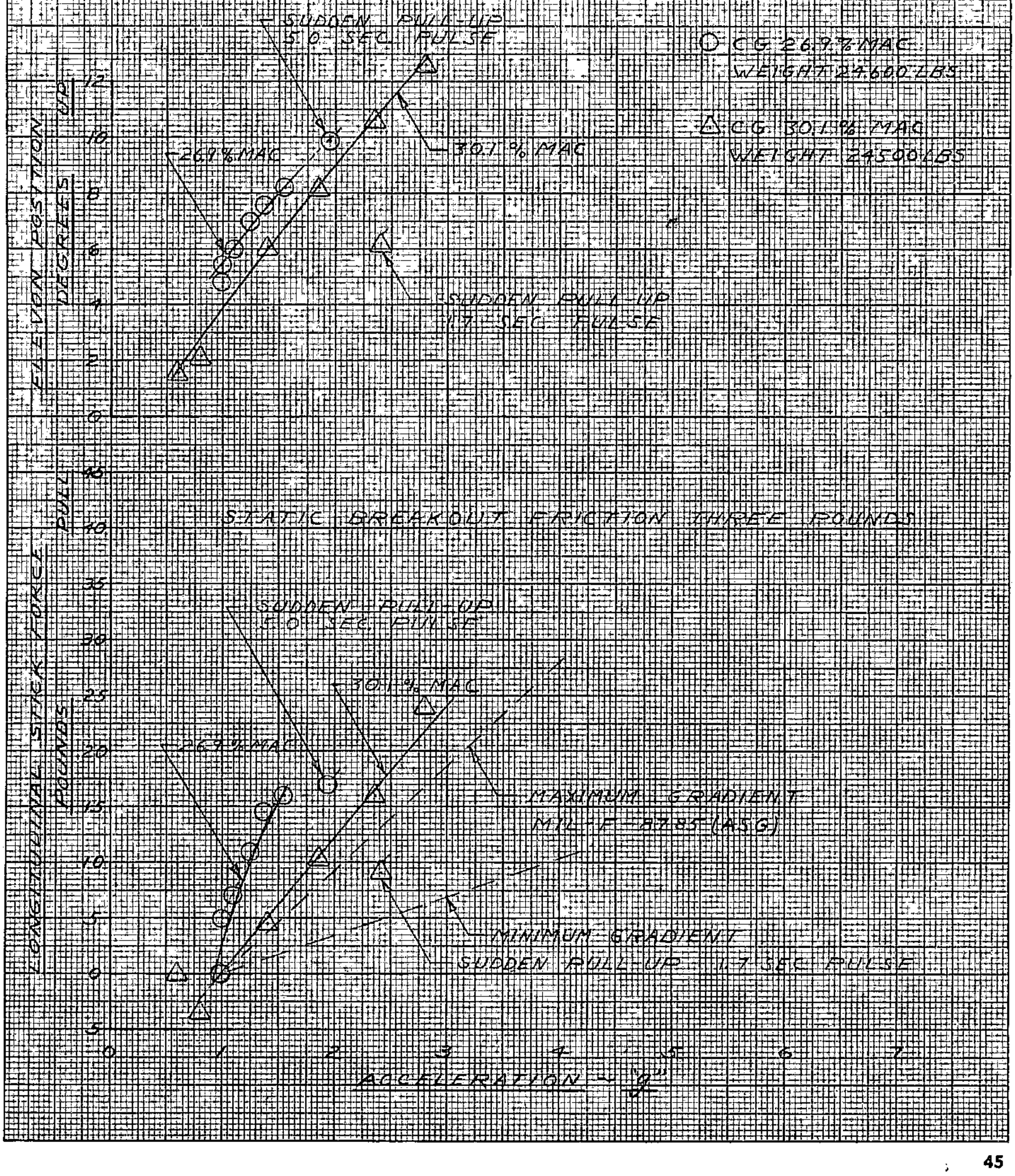


FIGURE NO. 28

MANEUVERING FLIGHT

F-102A USAF NO. 53-1796

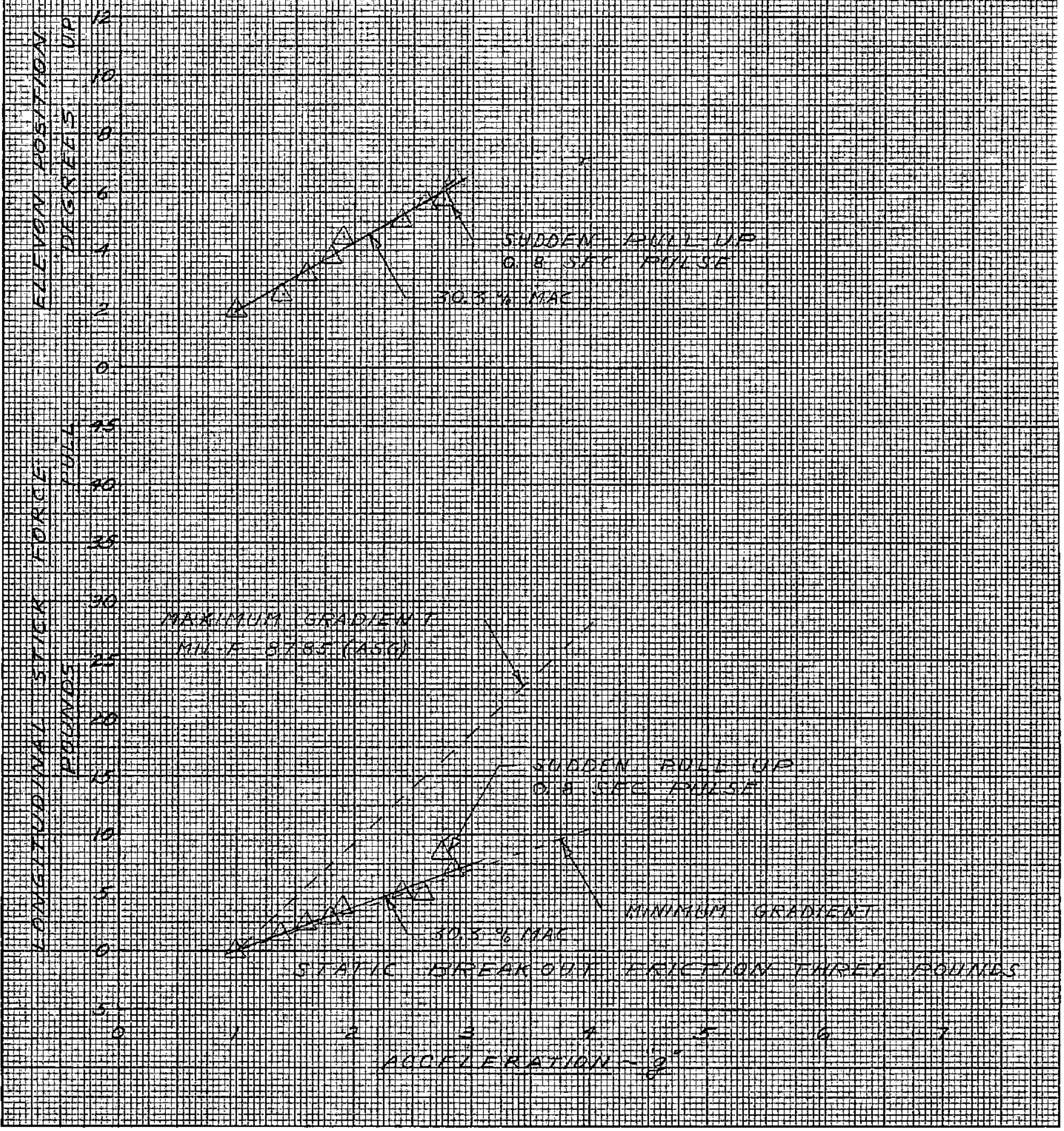
50500 FT. ALTITUDE

CRUISE CONFIGURATION

0.890 MACH NUMBER

C.G. 30.3% MAC

WEIGHT 23700 LBS



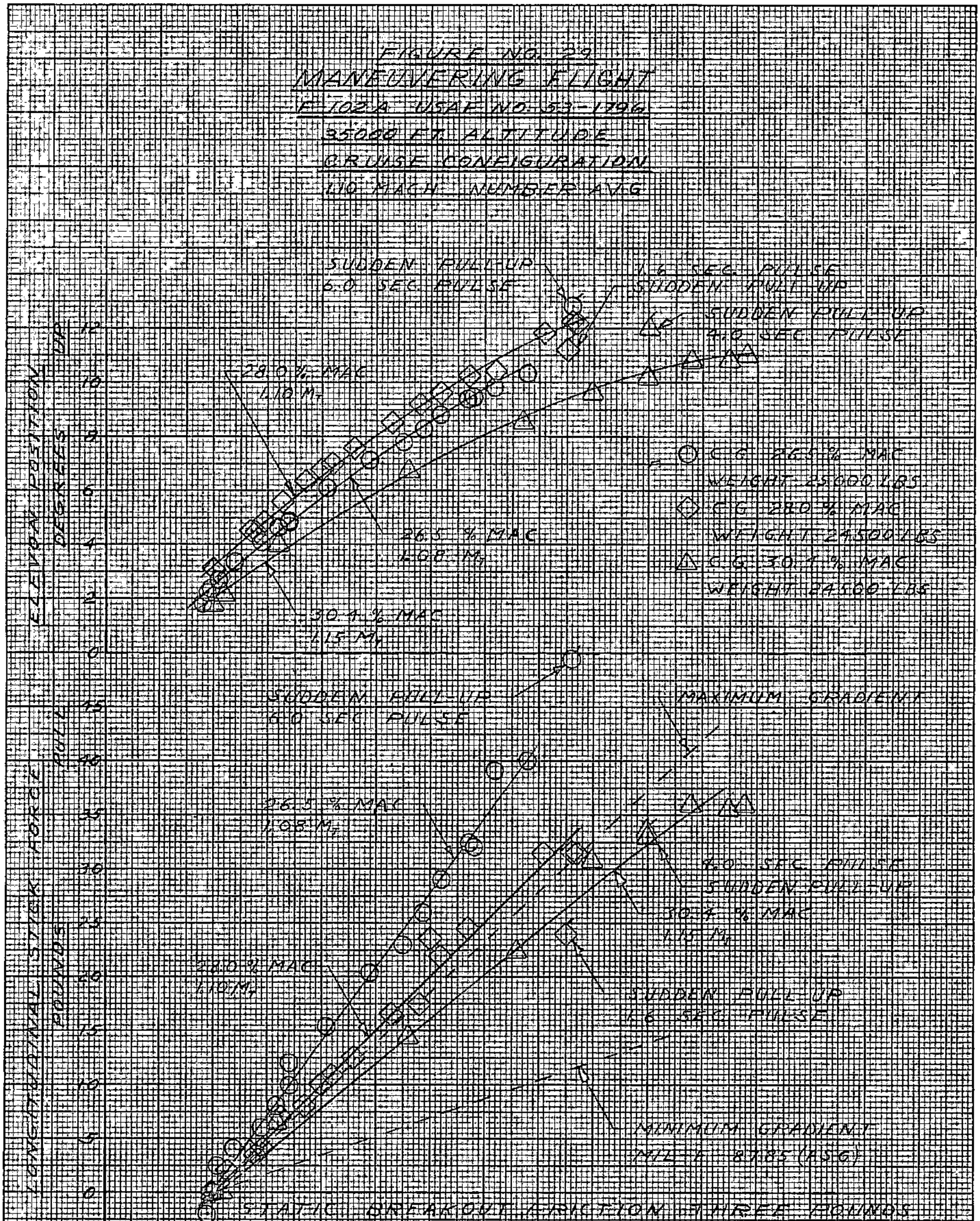


FIGURE NO. 29
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
35000 FT. ALTITUDE
CRUISE CONFIGURATION
1.10 MACH NUMBER AVG

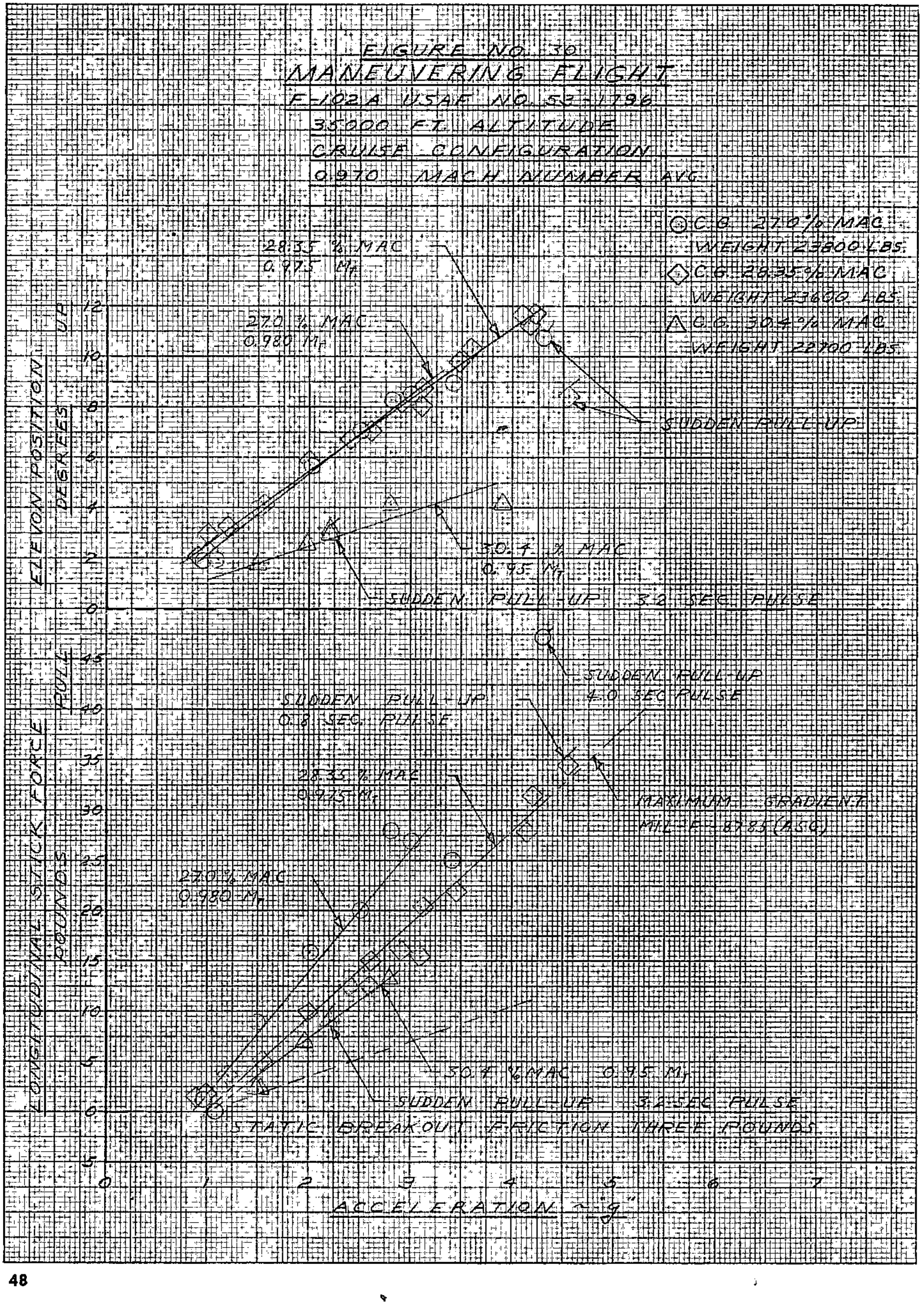


FIGURE NO. 30
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
35000 FT. ALTITUDE
CRUISE CONFIGURATION
0.970 MACH NUMBER AVG

FIGURE NO. 31
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
35000 FT. ALTITUDE
CRUISE CONFIGURATION
0.855 MACH NUMBER AVG

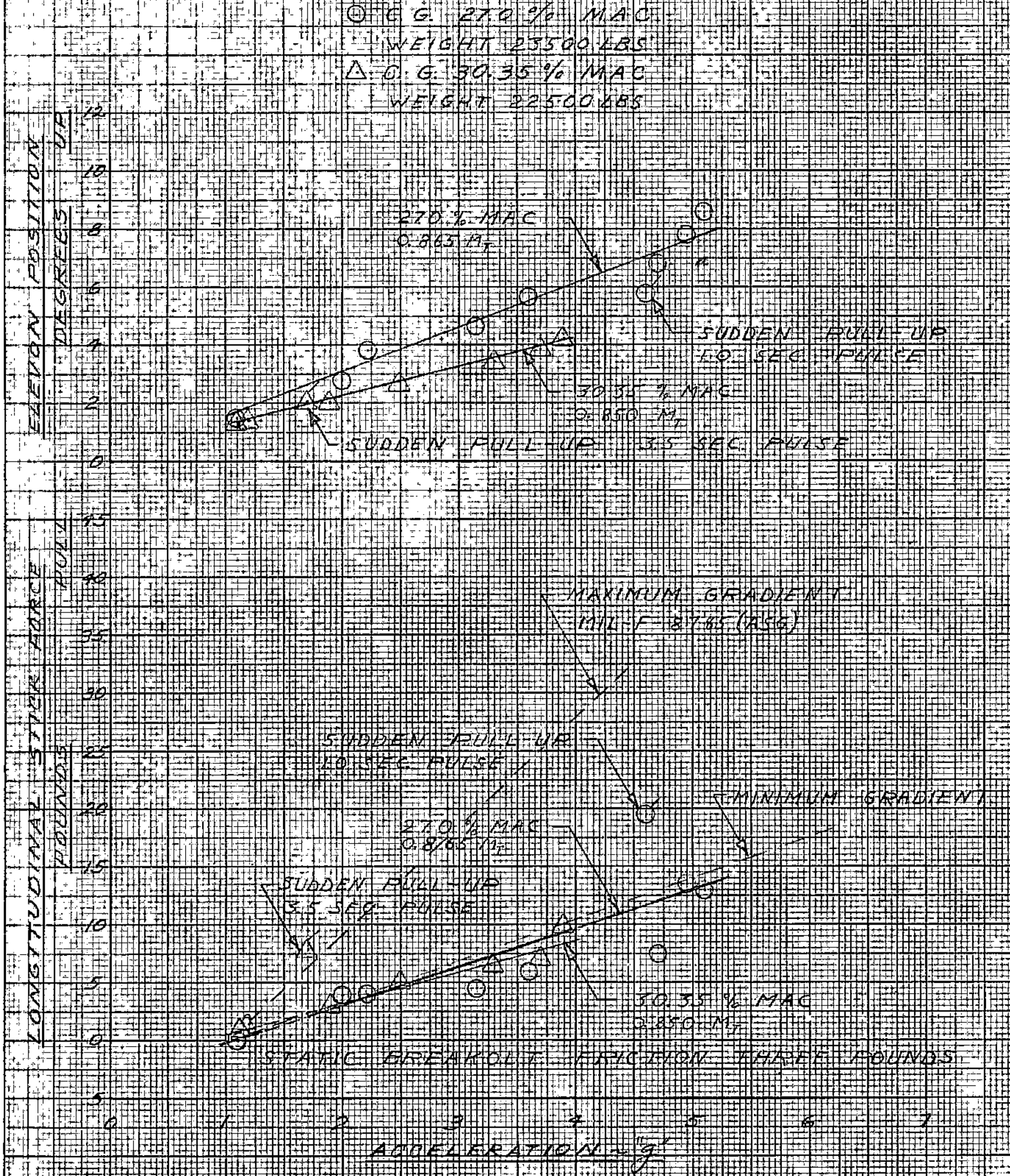


FIGURE NO. 32

MANEUVERING FLIGHT

F-102A USAF NO. 53-1196

35000 FT ALTITUDE

CRUISE CONFIGURATION

0.72 MACH NUMBER

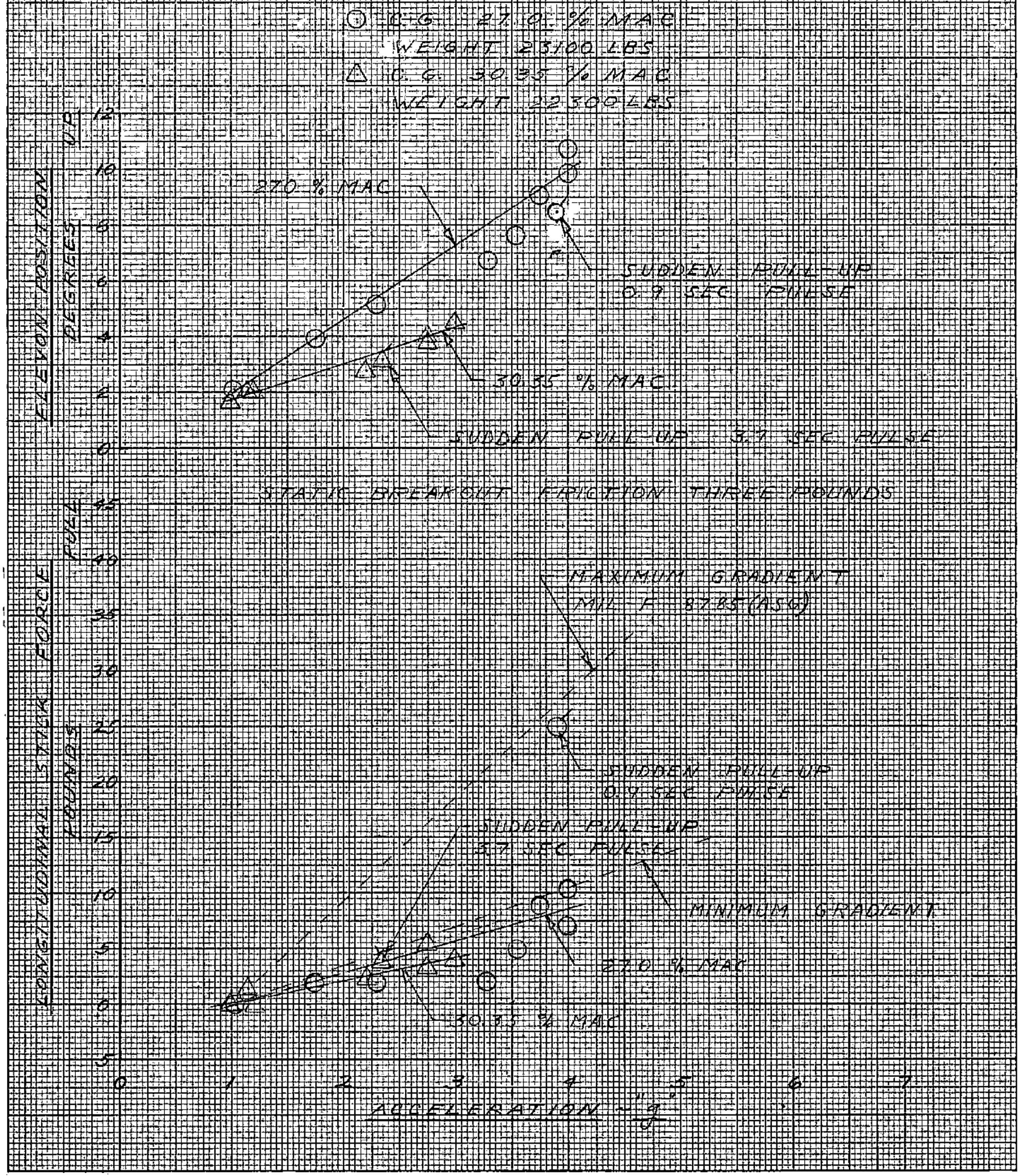


FIGURE NO. 33
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
10000 FT. ALTITUDE
CRUISE CONFIGURATION
0.90 MACH NUMBER AVG

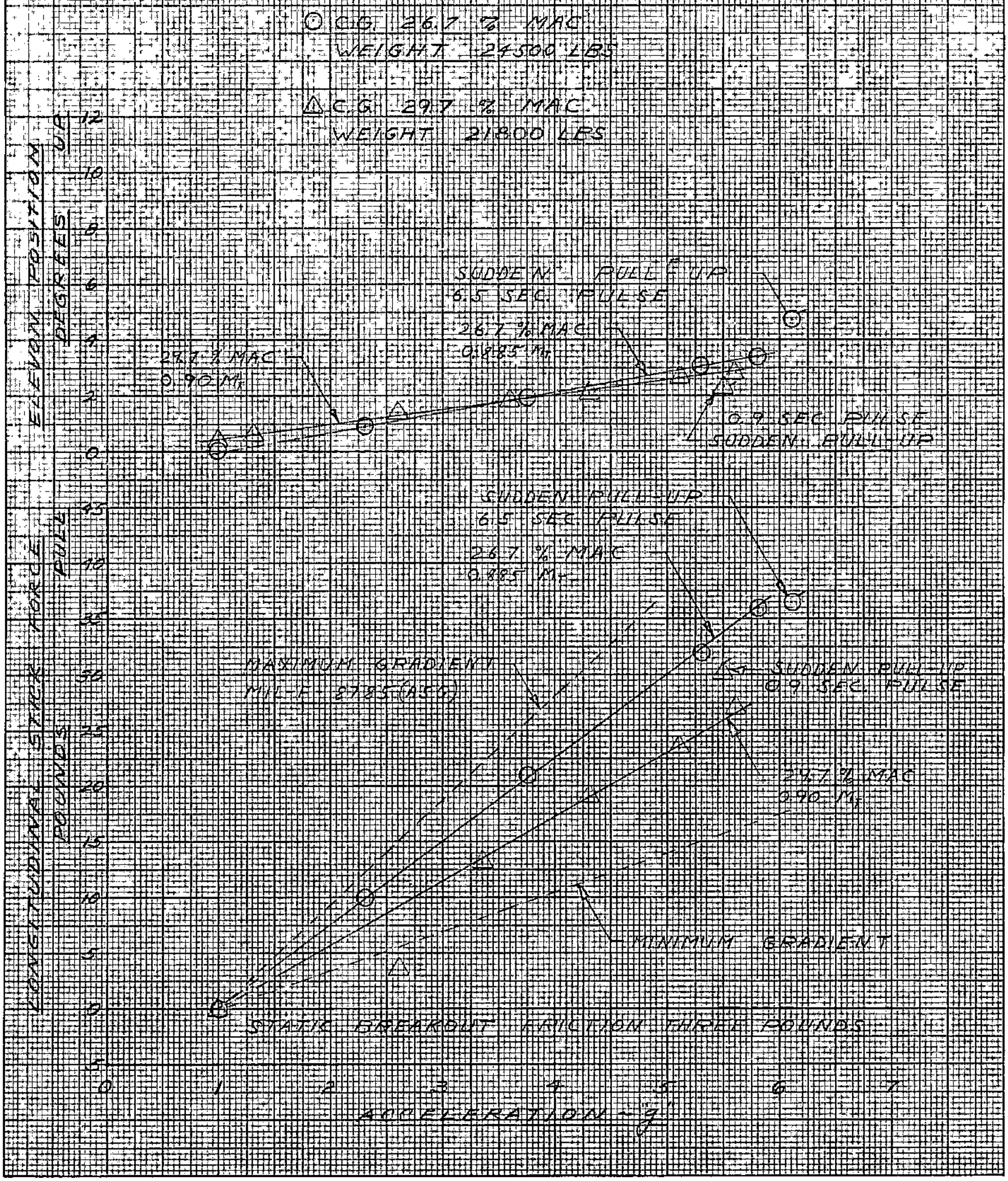


FIGURE NO. 34

MANEUVERING FLIGHT

F-102A USAF NO. 53-1796

10000 FT. ALTITUDE

CRUISE CONFIGURATION

0.85 MACH NUMBER AVG

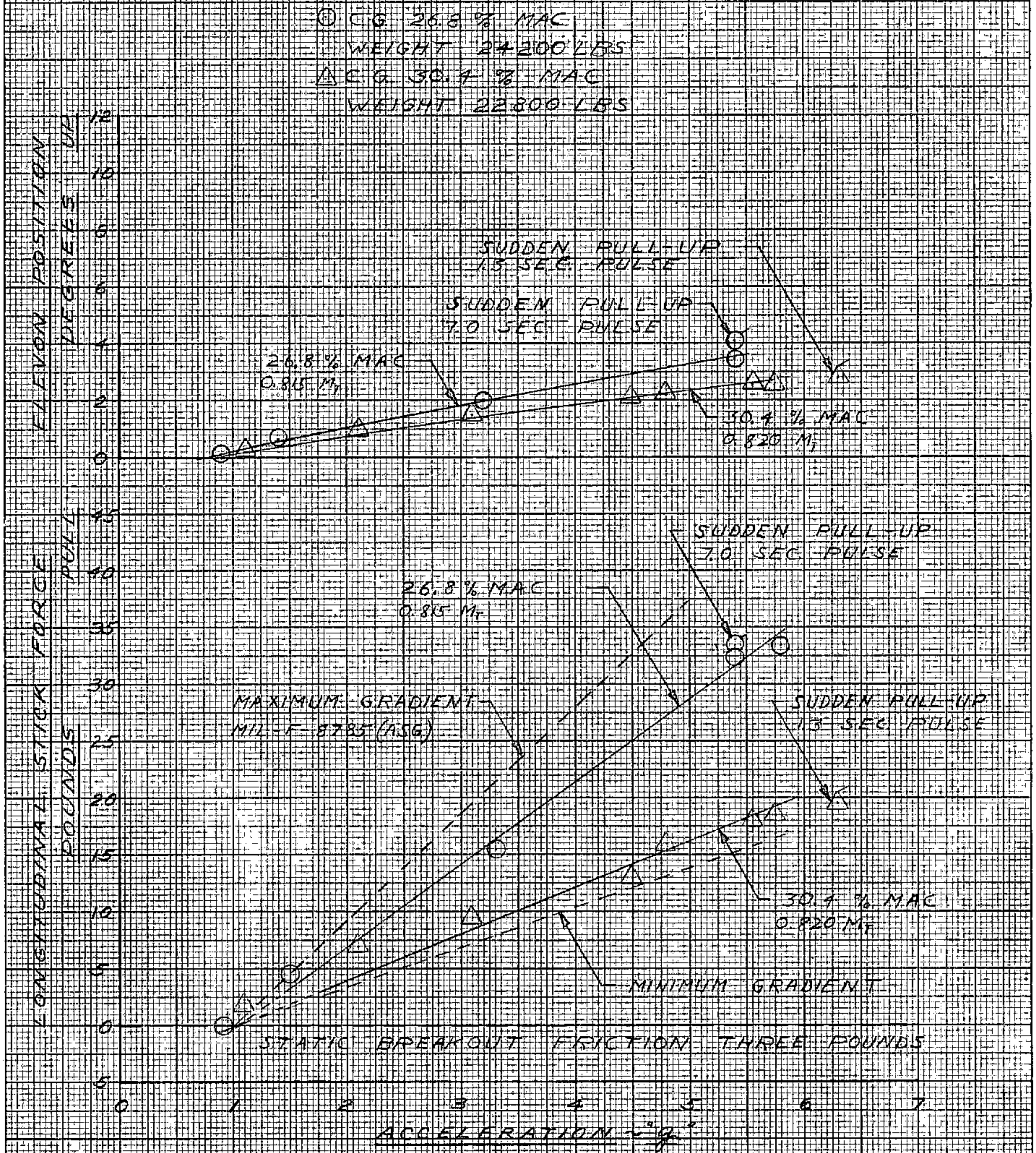


FIGURE NO. 35

MANEUVERING FLIGHT

F-102A USAF NO. 53-1796

10000 FT ALTITUDE

CRUISE CONFIGURATION

0.60 MACH NUMBER AVG

① C.G 26.95 % MAC
WEIGHT 24000 LBS

△ C.G 29.1 % MAC
WEIGHT 21400 LBS

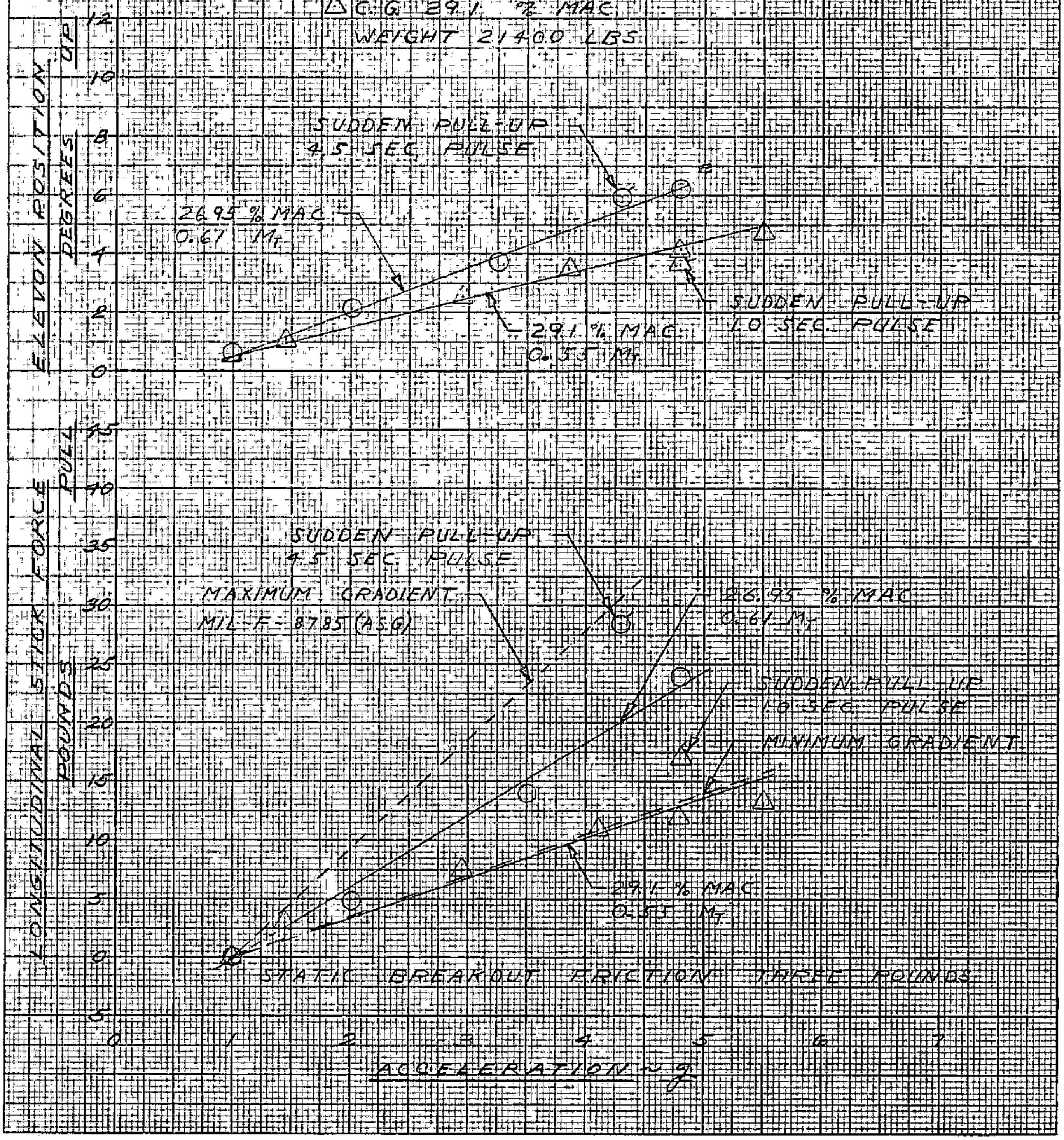


FIGURE NO. 36

MANEUVERING FLIGHT

F-102A USAF NO 53-1796
20000 FT. ALTITUDE
POWER APPROACH CONFIGURATION
180 KNOTS CALIBRATED AIRSPEED

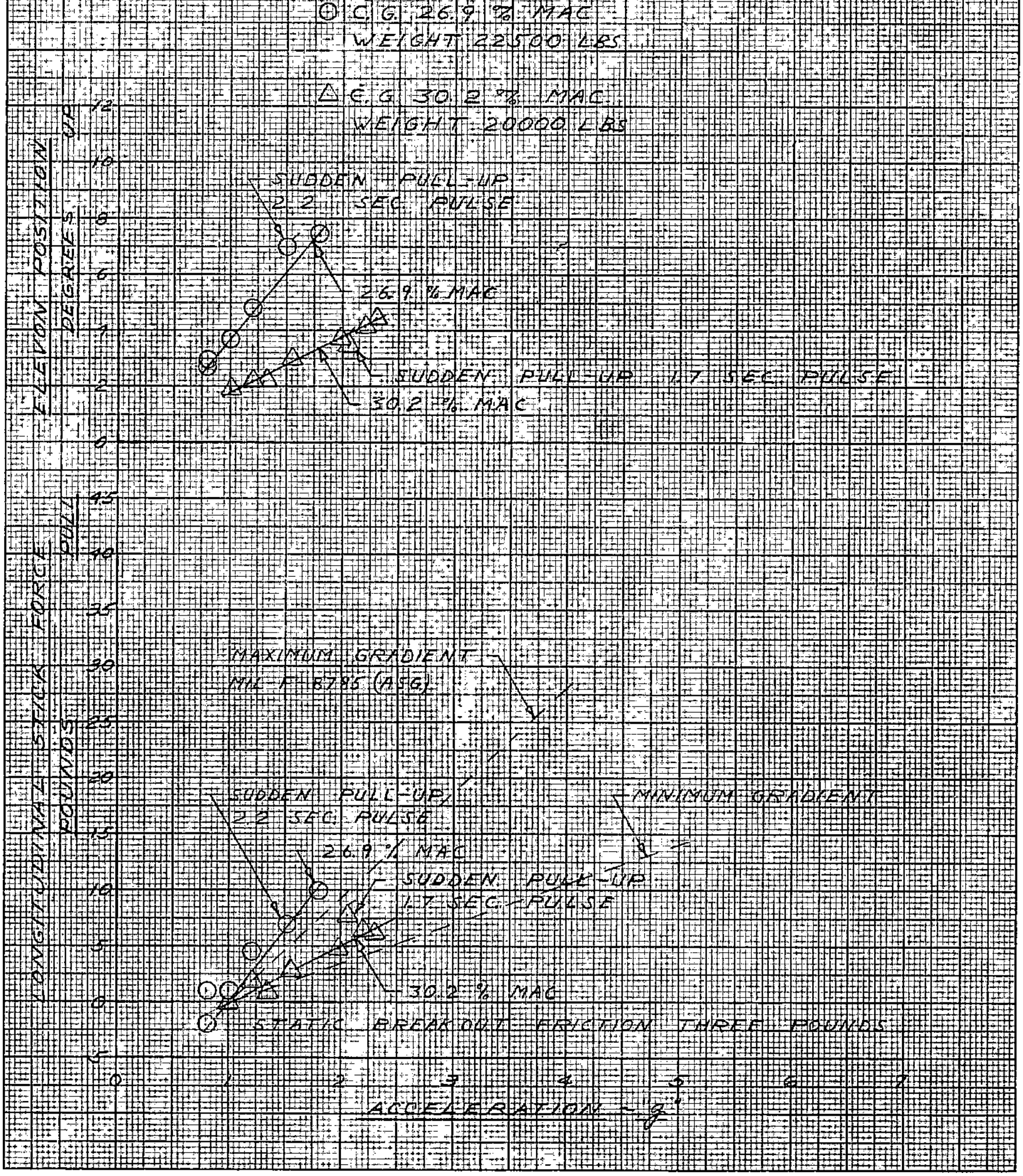


FIGURE NO 37

MANEUVERING FLIGHT

F-102A USAF NO 53-1776

30000 FEET ALTITUDE

DECELERATING TURN

CRUISE CONFIGURATION

TRIM SERVO OFF

C.G. 26.5 % MAC WEIGHT 25000 LBS

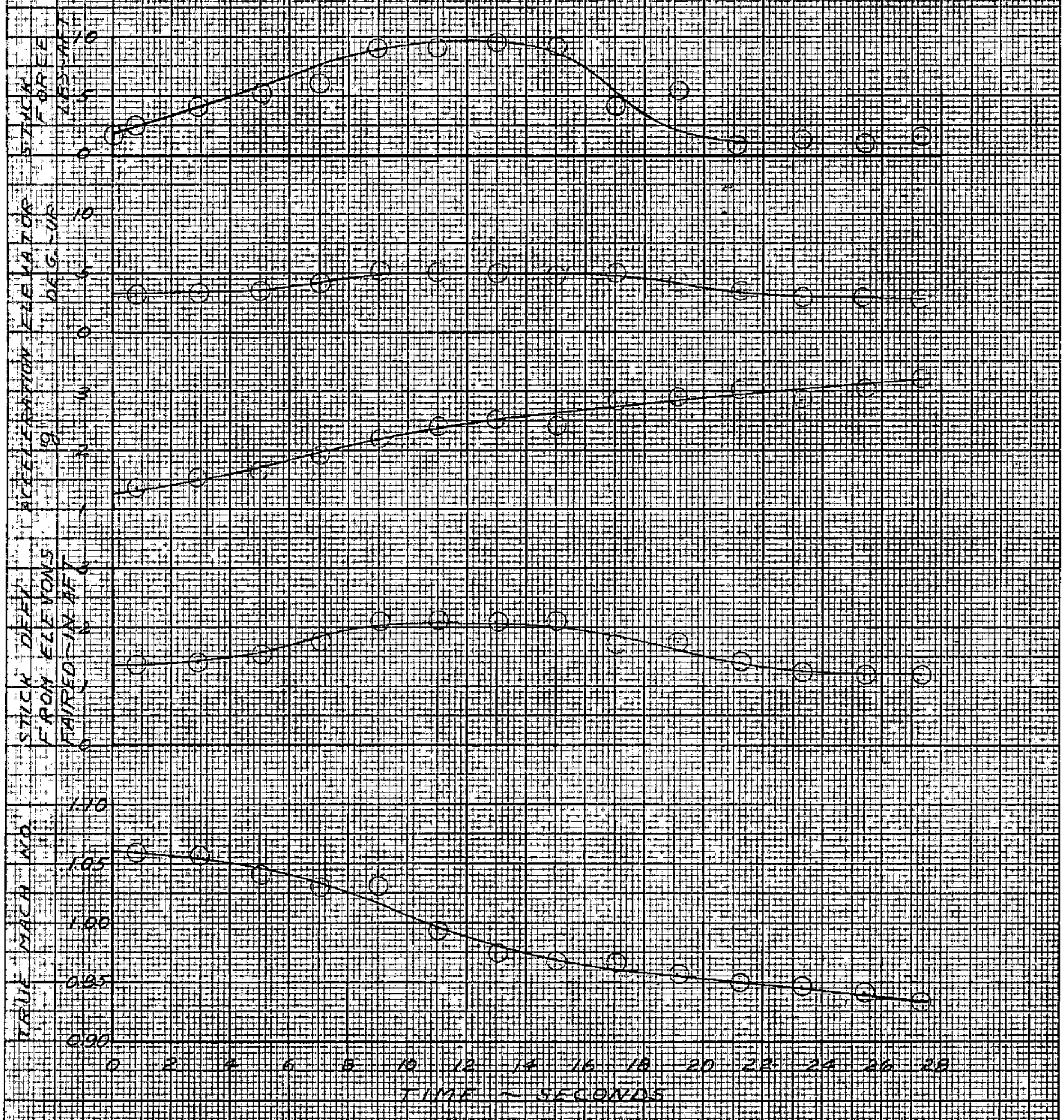


FIGURE NO. 38
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
40000 FEET ALTITUDE
DECELERATING TURN
CRUISE CONFIGURATION
TRIM SERVO ON

C.G. 26.7 % MAC WEIGHT 24500 LBS

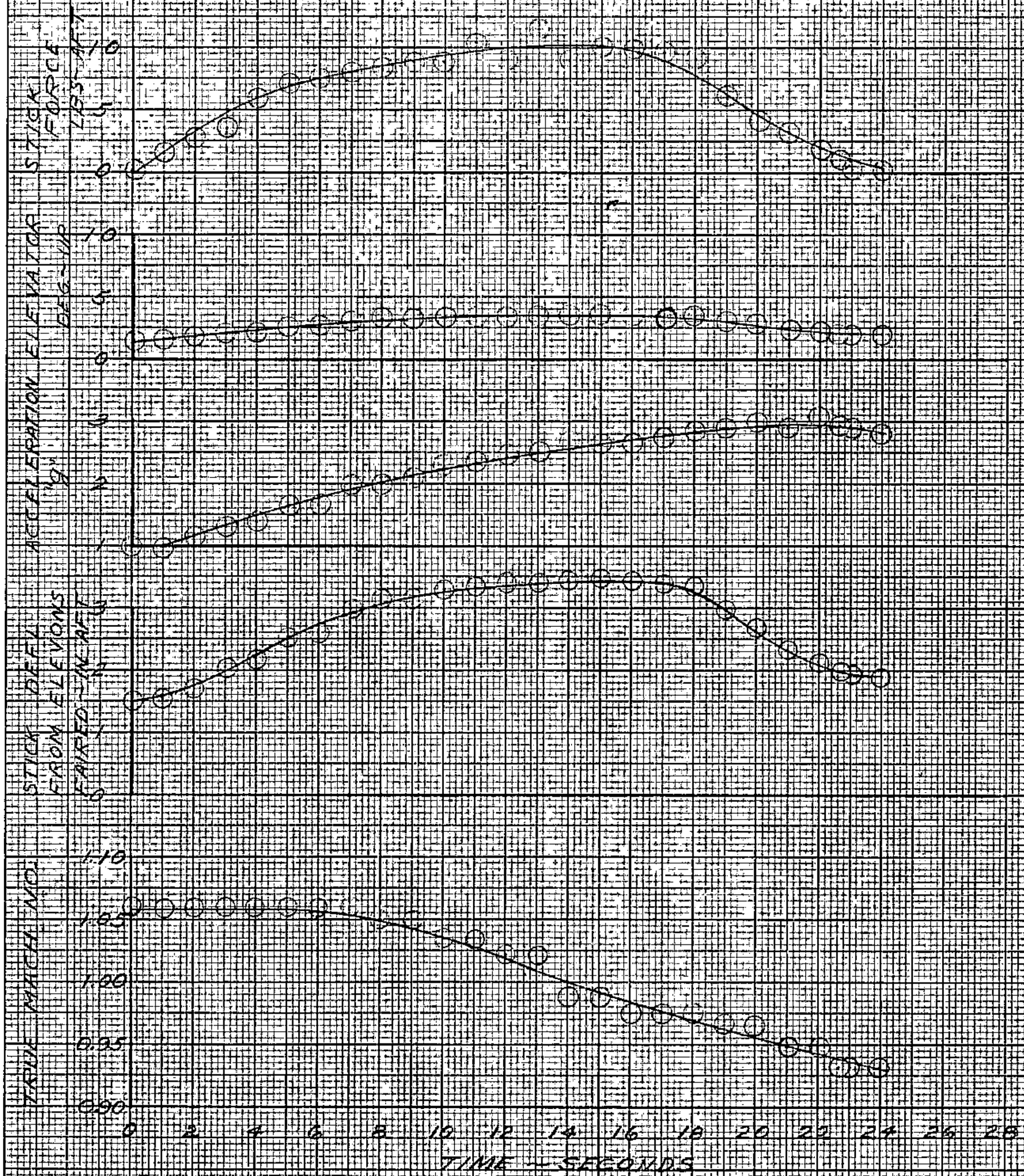
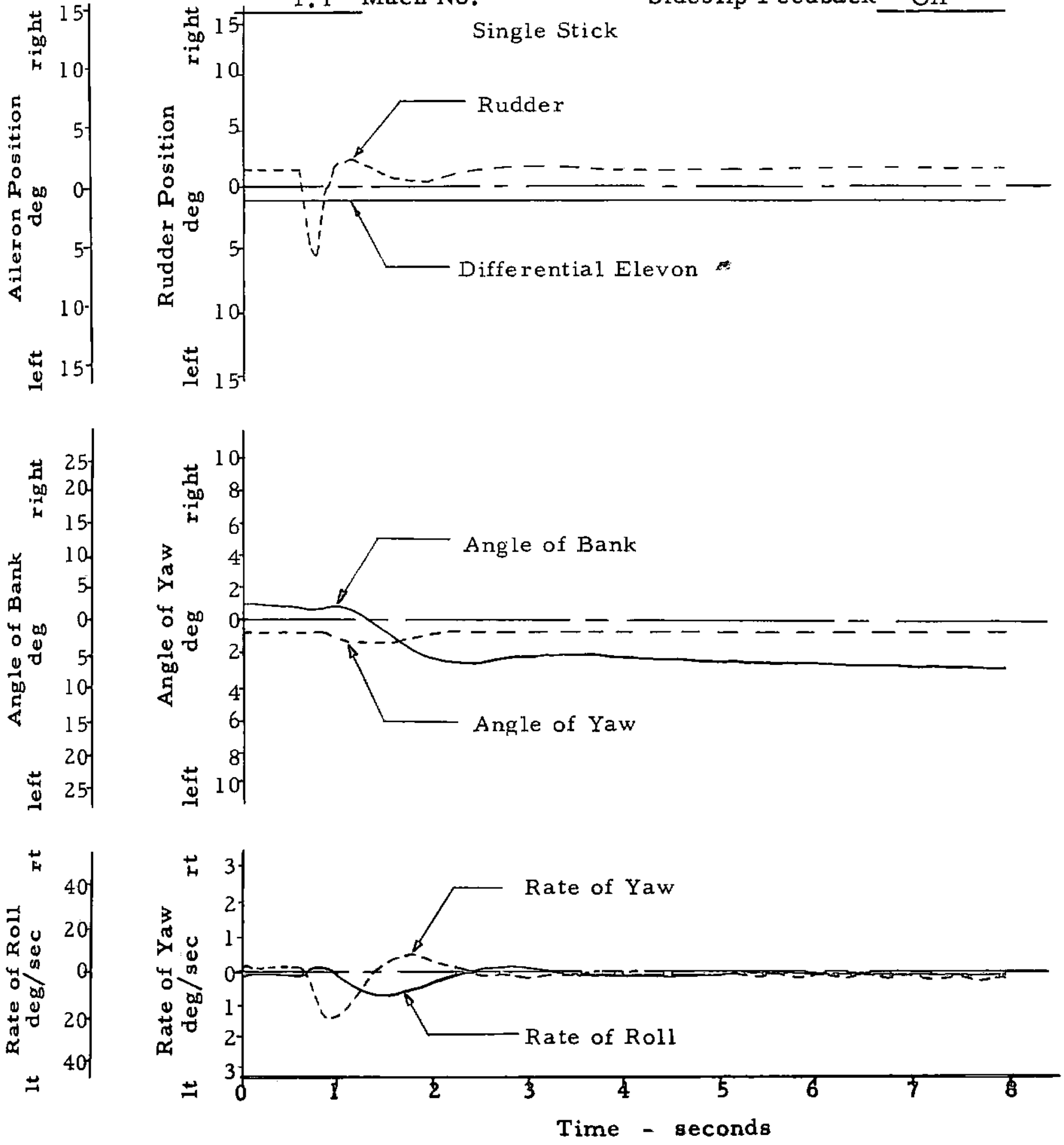


Figure No. 39

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
45,000 Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off



APPENDIX I

Figure No. 40

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
45,000 Altitude
0.77 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

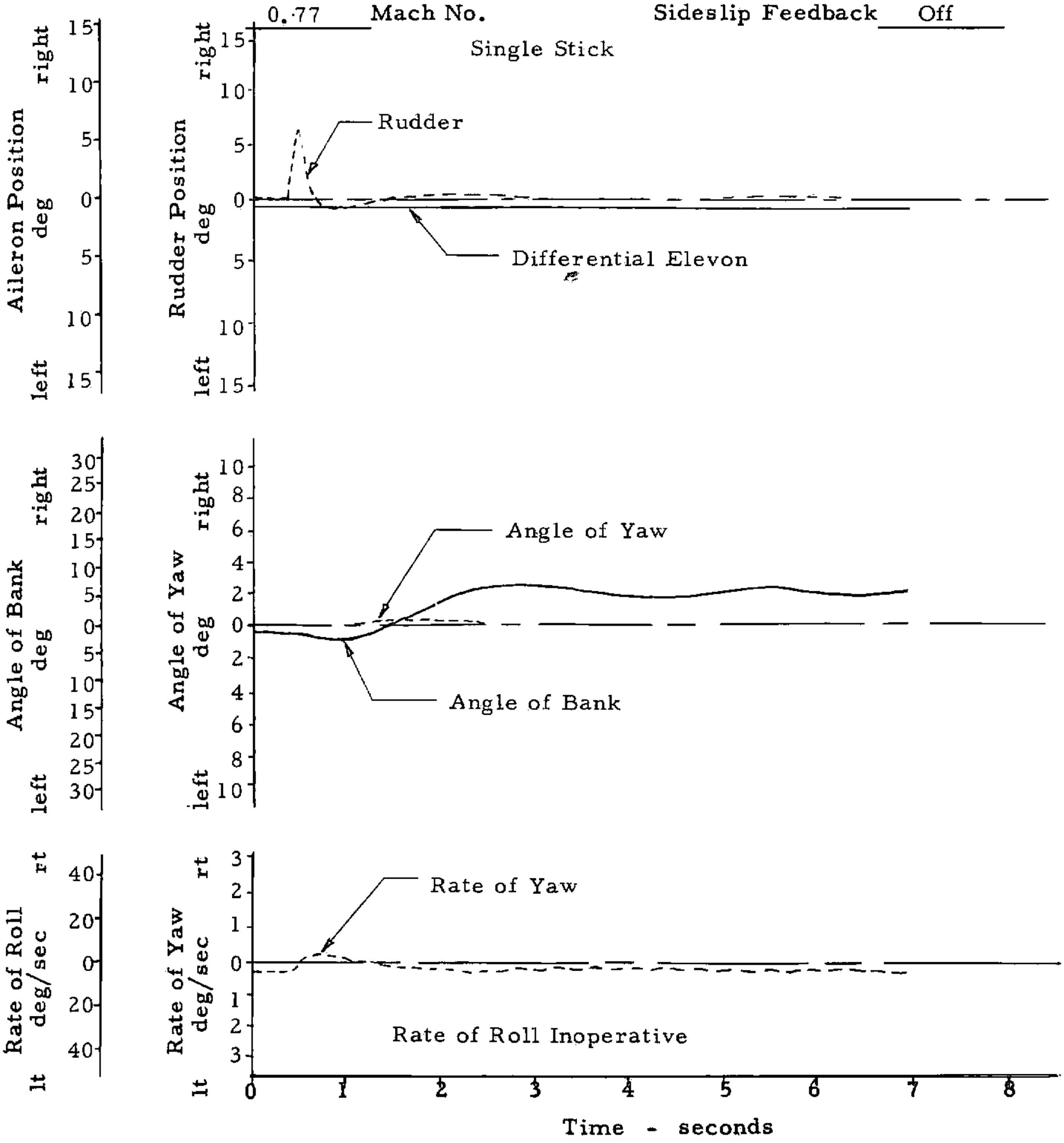


Figure No. 41

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

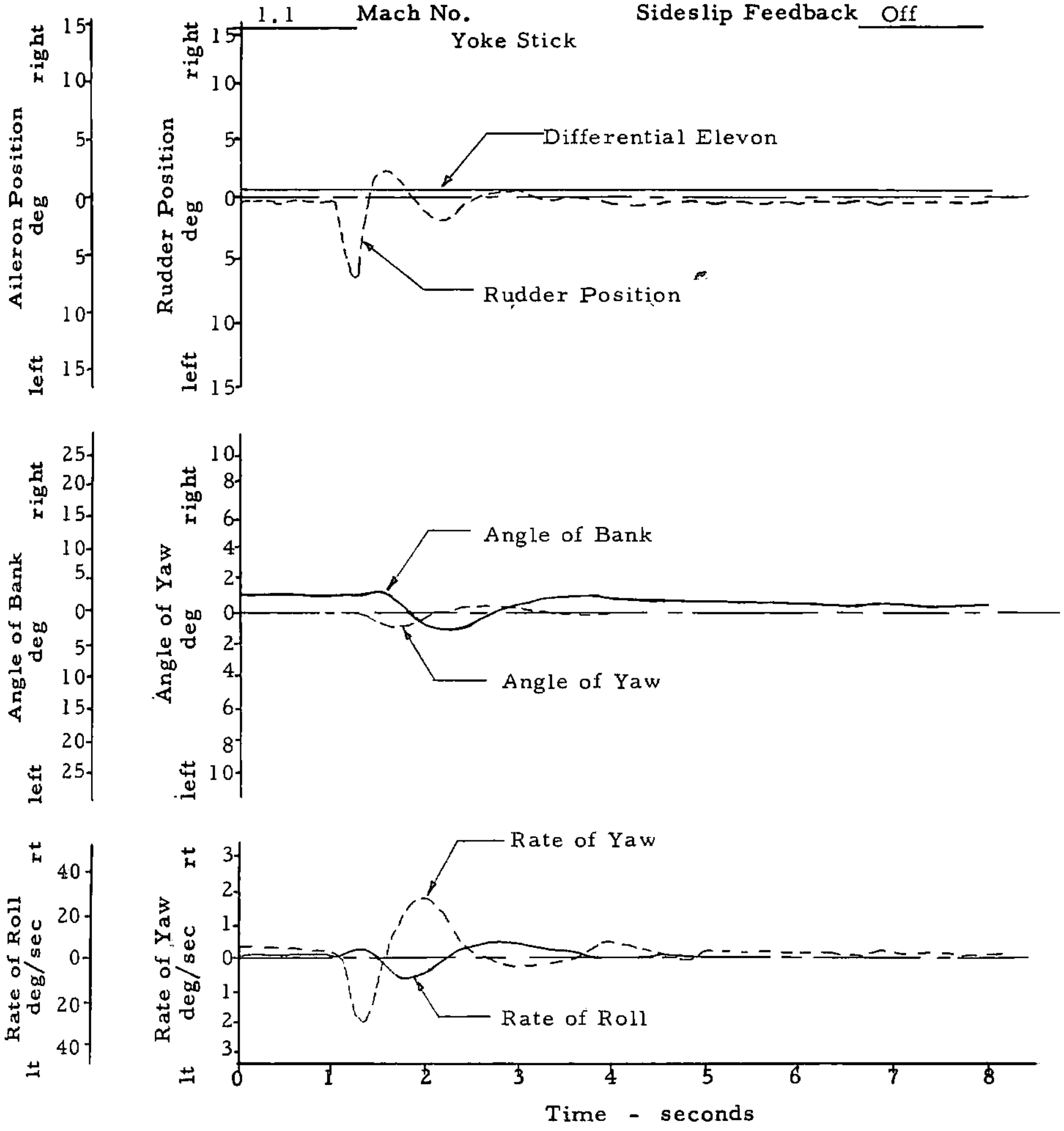


Figure No. 42

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.
Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

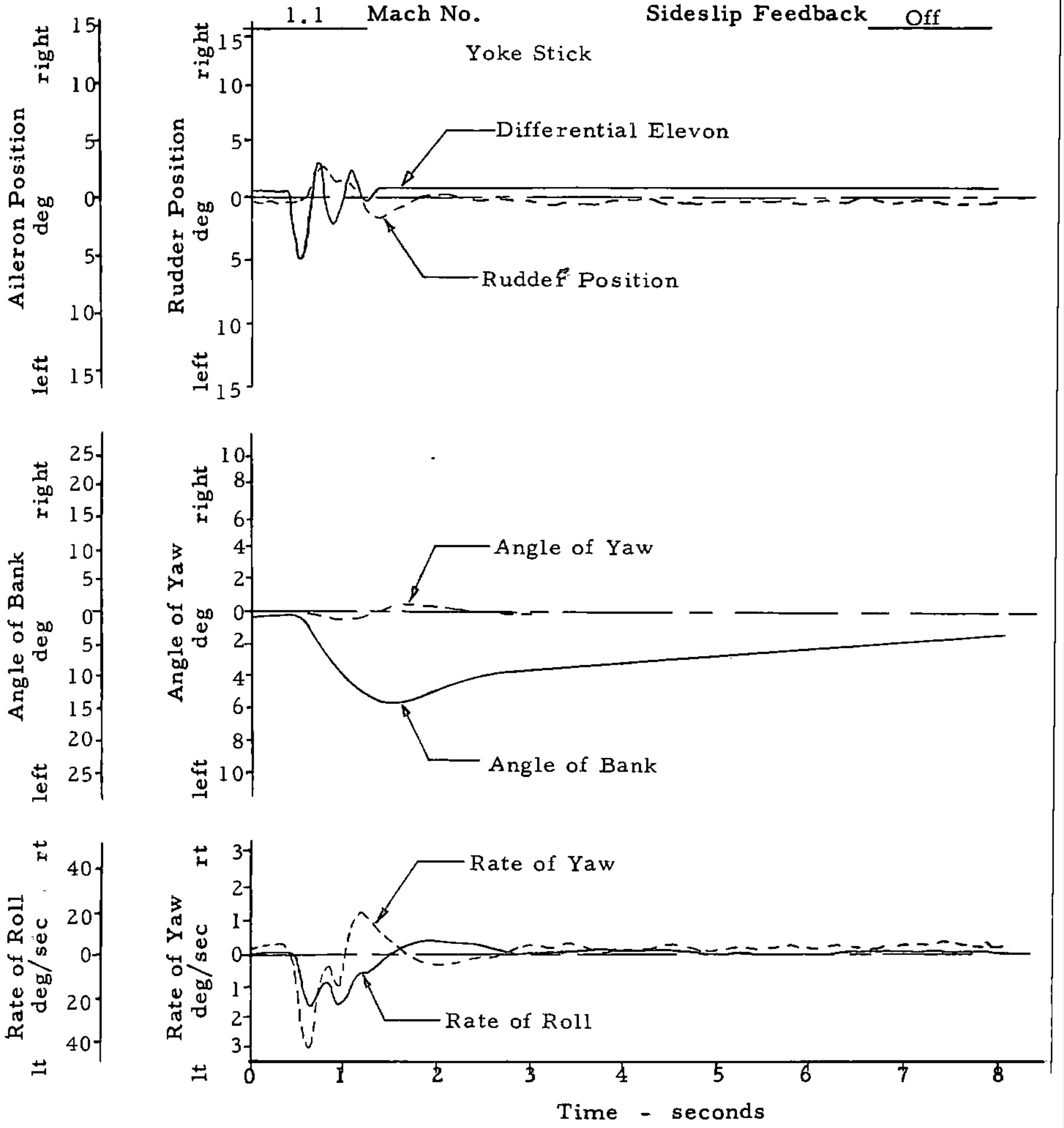


Figure No. 43

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper Off
Turn Coordinator Off
Sideslip Feedback Off

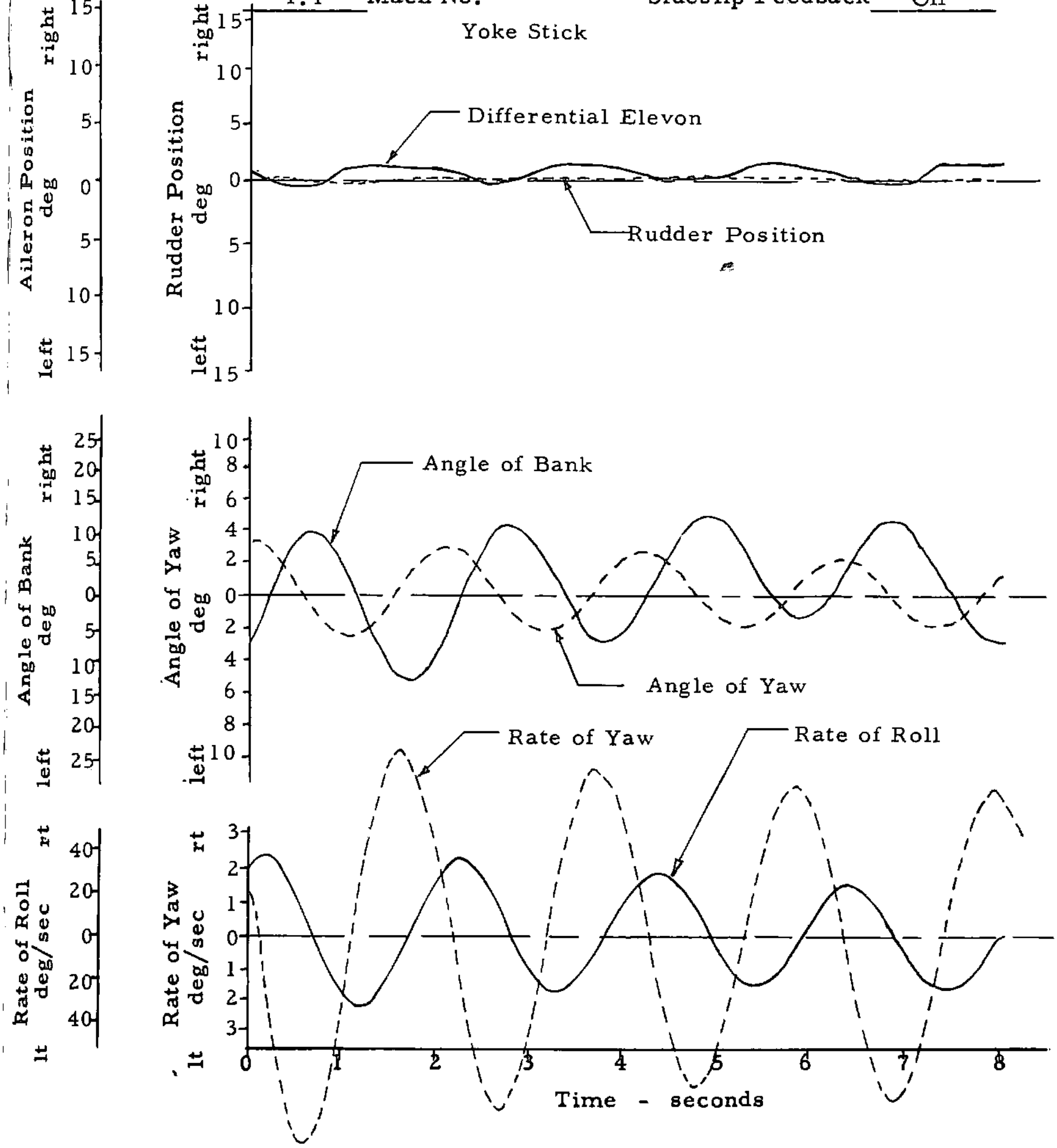


Figure No. 44

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator On
Sideslip Feedback Off

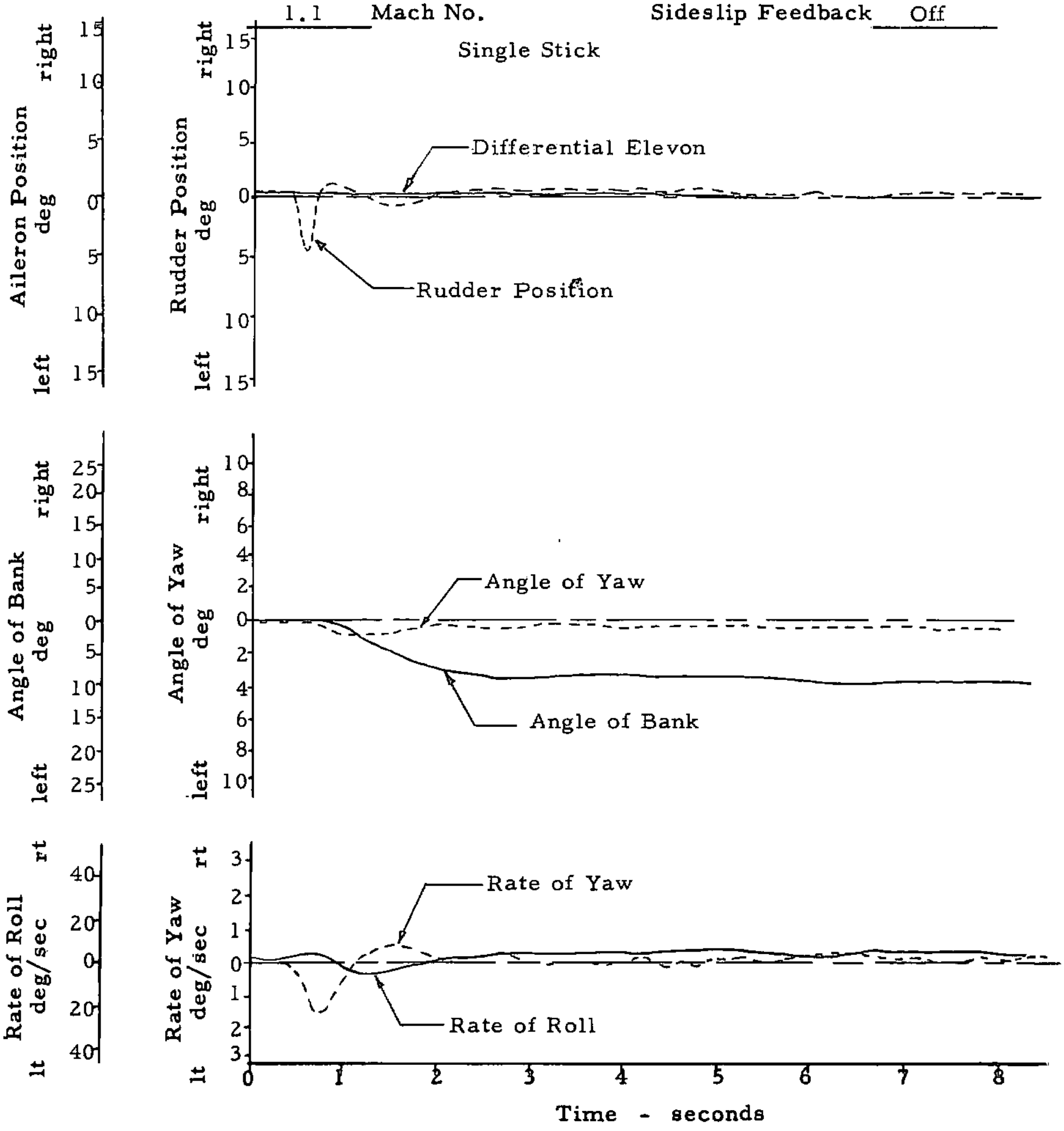


Figure No. 45

DYNAMIC LATERAL DIRECTIONAL STABILITY

F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator On
Sideslip Feedback Off

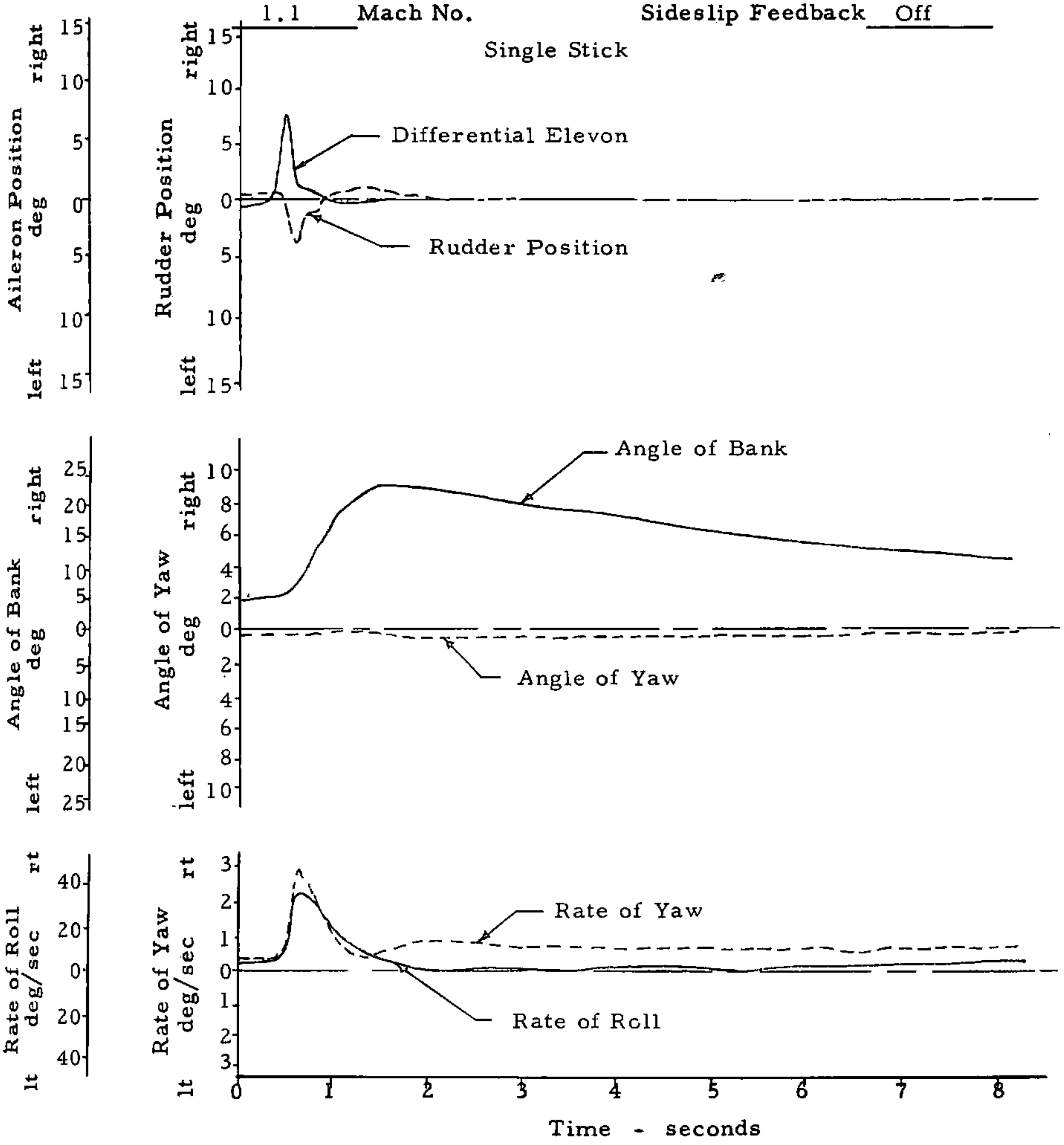


Figure No. 46

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper Off
Turn Coordinator Off
Sideslip Feedback Off

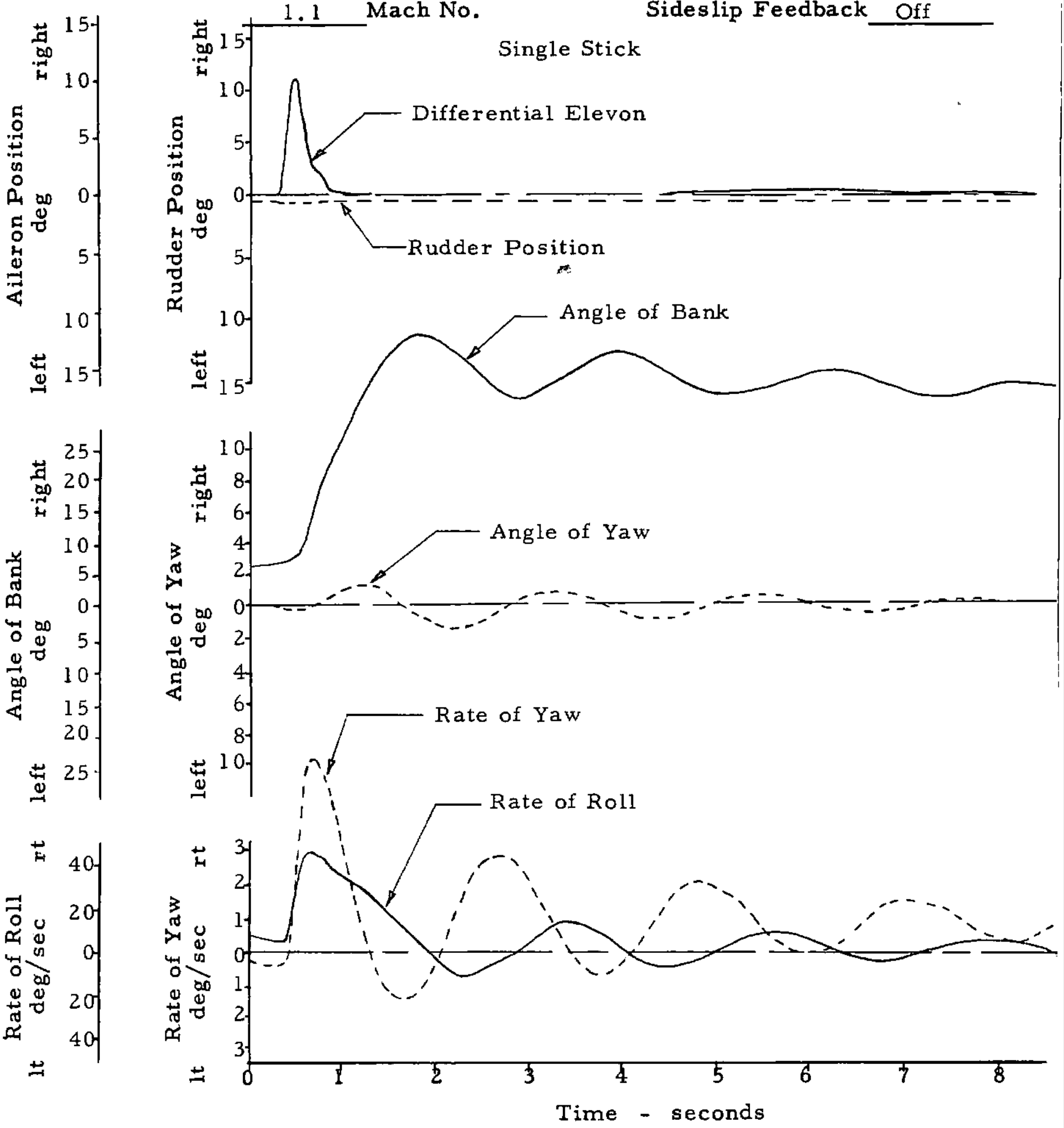


Figure No. 47

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35, 000 Altitude
0. 9 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

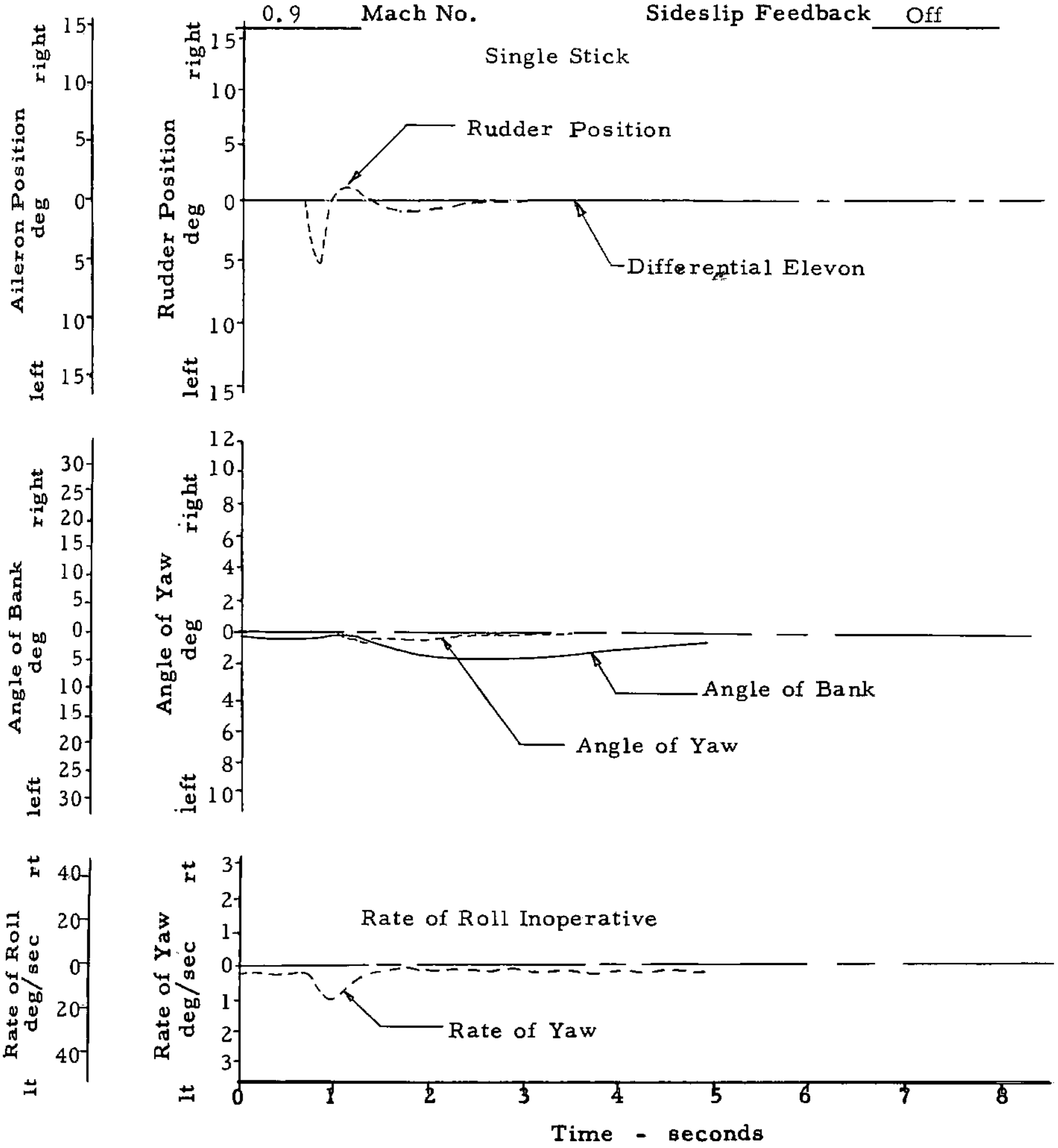


Figure No. 48

DYNAMIC LATERAL DIRECTIONAL STABILITY

F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
0.7 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

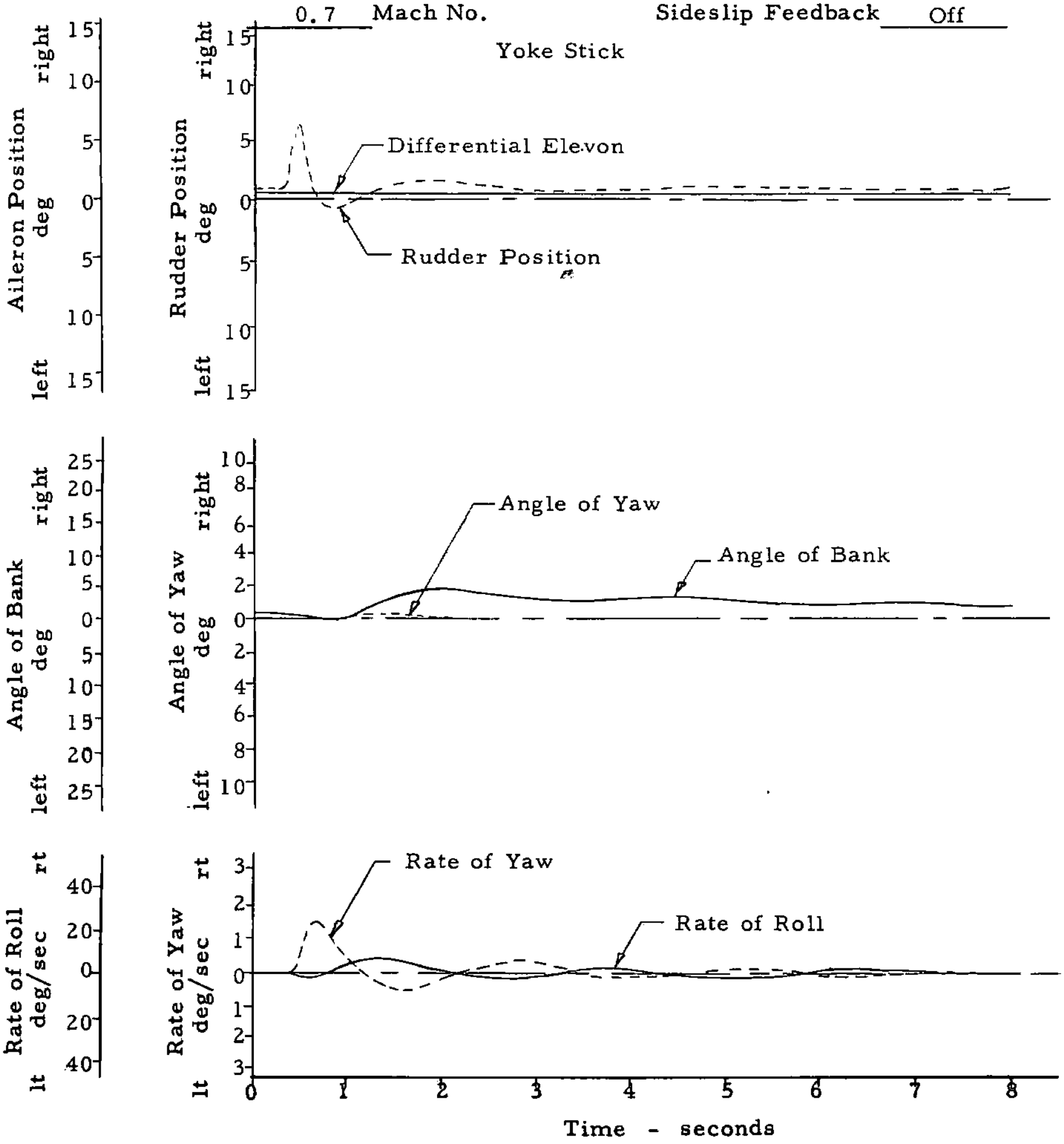


Figure No. 49

DYNAMIC LATERAL DIRECTIONAL STABILITY

F-102A USAF No. 53-1796

Cruise Configuration
35, 000 Altitude
0. 7 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

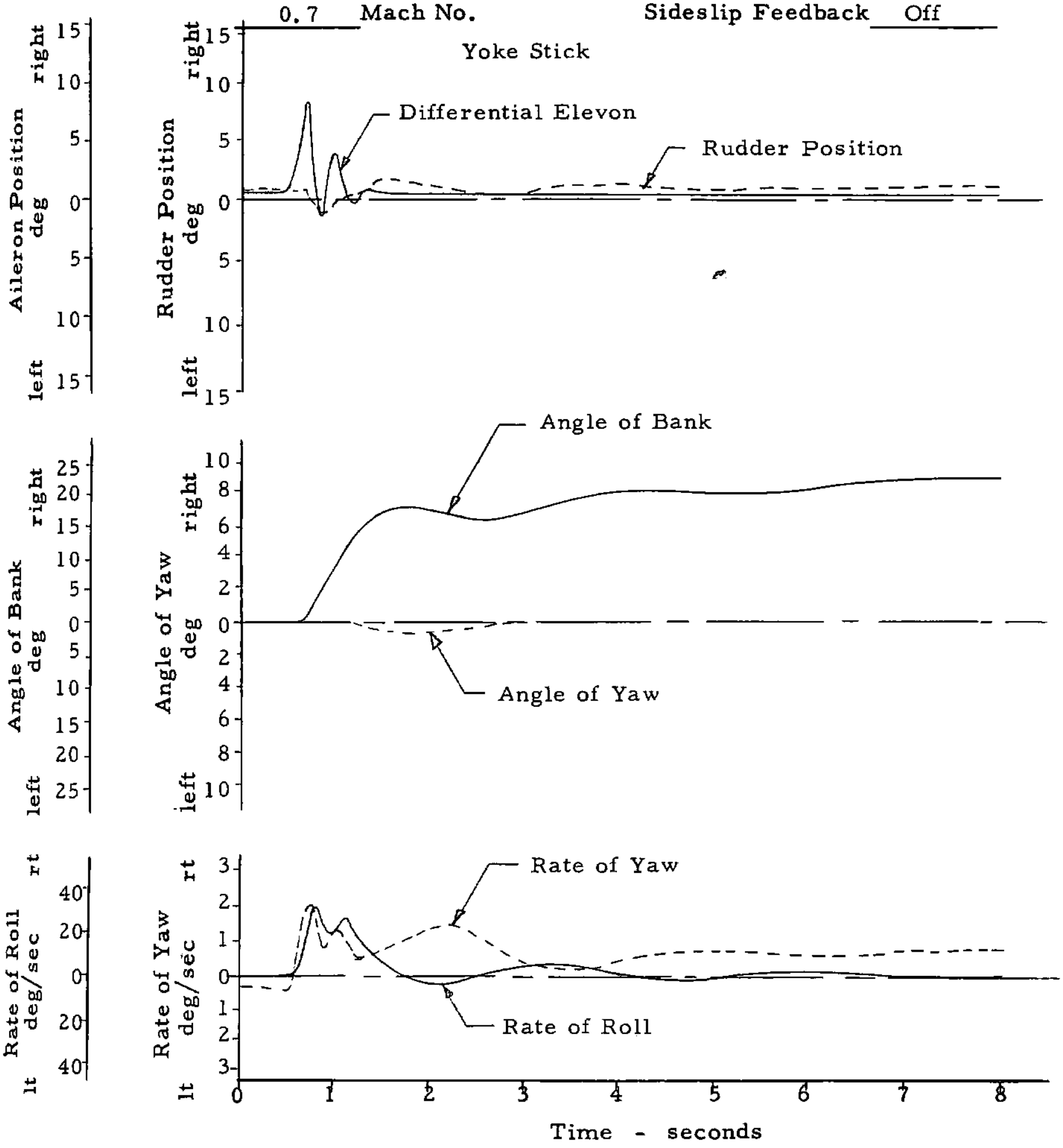


Figure No. 50

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
0.7 Mach No.

Yaw Damper Off
Turn Coordinator Off
Sideslip Feedback Off

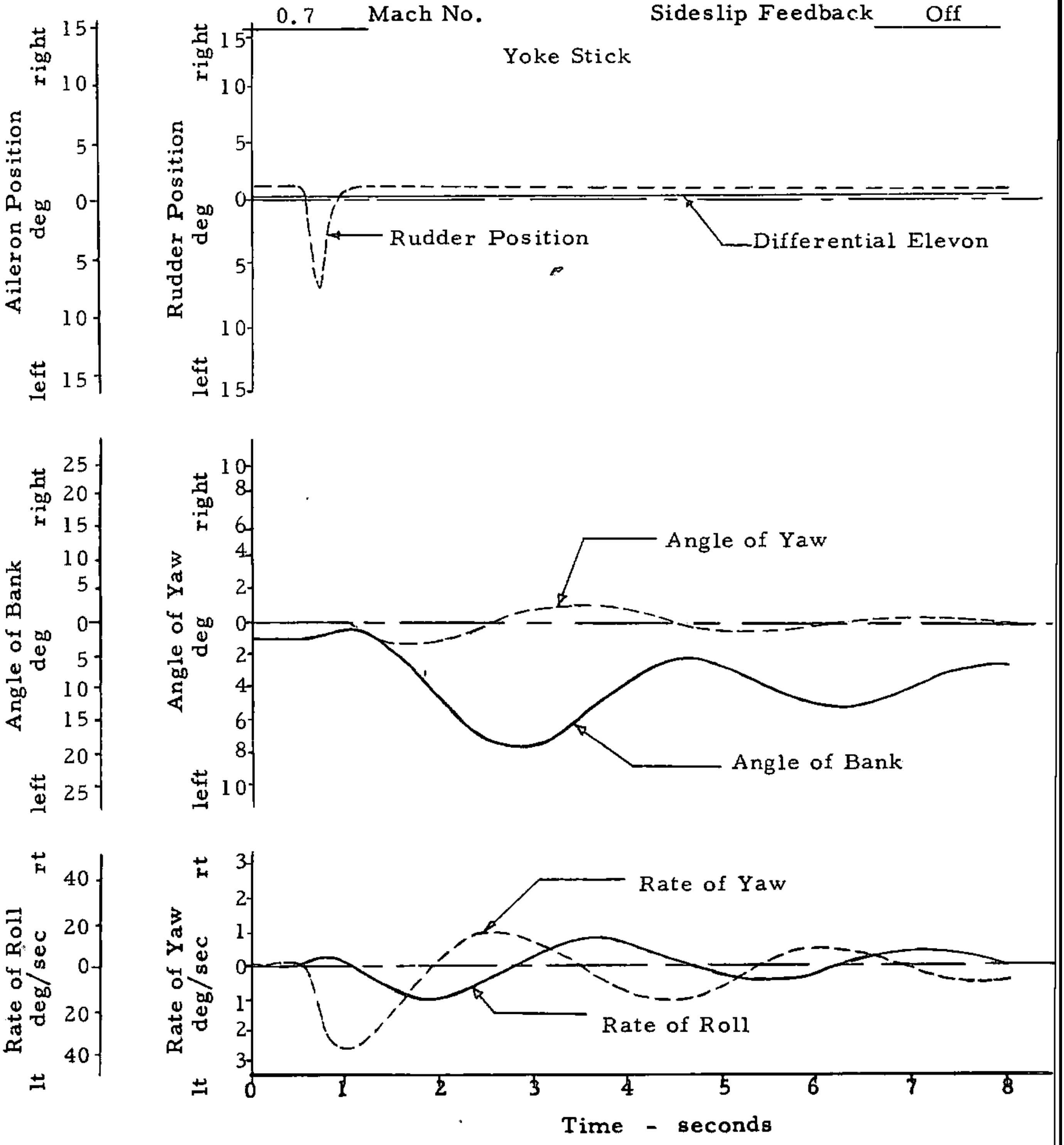


Figure No. 51

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
10,000 Altitude
0.7 Mach No.
Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

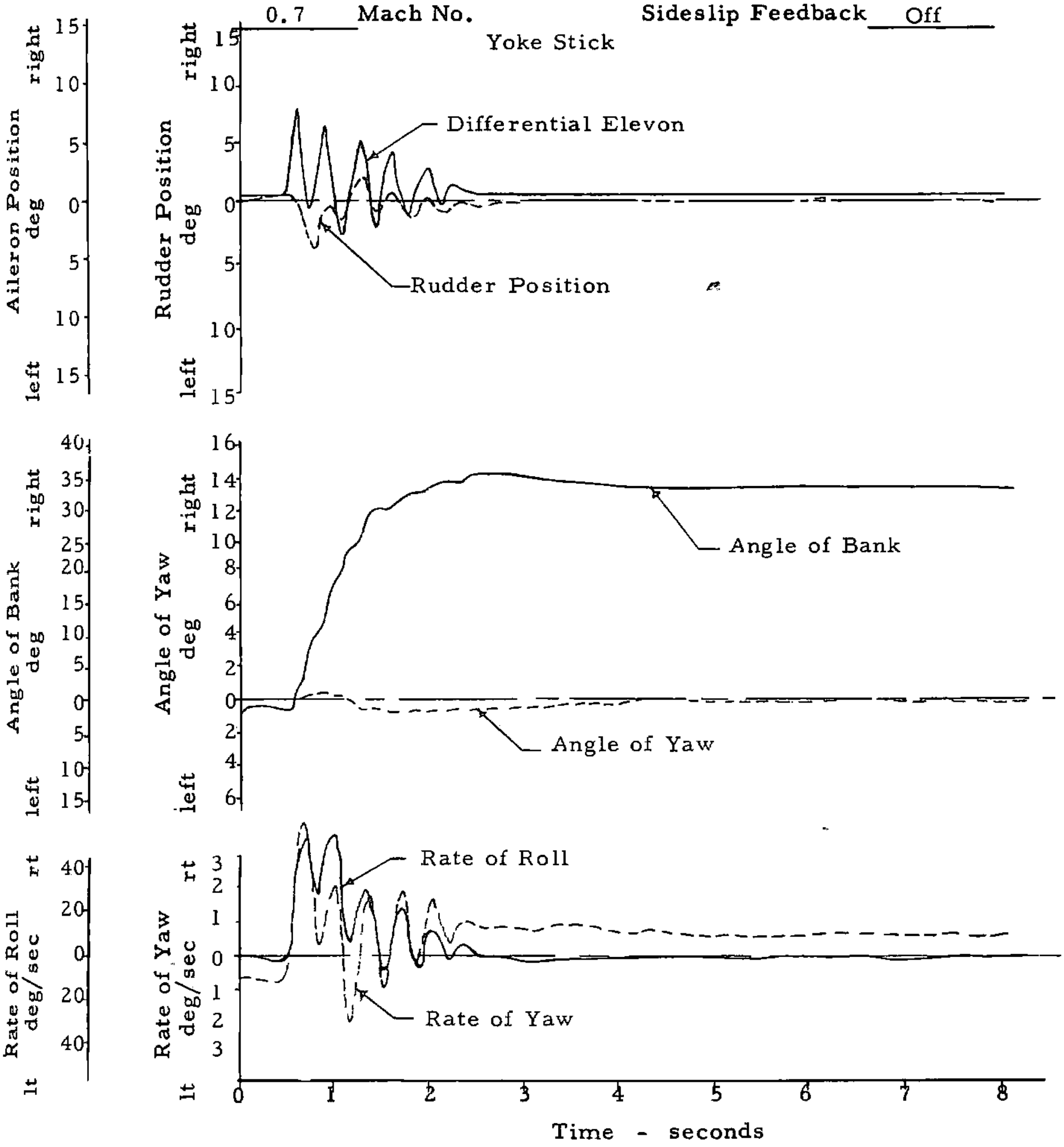


Figure No. 52

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
10,000 Altitude
0.5 Mach No.

Yaw Damper On
Turn Coordinator On
Sideslip Feedback Off

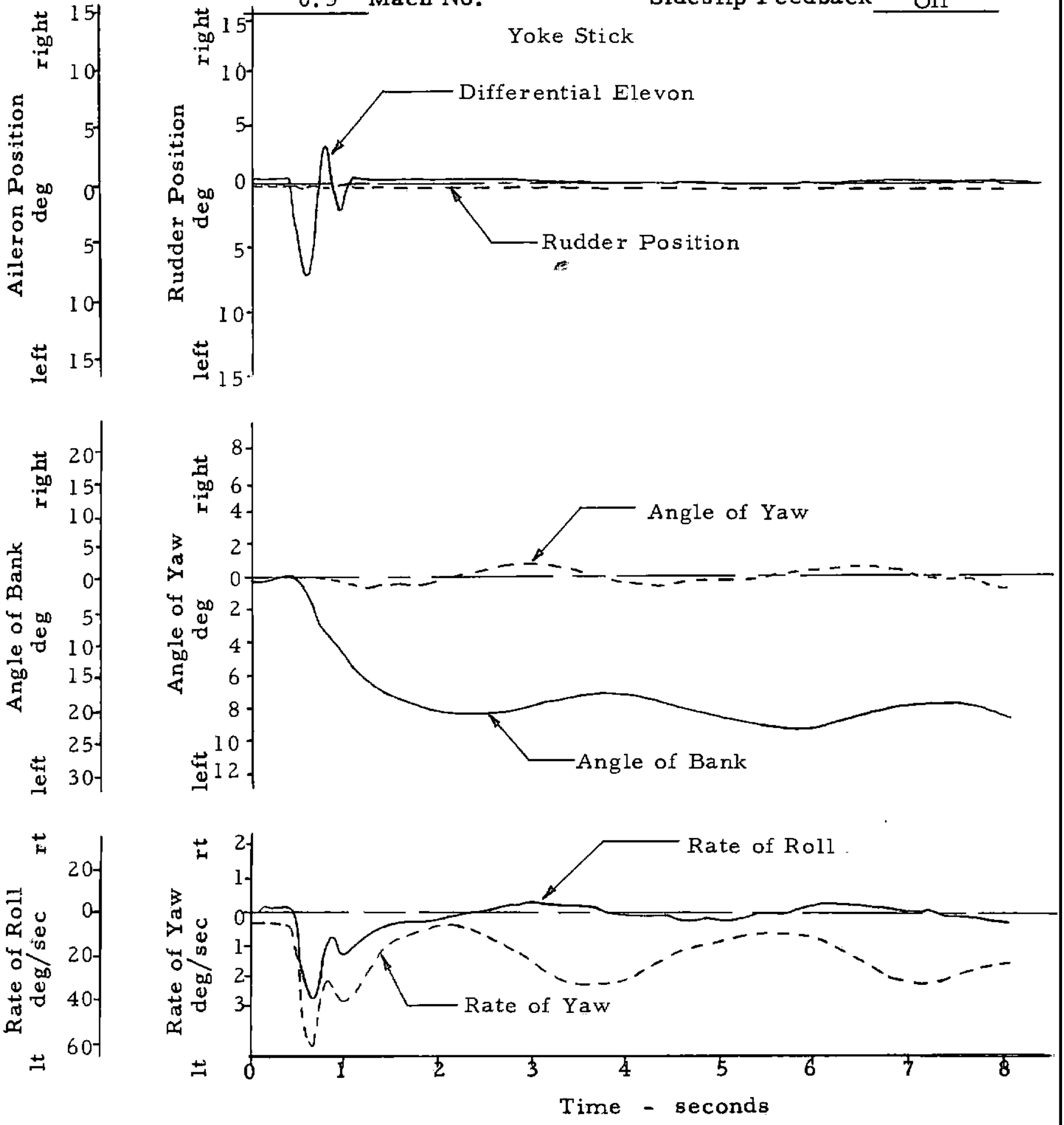


Figure No. 53

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
10,000 Altitude
0.5 Mach No.

Yaw Damper On
Turn Coordinator On
Sideslip Feedback Off

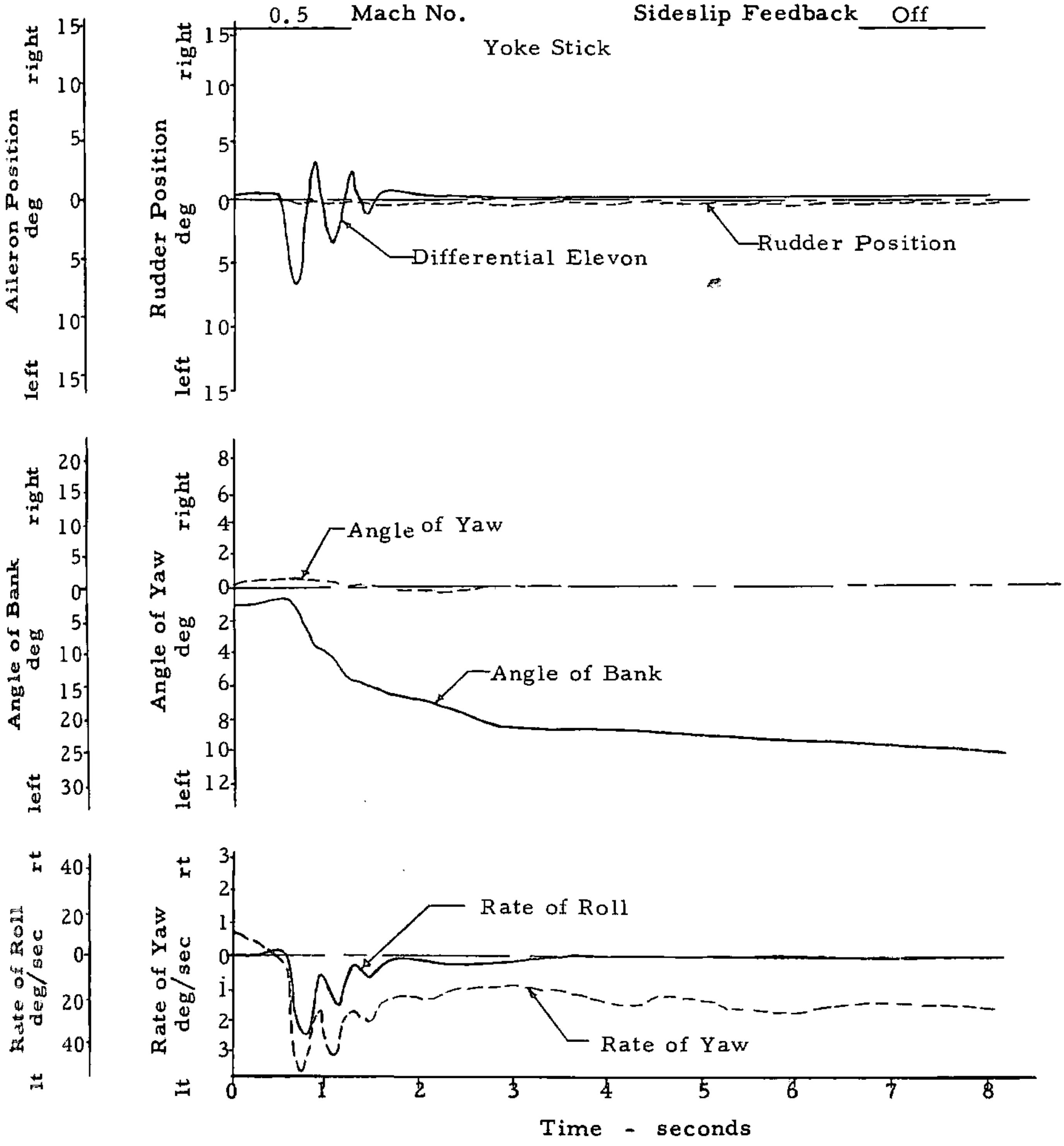


FIGURE NO. 54

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION

VARIATION IN RUDDER FORCE WITH MACH NUMBER TO MAINTAIN FOUR DEGREES OF RUDDER DEFLECTION

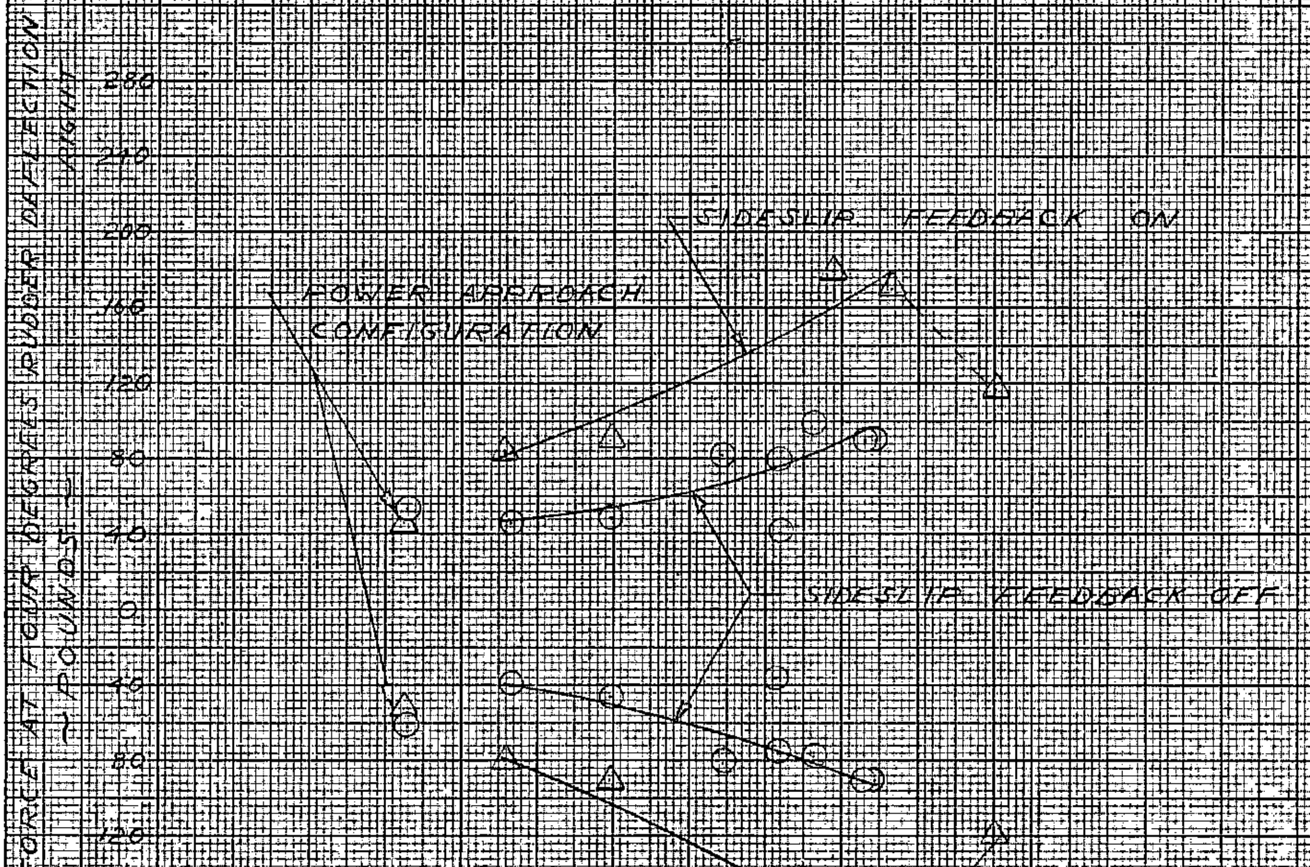


FIGURE NO. 55

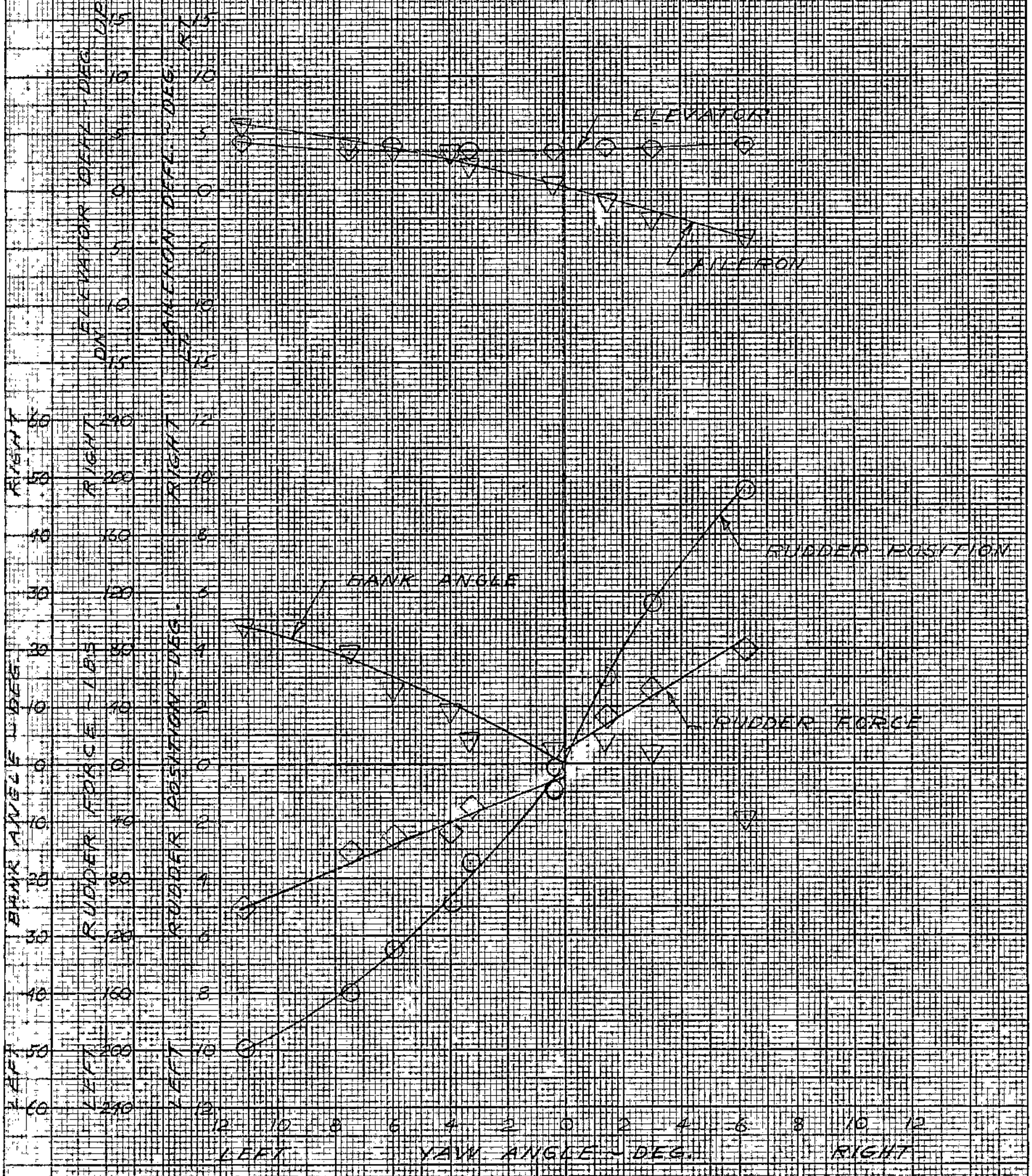
STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION β FEEDBACK OFF

45,400 FT. ALTITUDE MACH NO. 0.812

LIMIT STATIC RUDDER DEFLECTION ±25 DEG.



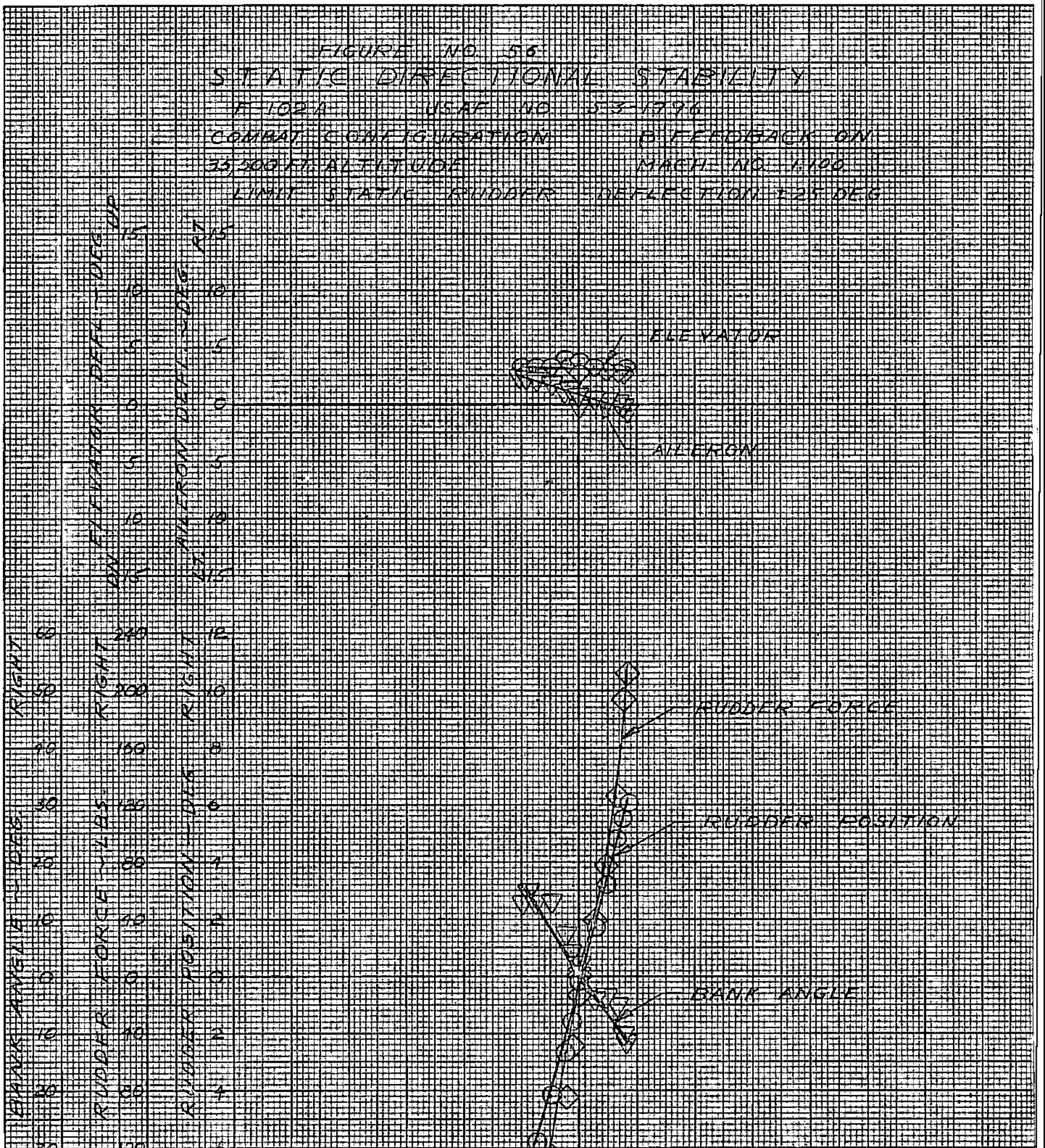


FIGURE NO. 56

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

COMBAT CONFIGURATION B FEEDBACK ON

35,300 FT. ALTITUDE MACH NO. 1.100

LIMIT STATIC RUDDER DEFLECTION ±25 DEG

FIGURE NO. 57

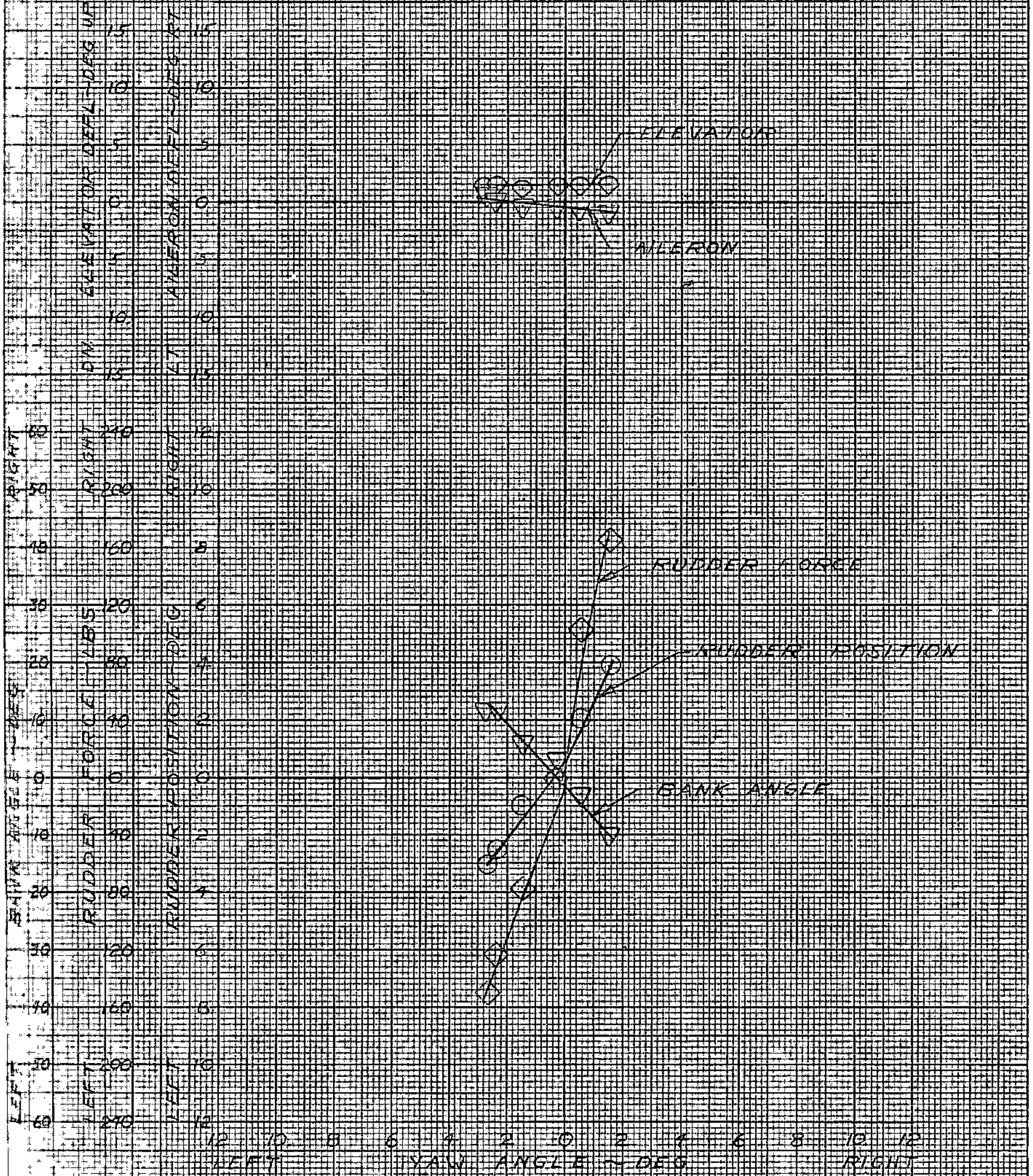
STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION B FEEDBACK ON

35,400 ALTITUDE MACH NO. 0.965

LIMIT STATIC RUDDER DEFLECTION ±25 DEG



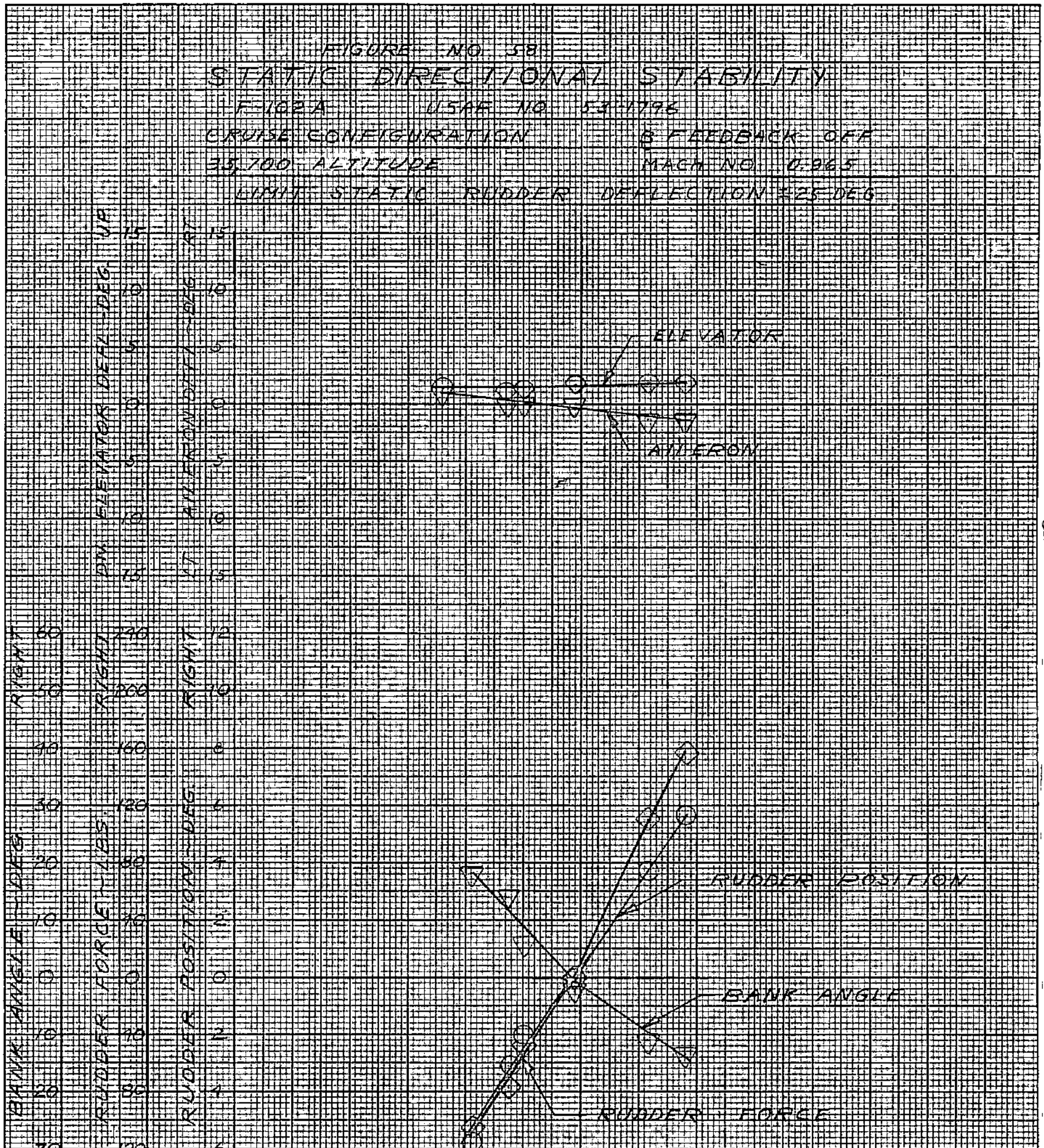


FIGURE NO. 58

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION β FEEDBACK OFF

35,700 ALTITUDE MACH NO. 0.965

LIMIT STATIC RUDDER DEFLECTION ±25 DEG

FIGURE NO. 57

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION β FEEDBACK ON

35,335 ALTITUDE MACH NO. 0.887

LIMIT STATIC RUDDER DEFLECTION ±25 DEG

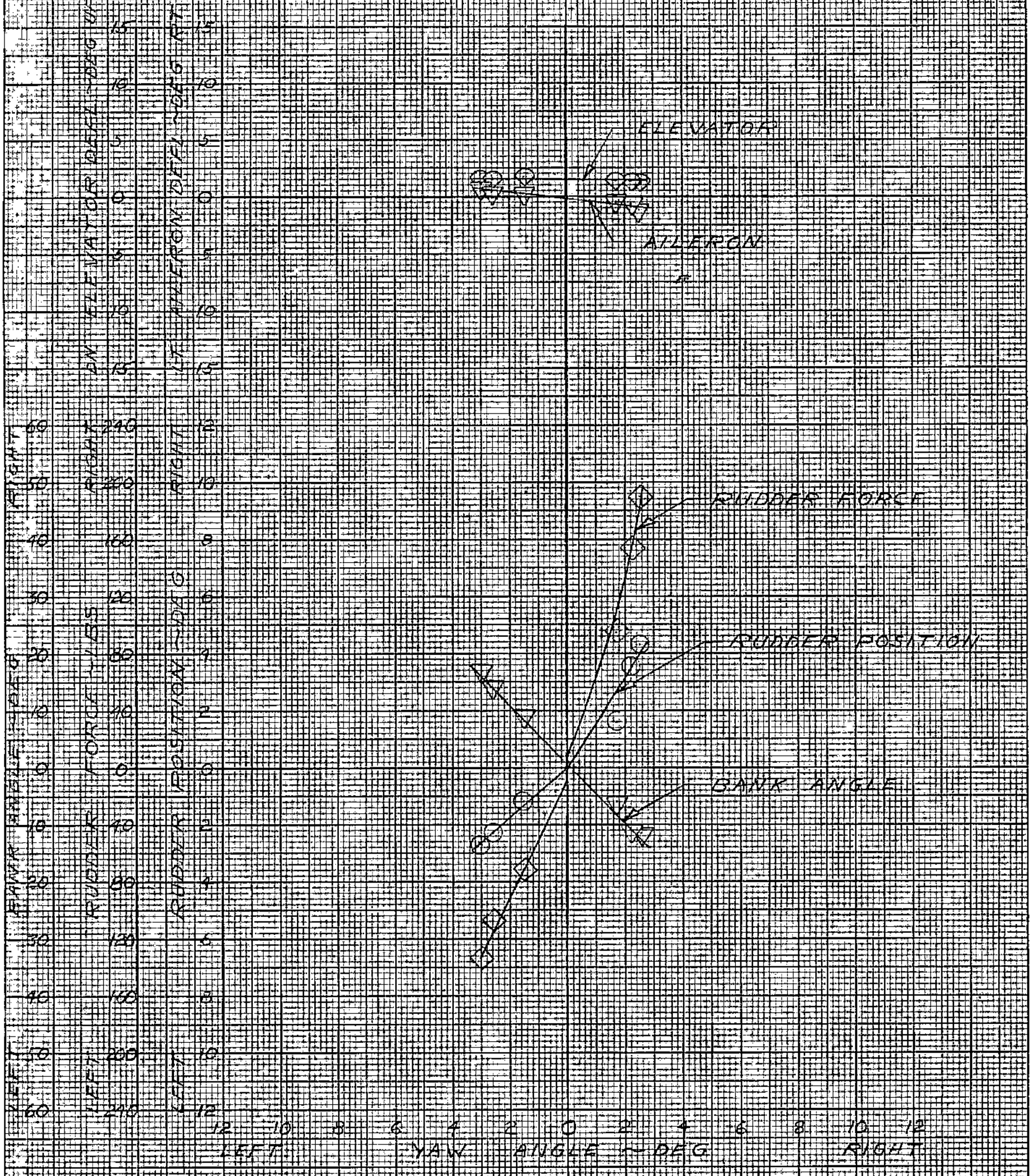


FIGURE NO. 50

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION R FEEDBACK OFF

35,280 ALTITUDE MACH NO. 0.887

LIMIT STATIC RUDDER DEFLECTION ±25 DEG

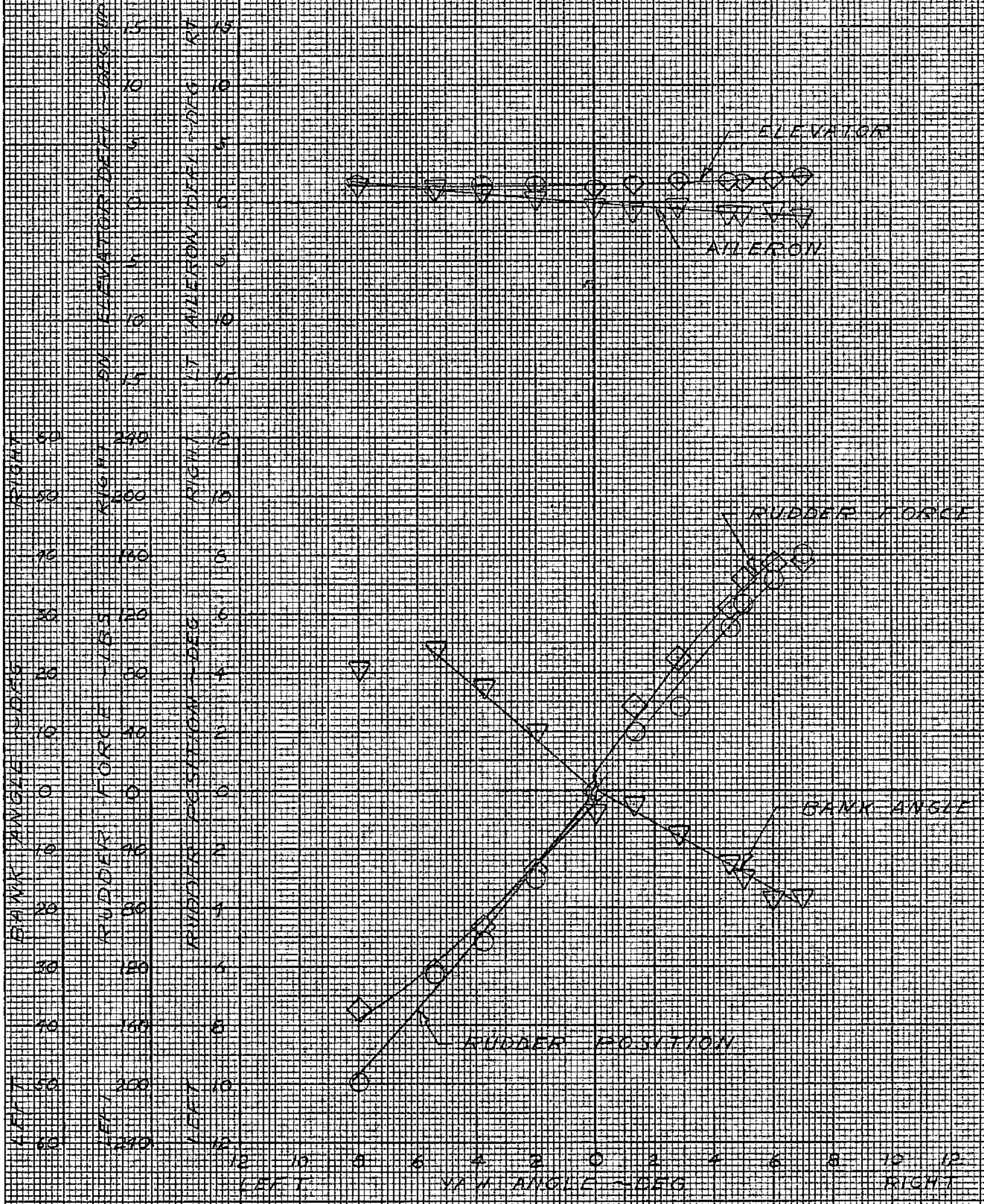


FIGURE NO. 61

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION β FEEDBACK OFF

35,000 ALTITUDE MACH NO. 0.741

LIMIT STATIC RUDDER DEFLECTION ± 25 DEG

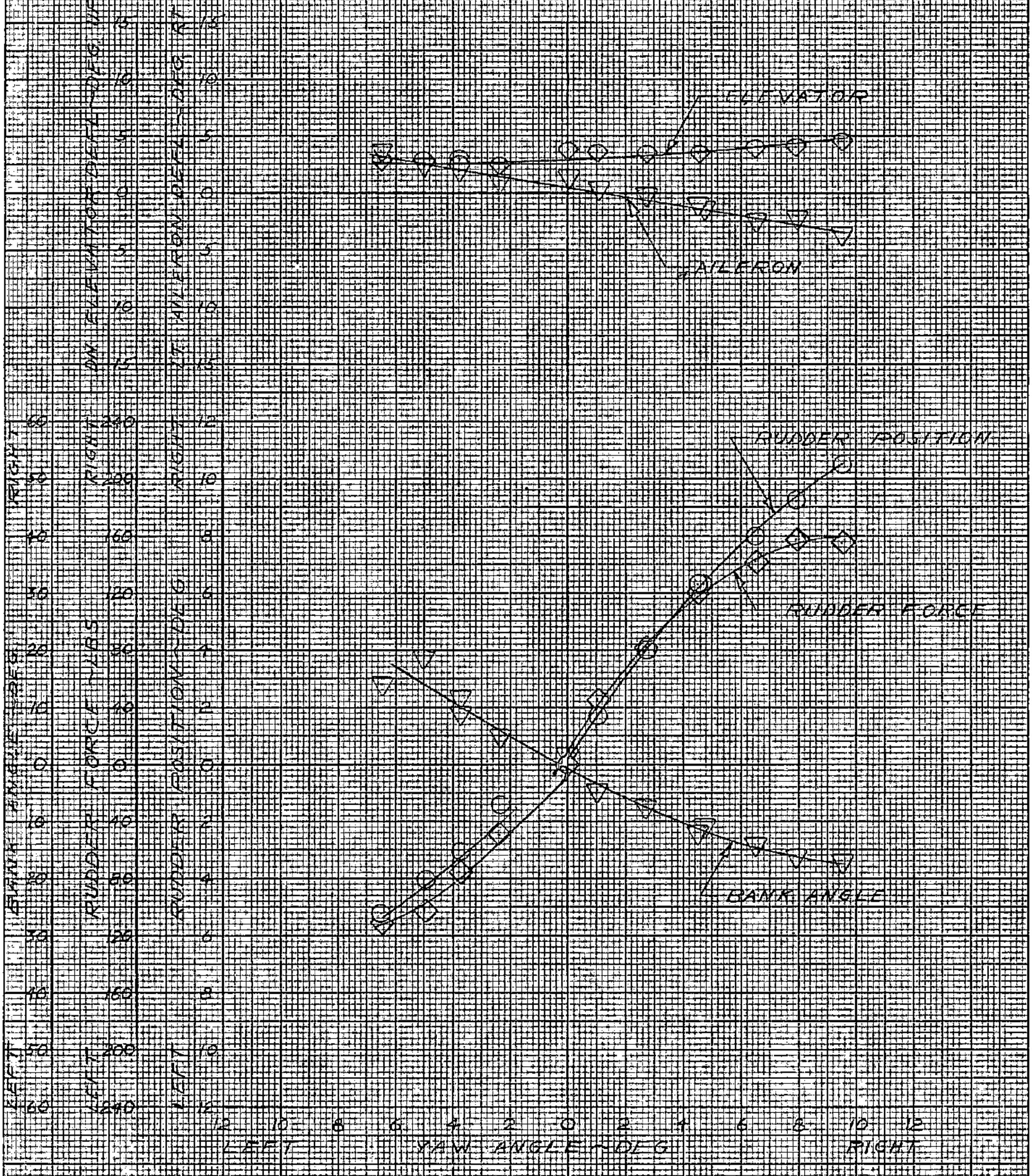


FIGURE NO. 62

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION β FEEDBACK OFF

12,000 FT. ALTITUDE MACH NO. 0.818

LIMIT STATIC RUDDER DEFLECTION ±25 DEG

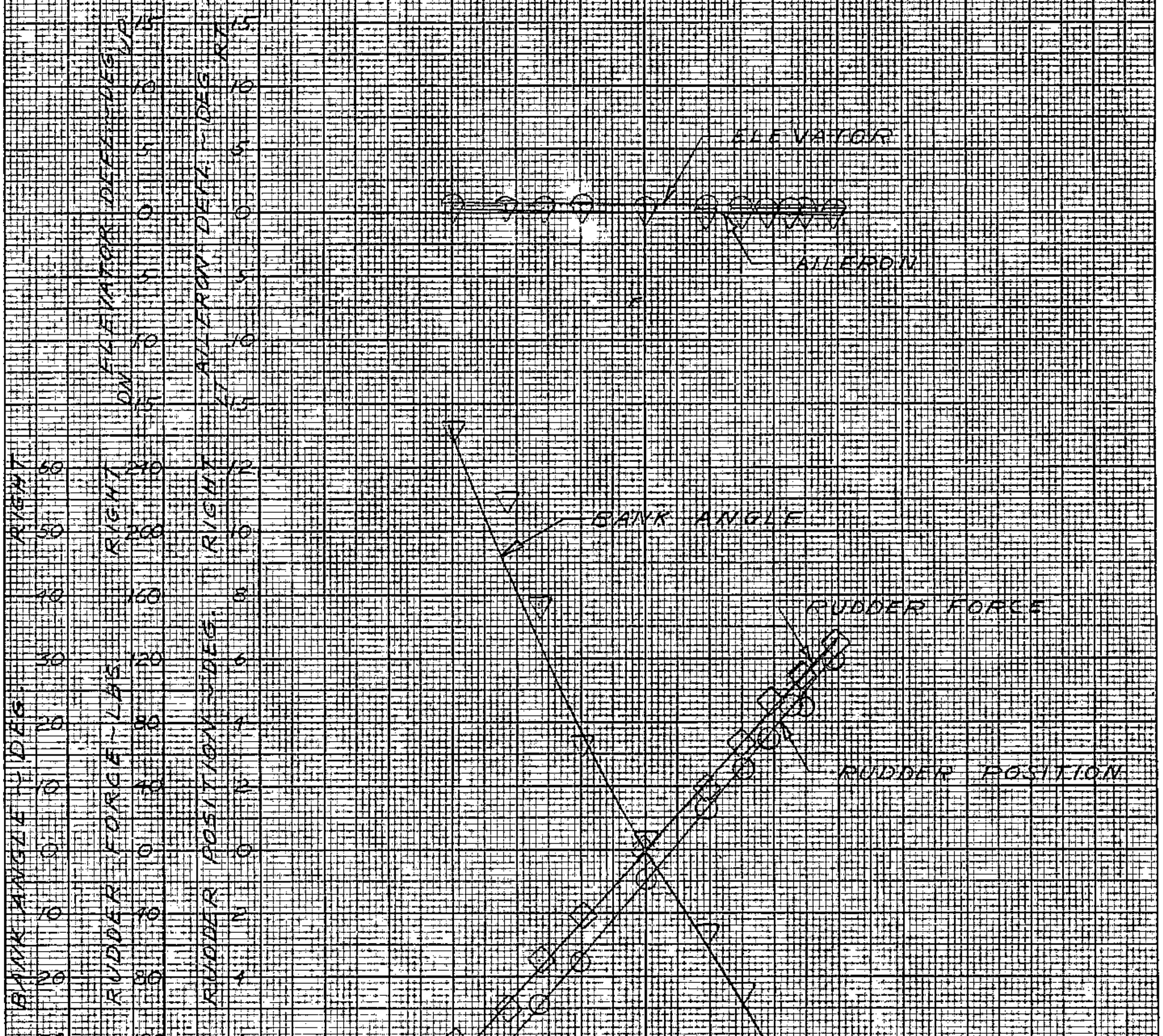


FIGURE NO. 63

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION β FEEDBACK ON

10,000 FT ALTITUDE MACH NO. 0.464

LIMIT STATIC RUDDER DEFLECTION ±25 DEG

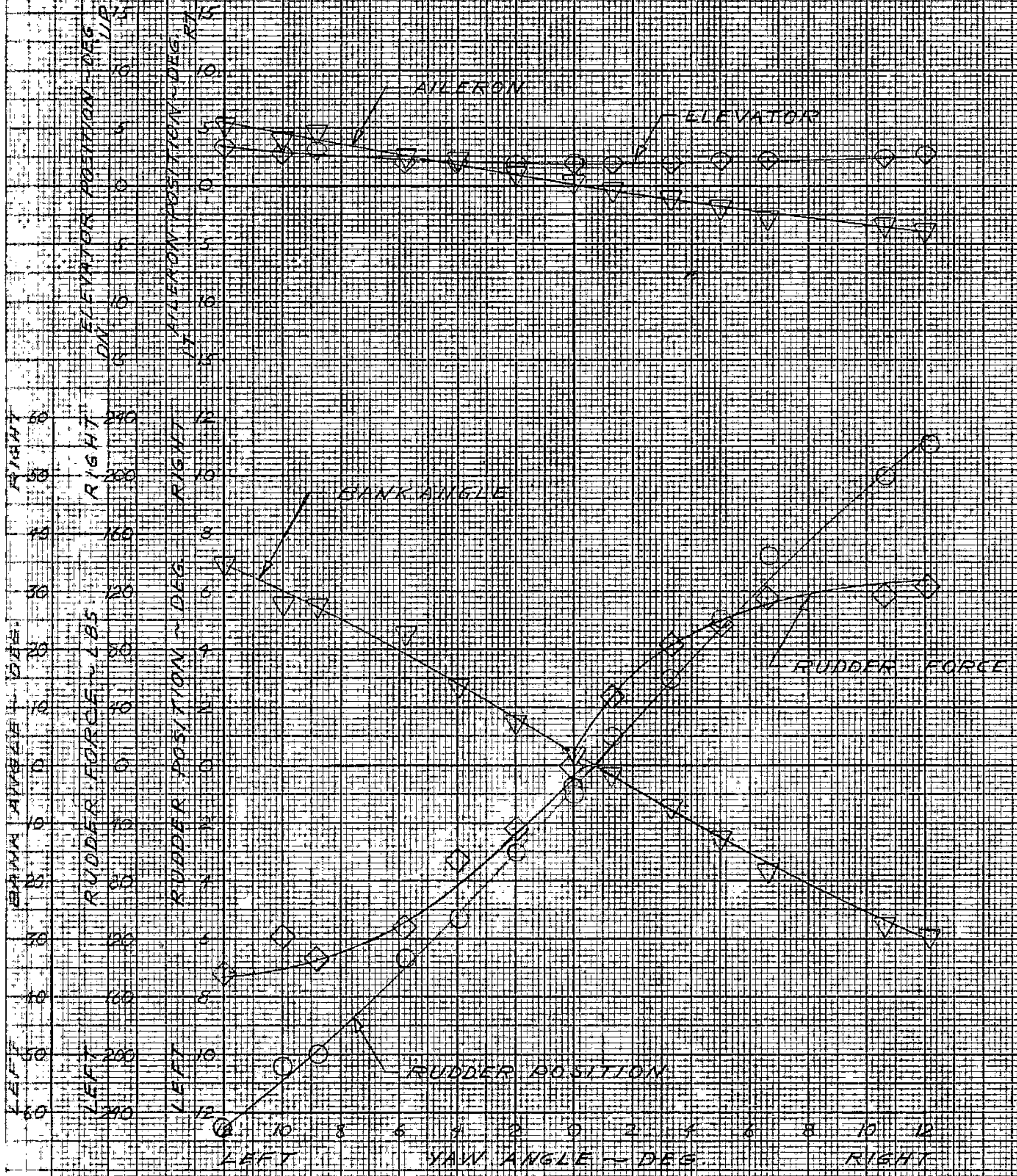
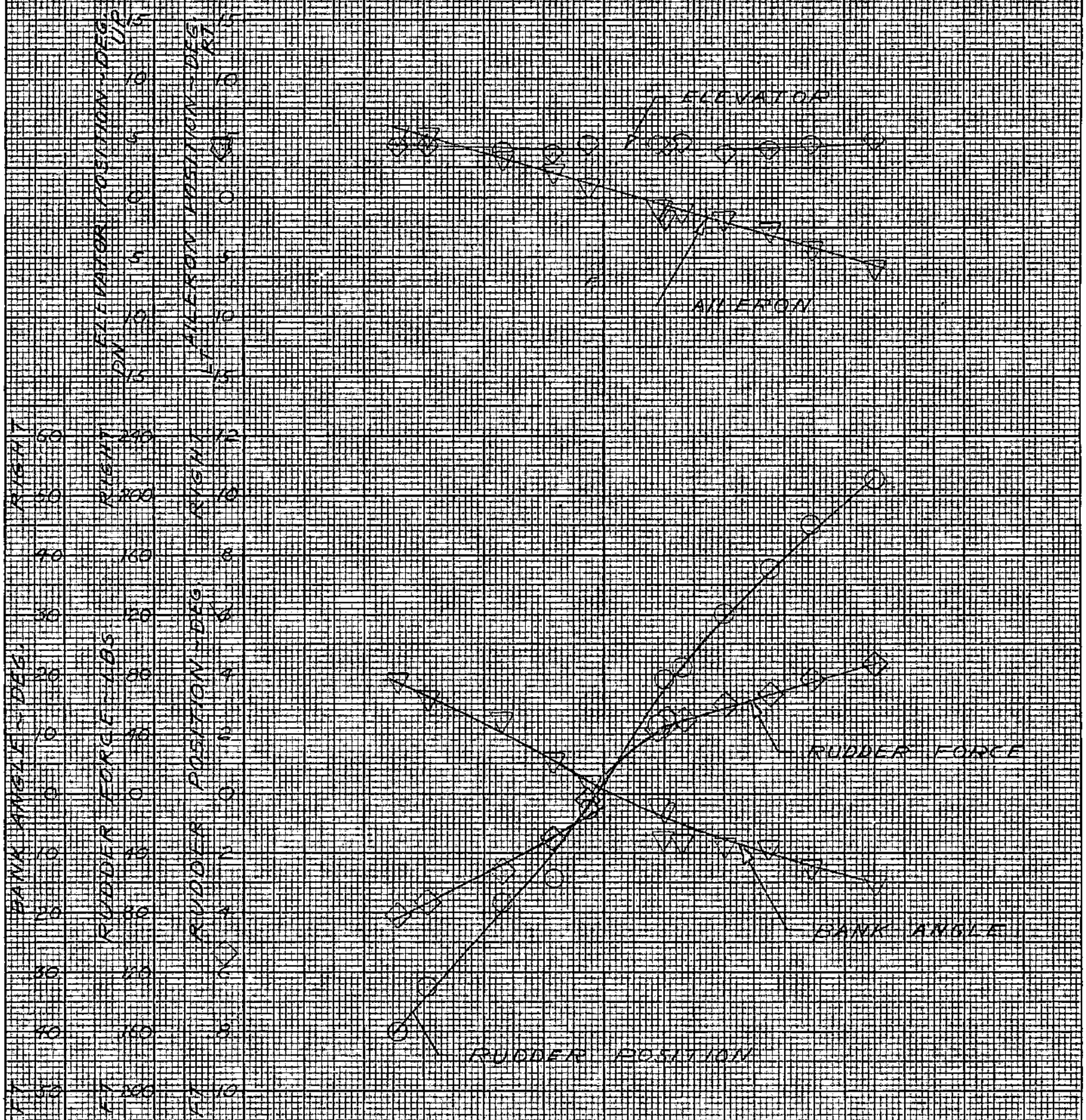


FIGURE NO. 64

STATIC DIRECTIONAL STABILITY

F-102A — USAF NO. 53-1796
PA CONFIGURATION β FEEDBACK ON
10,000 FT ALTITUDE AIRSPEED 180 KNOTS
LIMIT STATIC RUDDER DEFLECTION ± 25 DEG



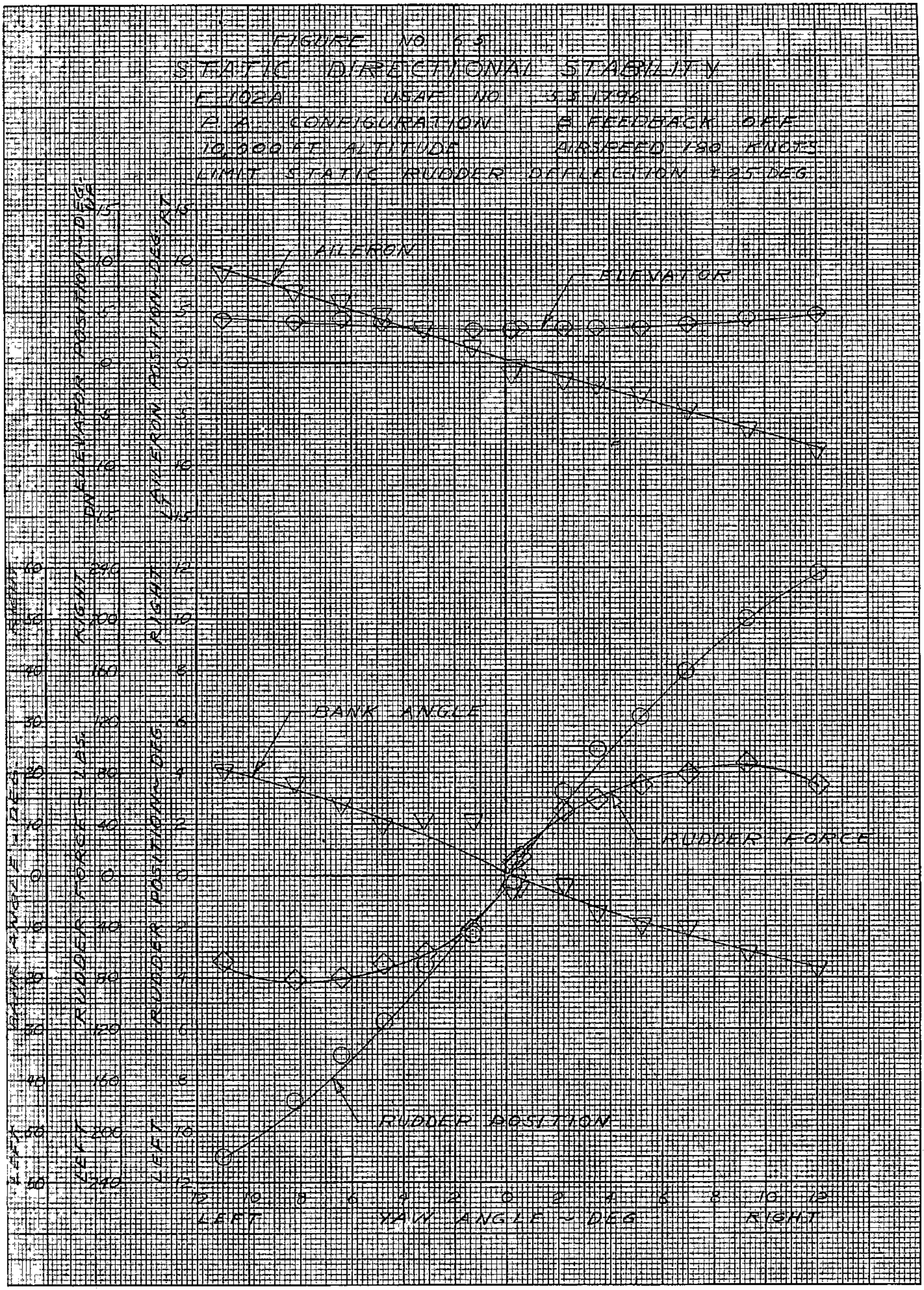


FIGURE NO. 65
STATIC DIRECTIONAL STABILITY
F-102A USAF NO. 53-1796
P.A. CONFIGURATION β FEEDBACK OFF
10,000 FT. ALTITUDE AIRSPEED 180 KNOTS
LIMIT STATIC RUDDER DEFLECTION ±25 DEG.

FIGURE NO. 66

LATERAL CONTROL

F-102A USAF NO 53-1796

YAW DAMPER ON β FEEDBACK ON

PITCH DAMPER ON TURN COORDINATOR ON

CRUISE CONFIGURATION

10.5 DEG. AILERON DEFLECTION

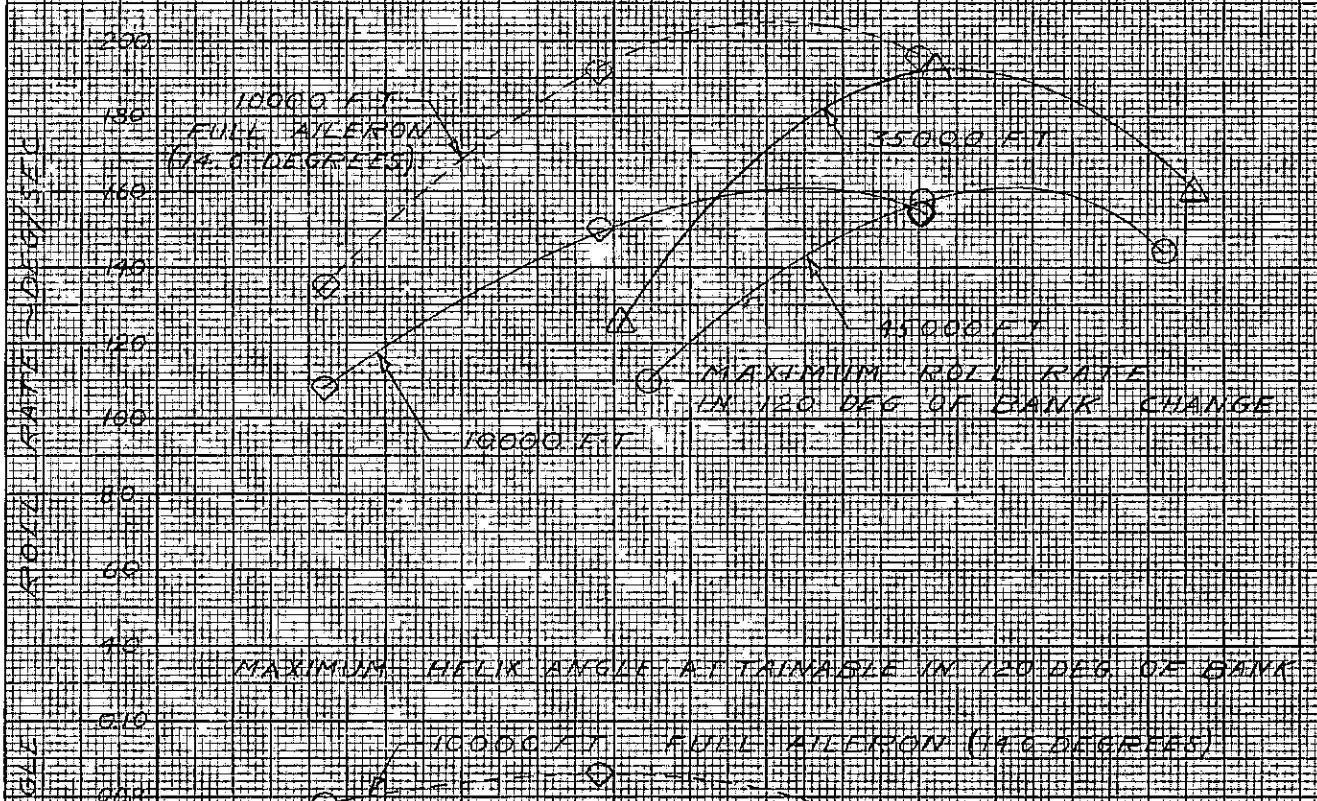


FIGURE NO. 67

LATERAL CONTROL

F-102A USAF. NO. 53-1796

CRUISE CONFIGURATION ALTITUDE 45,000 FT.

YAW DAMPER ON β FEEDBACK ON

PITCH DAMPER ON TURN COORDINATOR ON

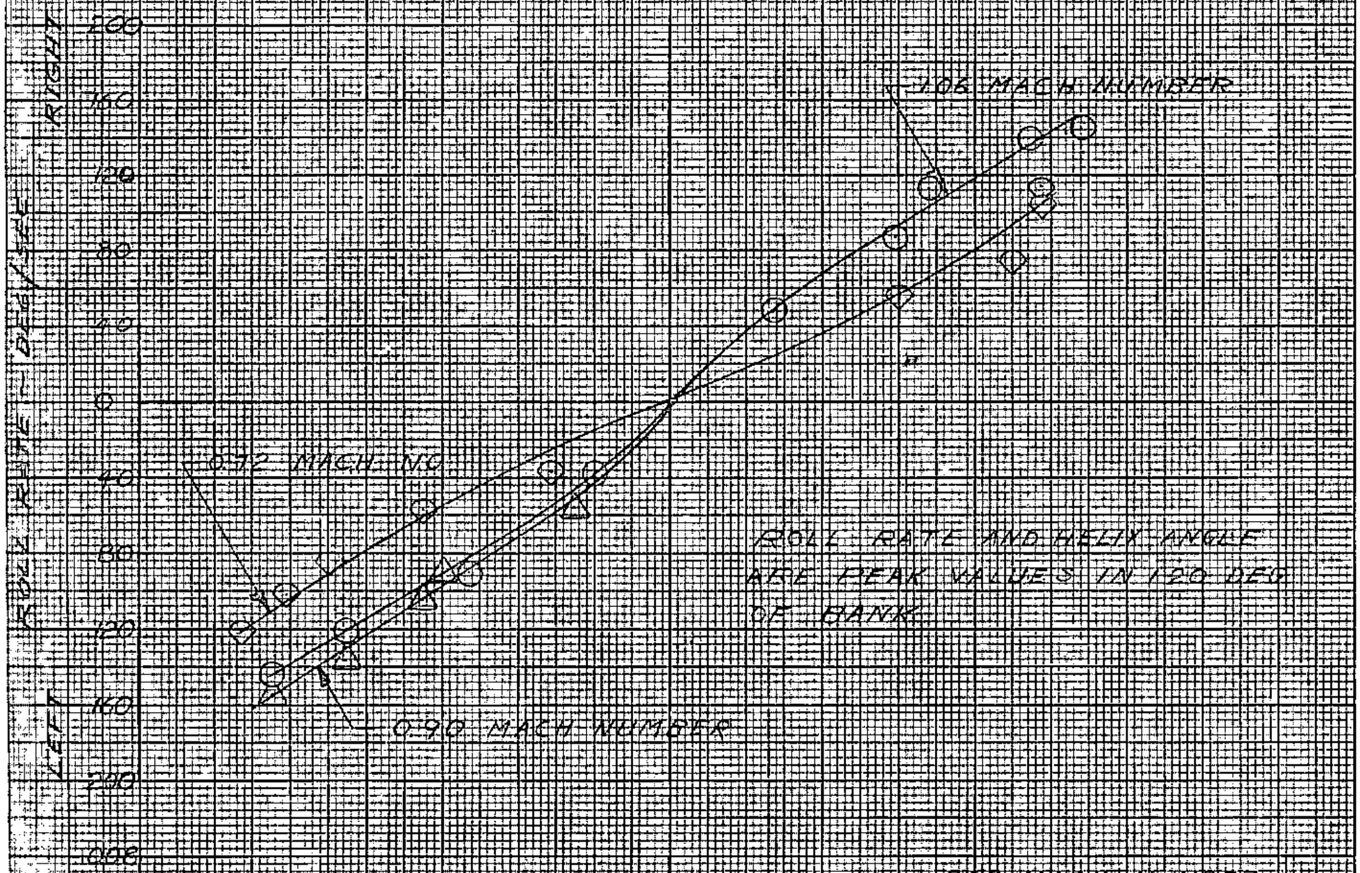


FIGURE NO 68
LATERAL CONTROL
F 102A USAF NO 53 1796
CRUISE CONFIGURATION ALTITUDE 35000 FT
YAW DAMPER ON β FEEDBACK ON
PITCH DAMPER ON TURN COORDINATOR ON

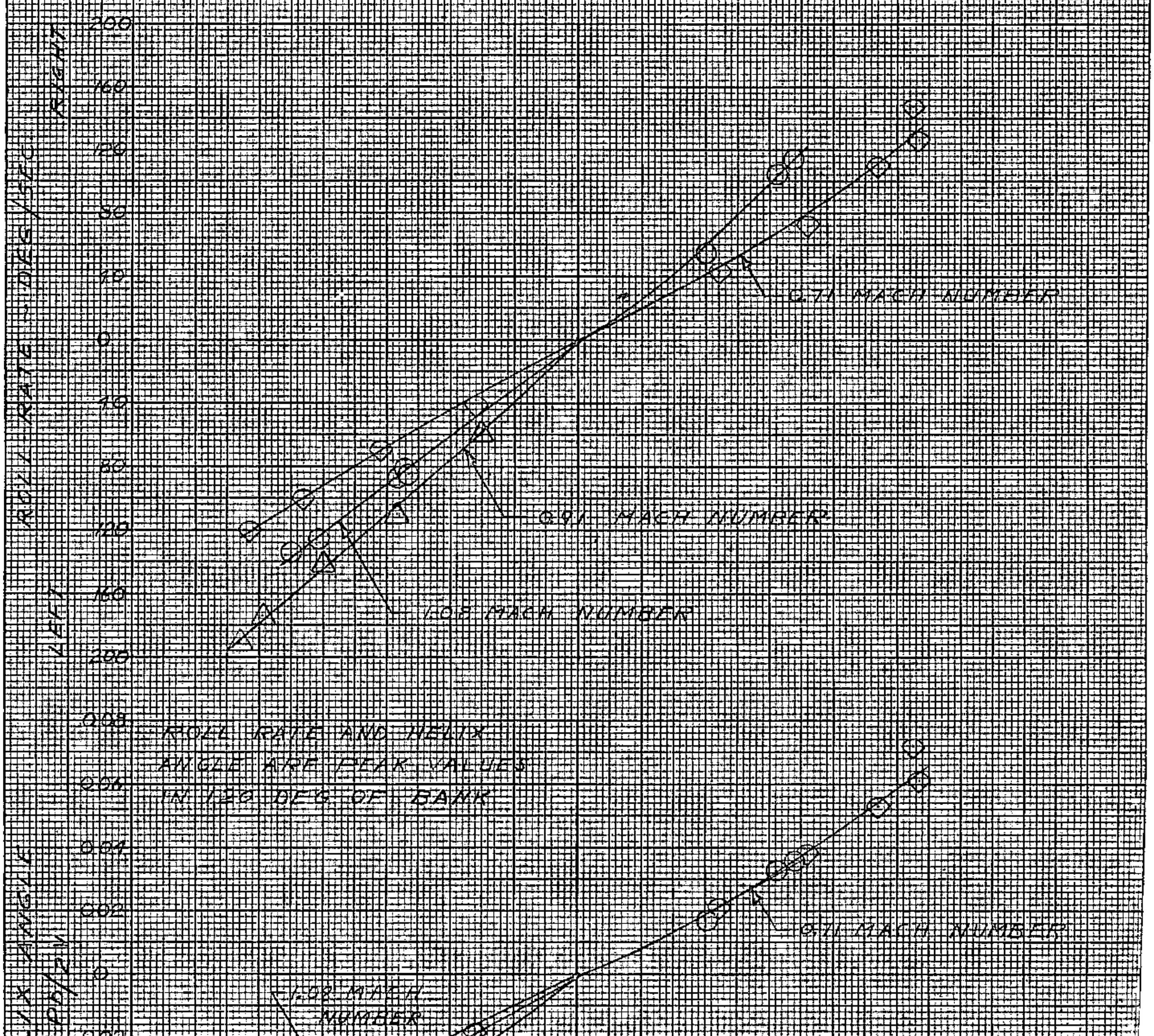


FIGURE NO. 69

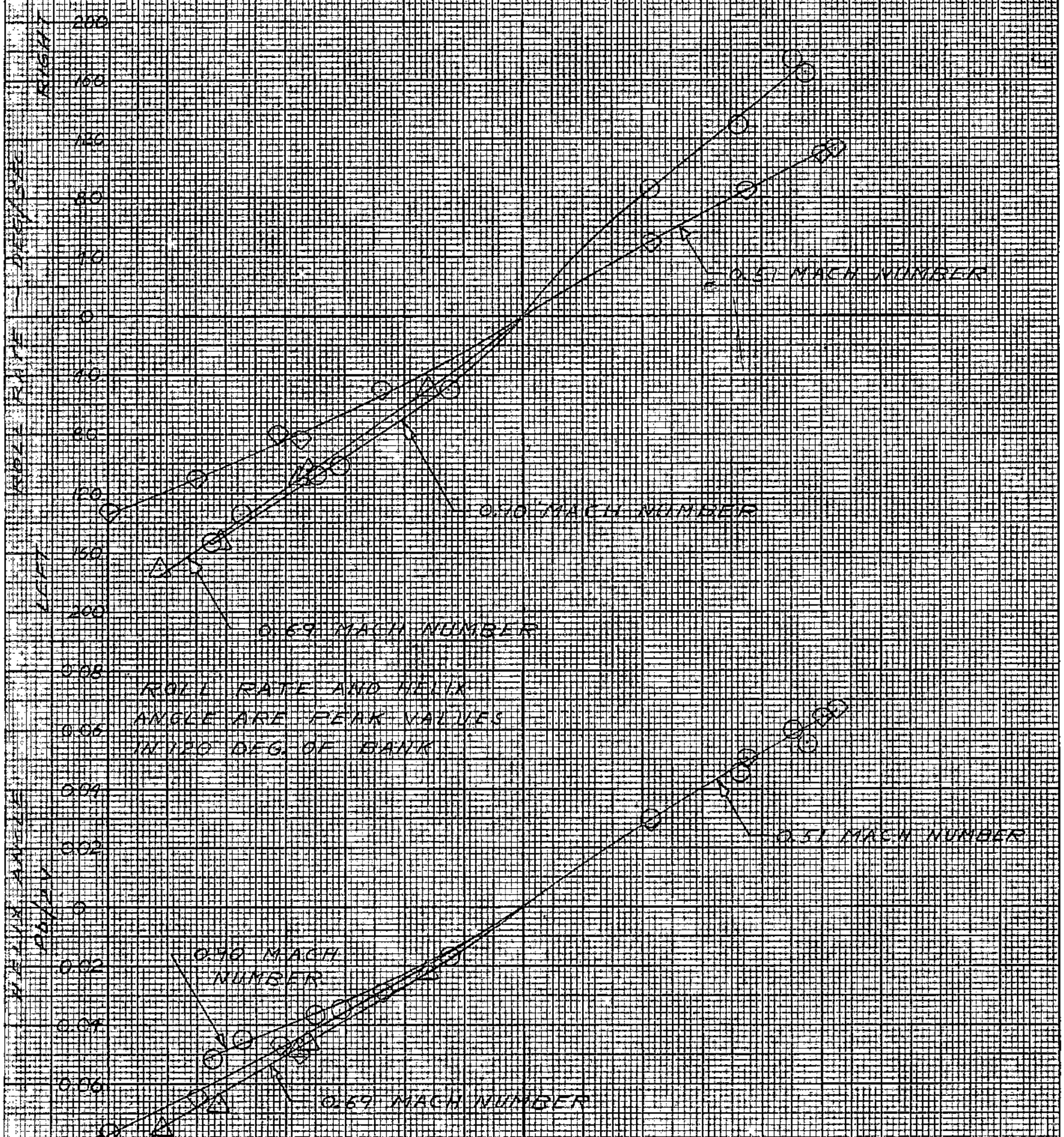
LATERAL CONTROL

F-102A USAF NO 53-1796

CRUISE CONFIGURATION ALTITUDE 10000 FT

YAW DAMPER ON β FEEDBACK ON

PITCH DAMPER ON TURN COORDINATOR ON



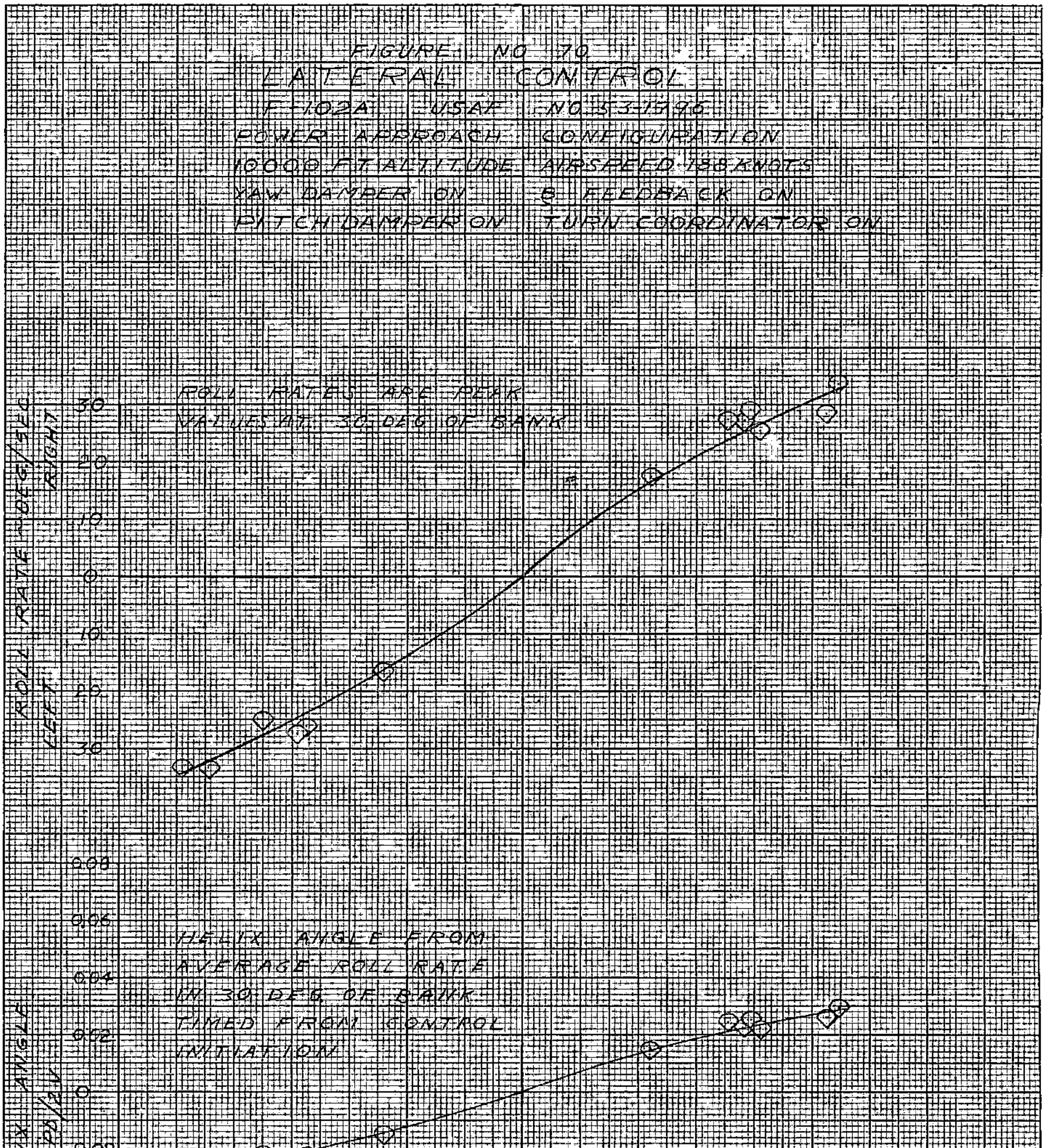


FIGURE NO 70
LATERAL CONTROL
F-102A USAF NO 53-1596
POWER APPROACH CONFIGURATION
10000 FT ALTITUDE AIRSPEED 188 KNOTS
YAW DAMPER ON β FEEDBACK ON
PITCH DAMPER ON TURN COORDINATOR ON

FIGURE NO. 71

INERTIAL COUPLING

F-102A USAF NO. 53-1796

SIDESLIP FEEDBACK ON

ROLL RATE 120 DEG/SEC.

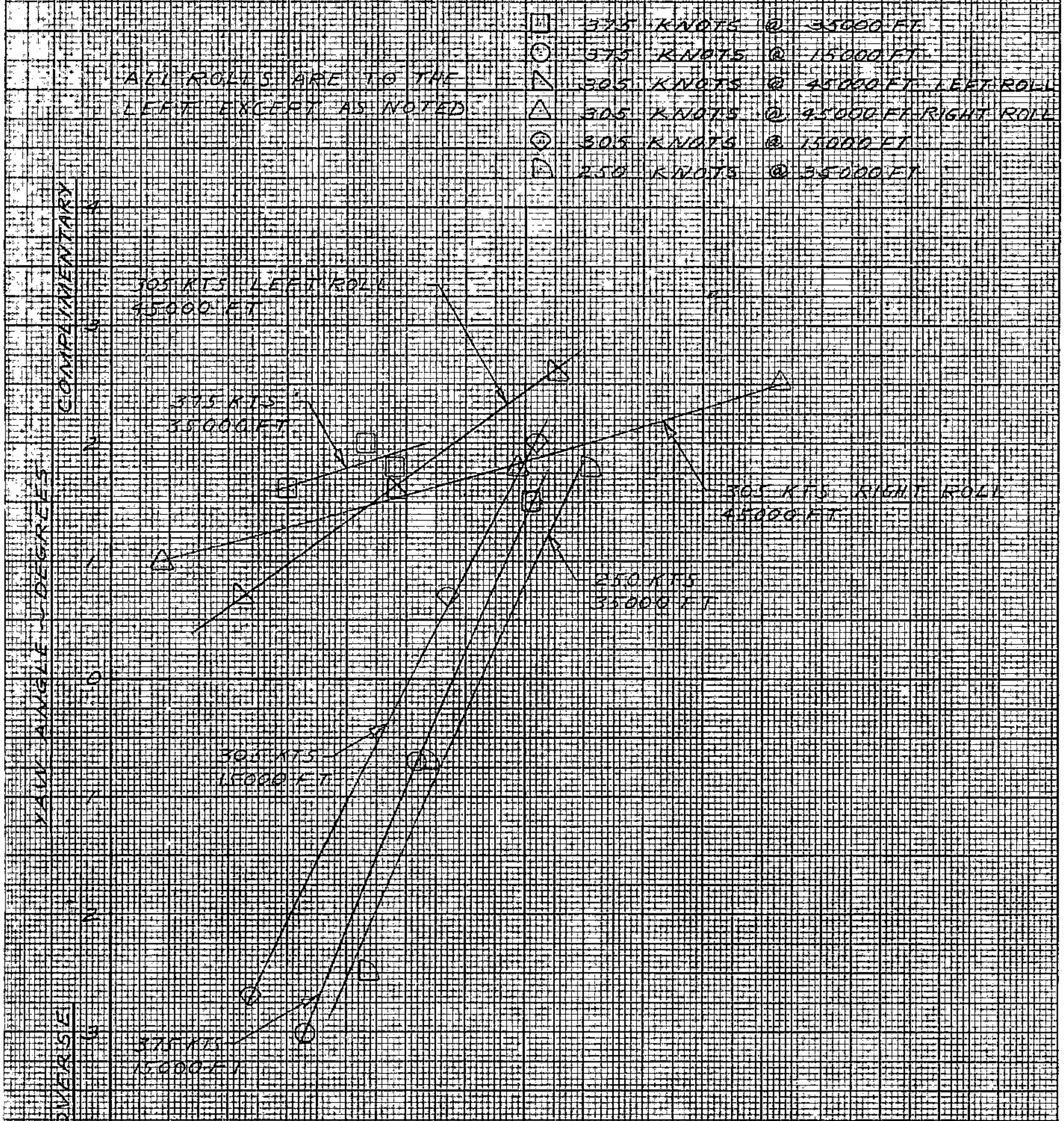


FIGURE NO. 72

INERTIAL COUPLING

F-102A USAF NO. 53-1796

SIDESLIP FEEDBACK OFF

ROLL RATE 80 DEG/SEC

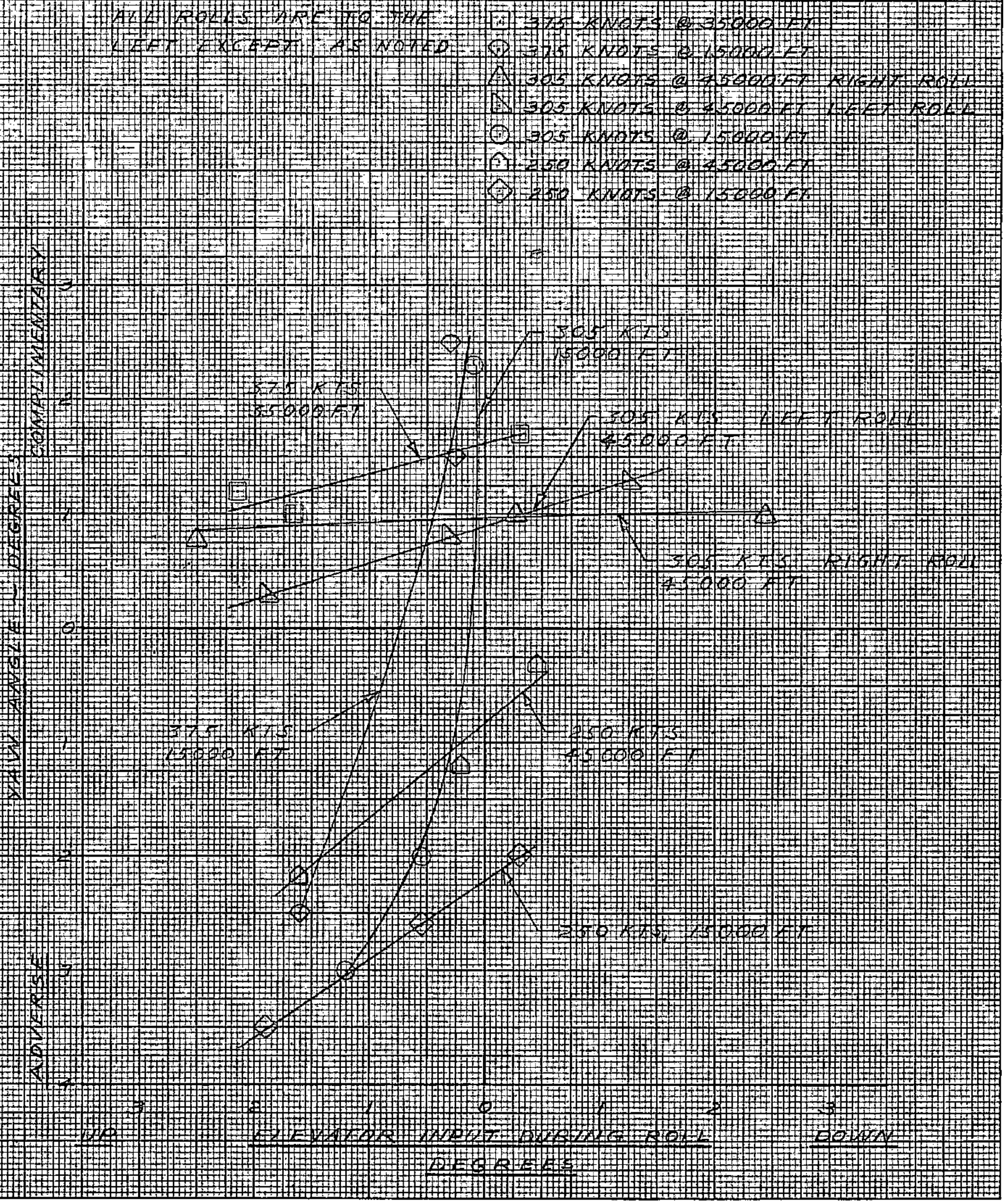


FIGURE NO. 73

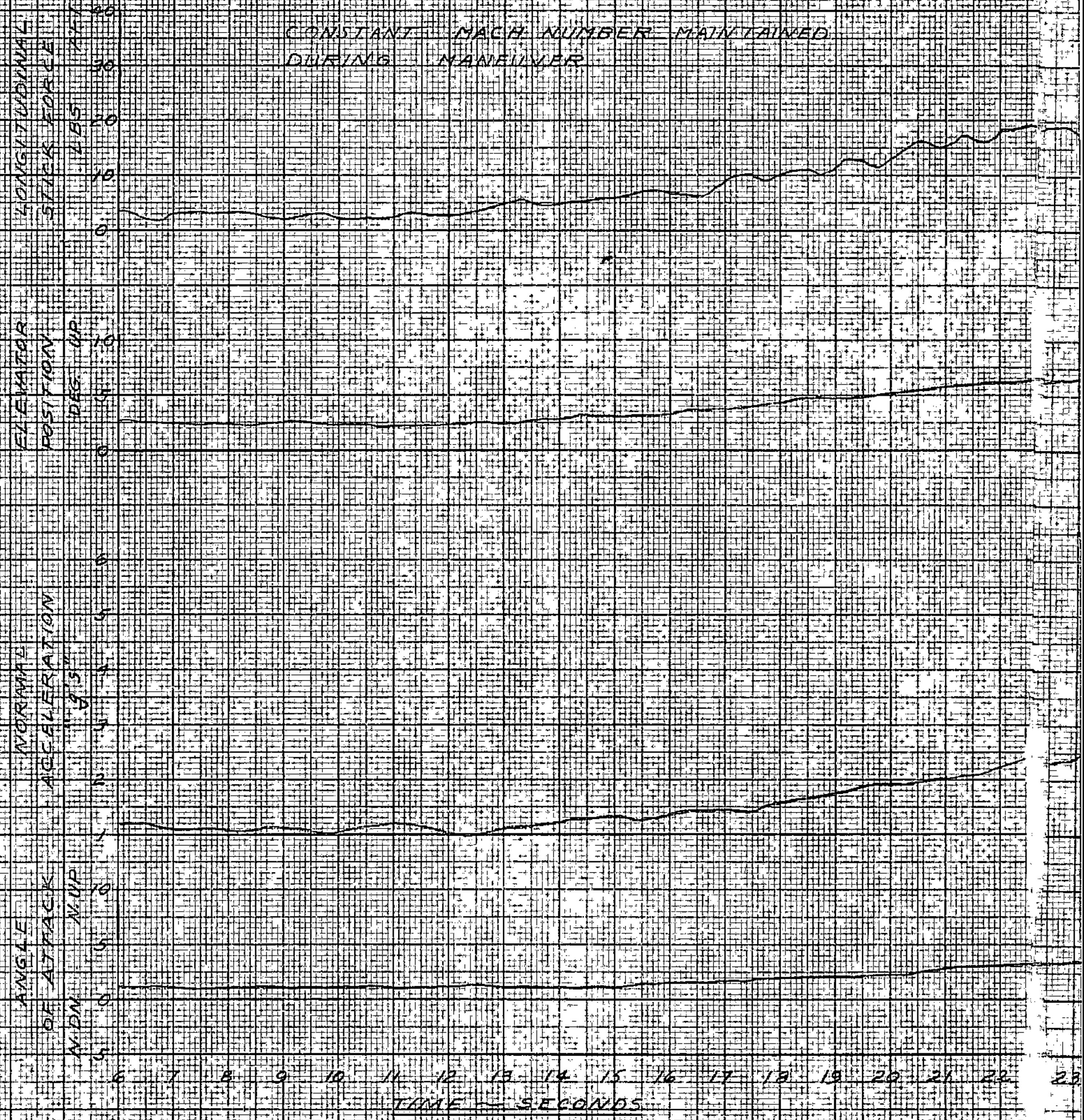
STALL TIME HISTORY

F-102A USAF NO. 53-1796

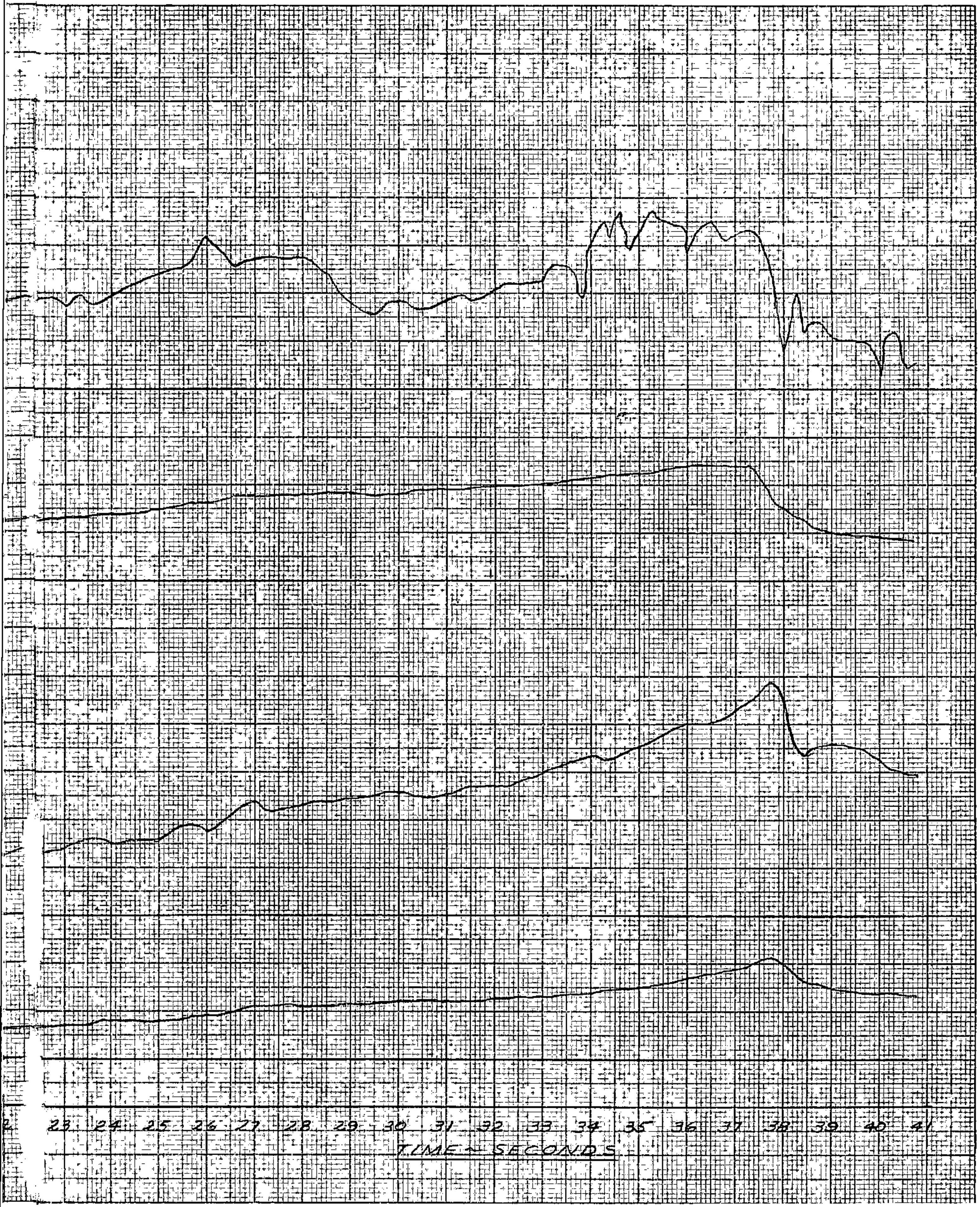
CRUISE CONFIGURATION ALTITUDE 35000 FT

WEIGHT 23800 LBS C.G. 27.0 PERCENT MAC

TRIM MACH NUMBER 0.975



CONSTANT MACH NUMBER MAINTAINED DURING MANEUVER


2
23
24
25
26
27
28
29
30
31
32
33
34
35
36
37
38
39
40
41
TIME ~ SECONDS

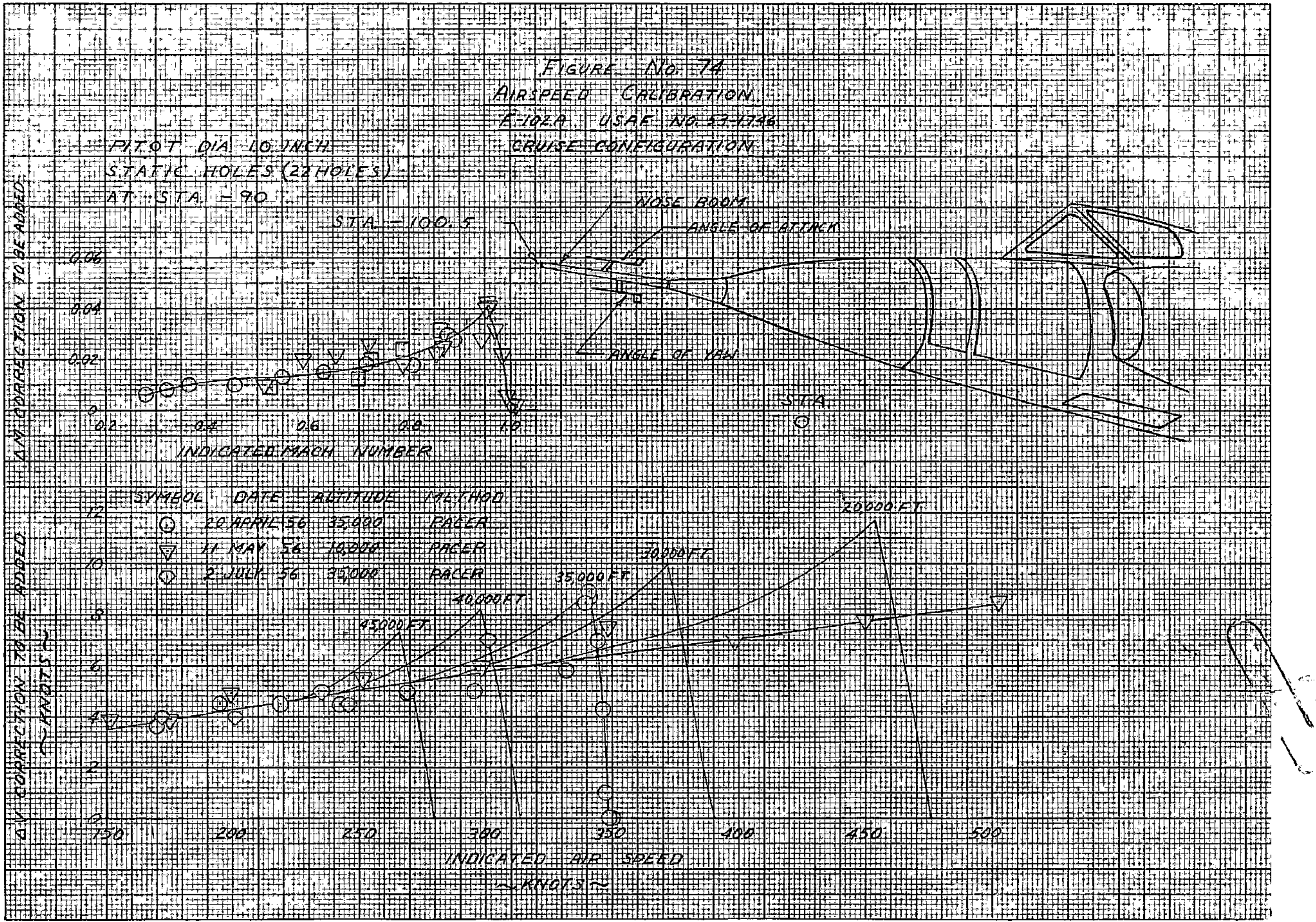


FIGURE No. 74
AIRSPEED CALIBRATION
F-102A USAF NO. 53-1746
CRUISE CONFIGURATION

FIGURE NO. 75

LONGITUDINAL BREAKOUT FRICTION

F-102A USAF No 53-1796

MEASURED IN A CLOSED HANGAR

TEMPERATURE 25° C

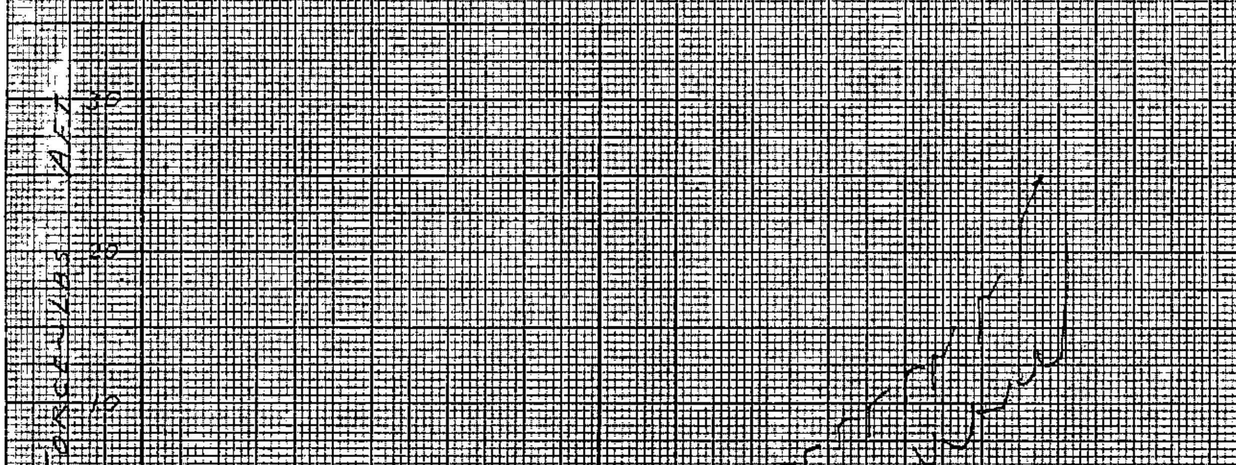


FIGURE NO. 76

LATERAL BREAKOUT FRICTION

F-102A USAF NO. 53-1796

MEASURED IN A CLOSED HANGAR

TEMPERATURE 20°C

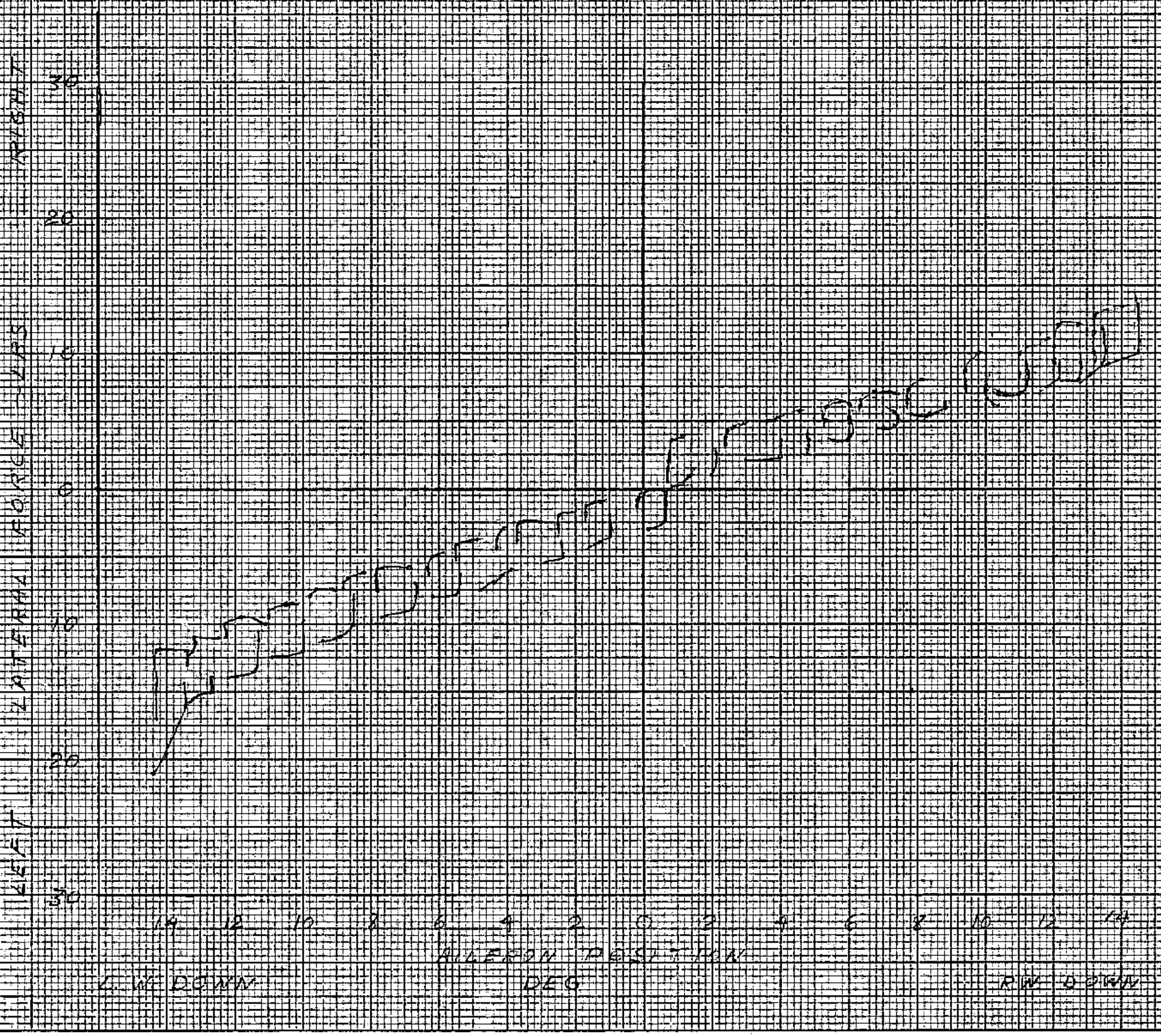
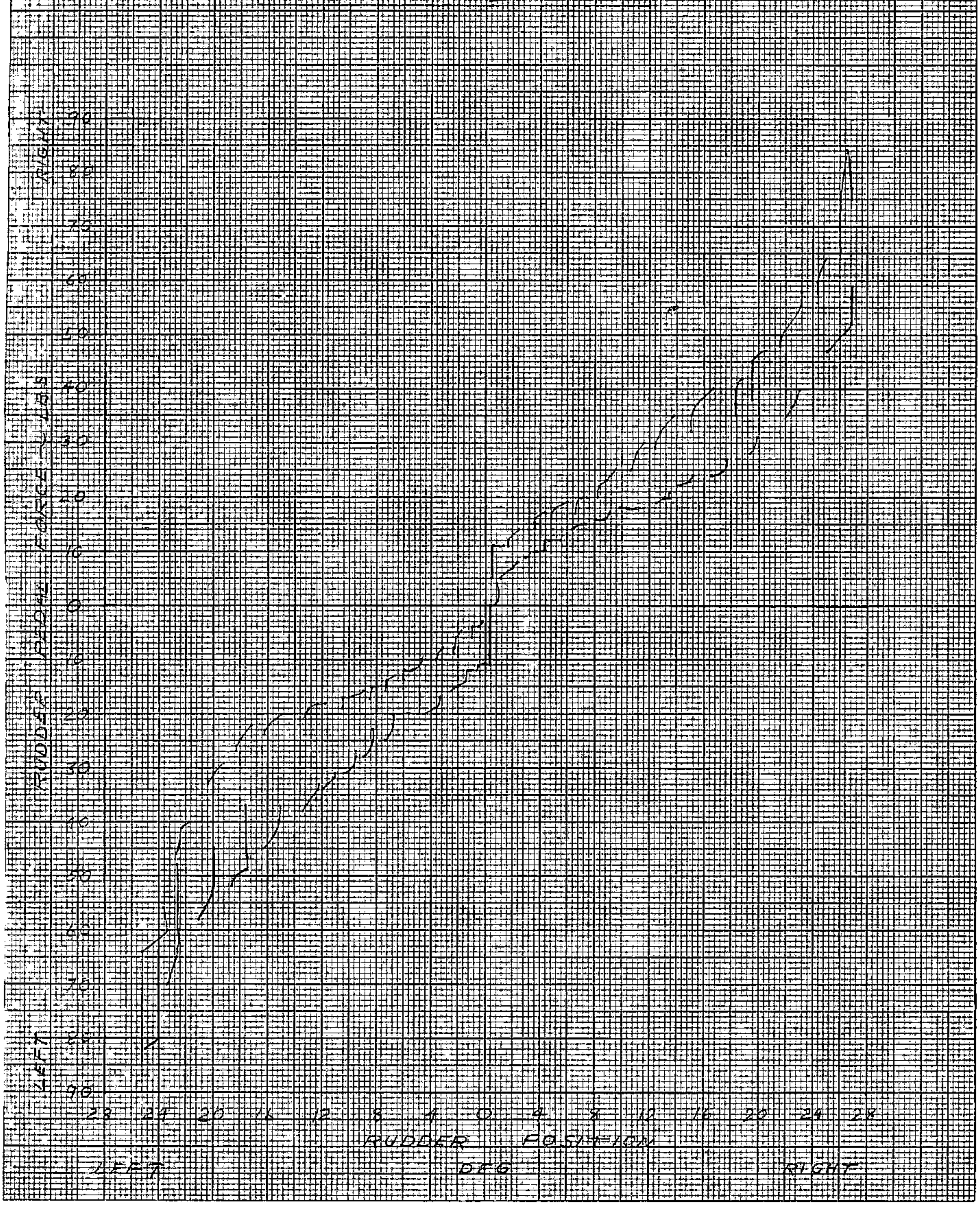


FIGURE NO. 77
RUDDER BREAKOUT FRICTION
F-102A USAF NO 53-1796
MEASURED IN A CLOSED HANGAR
TEMPERATURE 25° C



APPENDIX II

general aircraft information

WING:

area	695.05 sq. ft.
span	38 ft. 1.6 in.
MAC	23 ft. 9.1 in.
aspect ratio	2.189
airfoil section	NACA 0004-65 (mod)
incidence	0°
dihedral	0°
leading edge sweepback	60°
trailing edge sweep forward	5°
camber	7.5 percent C

VERTICAL TAIL:

area	68.33 sq. ft.
span	8.67 ft.
MAC	10.513 ft.
aspect ratio	1.10
airfoil	NACA 0004-65 (mod)
leading edge sweepback	60°
trailing edge sweep forward	5.0°

FUSELAGE:

length (including boom)	68 ft. 3.3 in.
wheel base	23 ft. 3.9 in.
tread (main wheels)	14 ft. 6.0 in.

ELEVONS:

area	68 sq. ft.
span	12 ft. 10.75 in.
deflection	34° −19°

RUDDER:

area	7.43 sq. ft.
deflection	25°

FIN:

area	60.90 sq. ft.

☐ flight limitations

ENGINE:

Maximum power	1 minute on ground 15 minutes inflight
Military power	5 minutes on ground 30 minutes inflight
exhaust gas temperature	630 °C below 35,000 ft.
limit	660 °C above 35,000 ft.

AIRSPEED:

clean	510 knots 1.25 Mach number
gear extended	220 knots
drag chute	160 knots
canopy open	30 knots
ram air turbine	345 knots

LOAD FACTORS:

normal	5.5 g −2.4 g
lateral	1.3 g

Aircraft is restricted from inverted flight and to 3 seconds of zero or negative g flight.

ROLL LIMITATIONS:

Maximum roll rate of 80° second for continuous roll maneuvers with sideslip feedback off.

Maximum rate of 120° second for continuous roll maneuvers with sideslip feedback on.

Unrestricted in roll rate in the top two quadrants.

Uncoordinated maneuvers are prohibited.

☐ miscellaneous

Armament bay doors closed in flight.
Pilot assist system deactivated.
Inlet duct ramps not installed.
Pitot heater inoperative.
No instrument panel lighting.

flight control systems

Longitudinal and lateral control is accomplished by movement of the elevons; in parallel motion as elevators, differentially as ailerons or in any combination of the two motions. Conventional stick movements are employed for elevon control. These control surfaces are actuated by irreversible full hydraulic power control systems which include artificial feel, gyro-stabilized yaw and pitch damper systems and trimming devices. Pilot control motion mechanically meters hydraulic pressure to the control surface actuating cylinders. The control surfaces move only while the controls are being moved. When pilot control motion stops, a mechanical follow-up mechanism shuts off the control valve.

A conventional rudder affords directional control. The rudder surface is actuated by an irreversible full hydraulic power system including a gyro stabilized yaw damper system, a trimming device, a turn coordinator and sideslip feedback. The hydraulic power system functions in the same manner as described above.

Elevator feel force is made proportional to control displacement by a variable linkage. A pneumatic cylinder, modulated by dynamic pressure, provides a variation with airspeed. A trim servo automatic trimming device that senses Mach number provides positive control in the unstable transonic speed range. Rudder feel is obtained from the combined action of an overload control spring assembly and a piston type feel cylinder which is loaded by amplified dynamic pressure. This system limits the lateral acceleration to a safe maximum, yet provides adequate travel for maneuverability. Aileron feel is made directly proportional to control displacement by a simple spring device.

Trim is accomplished electrically by displacing the control surface positions and in turn repositioning the neutral point of the cockpit controls.

fuel system

All fuel is carried in right and left wing tank systems. Each system consists of three integral tanks. Except for a common, single-point refuel-defuel fitting the two wing systems are independent of each other, and feed fuel to the engine through a common supply line either simultaneously or separately. Tank pressurization forces fuel from the aft outboard tank to the forward tank and then to the aft inboard tank where the booster pump is located. Total fuel capacity is 1,110 gallons, 1,084 gallons of which is usable fuel.

power plant

A Pratt and Whitney J57-P-23 turbojet engine with afterburner (flap type nozzle, single cylinder, sheet metal flap) is installed in this aircraft. The engine is rated at 16,000 pounds sea level static thrust with afterburning and 9,880 pounds non-afterburning.

weight and balance

The aircraft was ballasted prior to initiation of the program to a forward center of gravity at combat weight. The gross weight at engine start was 27,500 pounds including pilot, fuel and oil. The center of gravity varied with fuel consumption from 28.65 percent MAC at engine start to 23.5 percent MAC when the fuel load was expended. During the test program the aircraft was reballasted to 27,400 pounds with a center of gravity of 31.7 percent MAC at engine start. With this loading it was necessary to burn 1200 pounds of fuel to arrive at a center of gravity of 30.5 percent at take-off. The fuel expended center of gravity was 27.5 percent MAC. The aircraft was finally ballasted to a gross weight of 27,600 pounds with a mid center of gravity. The center of gravity was 29.8 percent MAC at engine start and 25.1 percent MAC with the fuel expended.

test instrumentation

Pilot's Panel:
- Angle of Attack
- Angle of Yaw
- Airplane Attitude
- Indicated Airspeed
- Indicated Mach Number
- Pressure Altitude
- Rudder Surface Position
- Left Elevon Position
- Right Elevon Position
- Normal Acceleration
- Rate of Roll
- Total Fuel
- Engine rpm (High Pressure Rotor)
- Turbine Exit Temperature
- Afterburner Nozzle Position
- Correlation Counter
- Outside Air Temperature
- Time of Day
- Turn and Bank Indicator
- Directional Heading
- Fuel Flow Totalizer

Photo Panel:
- Indicated Airspeed
- Pressure Altitude
- Airplane Attitude
- Outside Air Temperature
- Normal Acceleration
- High Pressure Rotor rpm
- Low Pressure Rotor rpm
- Turbine Discharge Total Temperature
- Time of Day
- Correlation Counter
- Lateral Trim Position
- Longitudinal Trim Position
- Directional Trim Position
- Fuel Flow Totalizer
- Afterburner Nozzle Position
- Turbine Discharge Pressure
- Directional Heading

Oscillograph:
- Angle of Attack
- Angle of Yaw
- Longitudinal Stick Force
- Longitudinal Stick Position
- Lateral Stick Force
- Lateral Stick Position
- Rudder Surface Position
- Rudder Pedal Force
- Left Elevon Position
- Right Elevon Position
- Normal Acceleration
- Lateral Acceleration
- Free Stream Differential Pressure
- Free Stream Static Pressure
- Rate of Roll
- Rate of Yaw
- Rate of Pitch
- Angle of Bank
- Angle of Pitch
- Instrumentation Voltage Reference
- Camera Pulse Interval

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The F-102A has no stability and control deficiencies which seriously affect its all-weather intercept capability as long as the stability augmentors are operating. The aircraft can be flown safely without stability augmentation, but its ability to successfully accomplish an intercept mission under these conditions is questionable.

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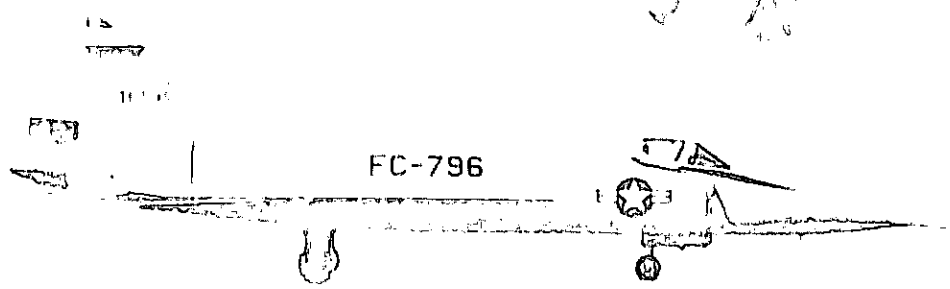

F-102A
PHASE IV
STABILITY TEST

AFFTC-TR-57-5 ☐ April 1957

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J. Reverdy Allender, Project Engineer

Robert M. White, Captain, USAF, Project Pilot

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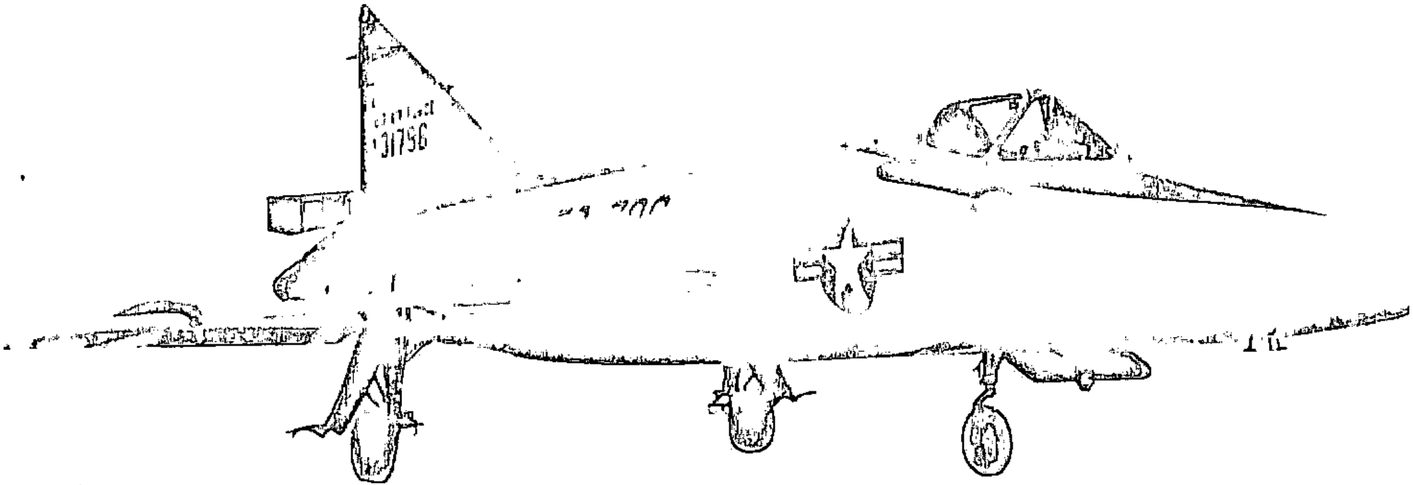
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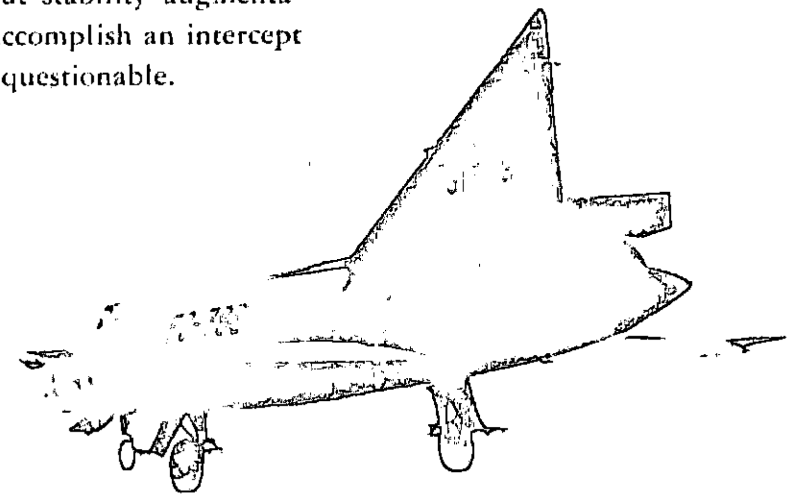
ABSTRACT

The F-102A has no stability and control deficiencies which seriously affect its all-weather intercept capability as long as the stability augmentors are operating. The aircraft can be flown safely without stability augmentation, but its ability to successfully accomplish an intercept mission under these conditions is questionable.



This report has been reviewed and approved **19 MARCH 1957**

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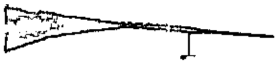
INTRODUCTION

Phase IV stability and control flight tests were conducted on F-102A USAF S/N 53-1796 to obtain data for the Pilot's Handbook of Operating Instructions and to determine compliance with the flying requirements of Military Specification MIL-F-8785 (ASG).



The tests were performed at the Air Force Flight Test Center, Edwards Air Force Base, between 19 April and 14 September 1956. The program consisted of 41 flights, requiring 38 hours and 55 minutes.

The F-102A is a single-place, high altitude, all weather interceptor powered by a J57-P-23 jet engine rated at 16,000 pounds static thrust with afterburning.

The airplane is equipped with yaw and pitch dampers, a turn coordinator, and a sideslip feedback. The sideslip feedback is provided only on the "small tail" airplanes until such time as they are retrofitted with a vertical tail of 40 percent larger area. (The first 62 production aircraft will initially have the small tail.) On the test aircraft provision was made to control the yaw and pitch dampers, turn coordinator, and sideslip feedback individually. On the standard aircraft one switch would control all these functions.

The dual head "yoke" stick, with radar control in the left grip, was installed during the tests.

The control system employs two separate irreversible hydraulic systems with a bungee system to provide feel. In addition to the bungee system there is a "q" (dynamic pressure) sensing device in the directional and longitudinal systems to vary control force with airspeed, and an automatic trim servo in the longitudinal system to compensate for unstable stick force in the transonic region. Control is provided by elevons located on the trailing edge of the wing and by a conventional rudder.

The airplane is characterized by a thin, 60 degree delta wing and half delta vertical tail, a sharp "V" shaped canopy, waisted fuselage, two air intake ducts on the sides of the fuselage, and a fully enclosed armament bay. The main gear retracts inboard into the wing and fuselage, and the nose gear retracts forward into the nose section of the fuselage. A clamshell type speed brake is installed above the tail cone. The speed brakes also act as doors for the drag chute compartment.

Sm[illegible]

[illegible]

TEST RESULTS

cockpit evaluation

Entrance to the cockpit is made by climbing a ladder attached to the left engine duct. It is difficult to climb the ladder when wearing full flight equipment, since it hangs past the vertical. The ladder should be modified so that the pilot moves forward toward the airplane as he climbs up. Emergency ground exit can be made by hanging from the engine duct and dropping to the ground.

The canopy hold-open rod consists of a tubular metal bar attached to the canopy, which rests on a receptacle inside the canopy sill. The rod is unsatisfactory for taxi operation, since winds above 30 knots will cause the canopy to fly up with possible damage to the airplane, and vibration from taxi operation may cause the hold-open rod to slip, allowing the canopy to close with possible injury to the pilot. Since taxi operation with the canopy closed in hot weather causes excessive pilot discomfort, a means should be provided to positively lock the canopy in any position.

The "T" on the canopy locking handle is oriented vertically. The pilot rubs his knuckles against the side of the cockpit when moving the handle, and the handle partially obstructs view of the first three or four warning lights on the warning light panel when it is in the forward position. Both of these discrepancies can be eliminated by orienting the "T" horizontally.

When the canopy is almost shut, high pressure air is relieved from the counter balance cylinder to permit the locking claws to be engaged easily. On several occasions the air was not released, and two or three attempts and up to 10 minutes were required to get the canopy closed. Aside from anoyance and pilot effort involved, scramble take-off procedures cannot tolerate the time lost in this simple operation. Canopy cinch down must be made more reliable.

The control handles for the drag chute, emergency gear extension, and canopy locking should be identified by placarding directly on the handles.

The rudder trim switch, located on the instrument panel pedestal behind the control stick, is difficult to locate and operate. Since the rudder trim is used frequently, the switch should be made more accessible by moving it to the left console as near the throttle quadrant as possible.

Labeling of the yaw and pitch damper switch is ambiguous. It is called a "FLIGHT MODE" switch and its two positions are labeled "DIRECT MANUAL" and "MANUAL." The switch should be labeled "STABILITY AUGMENTORS" and the two positions labeled "ON" and "OFF."

The anti-ice switch is three position: "AUTOMATIC" in the up position, "OFF" at center, and "MANUAL" in the down position. When the switch is moved from "AUTOMATIC" to "OFF," the pilot can easily overshoot to the "MANUAL" position and deplete his anti-ice fluid reservoir. The switch should be replaced by one similar to the cabin pressure switch in this aircraft, which would eliminate the possibility of inadvertently moving the switch to the "MANUAL" position.

Five rheostat switches are presently used to control cockpit lighting: 1. flight instrument group edge lighting and standby compass; 2. instrument panel (other than flight instruments) edge lighting; 3. console edge lighting; 4. instrument panel floodlighting; and 5. console floodlighting. The number of controls should be reduced to three to simplify the problems of control identification and operation. One control should be provided for all 115 volt AC edge lighting, one control for floodlighting, and the third for the instrument flight group lighting (28 volt DC).

The MA-4 lap belt cannot be pulled tight enough to hold a small pilot securely. A seat belt which can be adjusted to fit all pilots is considered mandatory from a safety standpoint.

The oxygen hose provided as part of the ship's system is not long enough for proper use with the MC-1 or MC-3 altitude suit. The length of the oxygen hose should be increased at least 4 but not more than 6 inches. It is believed that a hose longer than 36 inches (6 inches longer than that presently installed) may interfere with operation of cockpit controls.

The present fuel quantity indication allows a check of total fuel on board and fuel remaining in the No. 3 tank in each wing, but does not permit a check of the total fuel remaining in each wing. In the case of asymmetric fuel flow such as with a flow divider failure, it is mandatory for the pilot to be able to monitor the total fuel remaining in each wing. The system should be modified to allow the pilot to read both the total fuel and the quantity in each wing.

There is no immediate and positive means by which the pilot can determine if the fuel shut-off valves have operated properly in accordance with the position of the fuel shut-off valve selector control. Small indicator lights should be installed on the fuel selector panel to show open positions of the shut-off valves.

The windshield anti-reflection curtain sags, stretches, and it is easily torn, resulting in interference with the pilot's vision. It should be replaced with a durable fixed divider.

The high pressure pneumatic system incorporates a priority valve to provide a reserve of 1500 psi air pressure for emergency gear extension, ram air turbine displacement, drag chute operation, and wheel brakes. A cockpit warning light indicates that the entire system has depleted to 1500 psi, but does not show the priority system air pressure. If the air pressure in this system were low the pilot could make a landing unaware of an emergency that would leave him without one or both wheel brakes, and without a drag chute. A priority system pressure gage should be provided in the cockpit to give the pilot an indication of priority system pressure.

The landing gear warning light, located in the landing gear control handle, is difficult to see and should be replaced with a red light located on the instrument panel. In the present installation, the light comes on when the gear is not down and locked at altitudes below 10,000 feet and at airspeeds below 275-300 knots IAS. When flying adverse weather holding patterns, the light is on during the preparatory phases to landing. Besides being annoying to the pilot, the early actuation of the light reduces its value as a warning device. Sensing should be changed so that the light is actuated at 200 knots IAS and below.

The emergency landing gear extension control must usually be held in the aft (extend) position. As a consequence, the pilot's left hand cannot be used for operation of the power lever. If the emergency landing gear control slips forward it could result in partial retraction of the landing gear or possible rupture of the secondary hydraulic system, depending on the normal gear control handle position. Action should be taken to insure that all F-102 aircraft are provided with a positive catch to hold the emergency landing gear control in the aft position after operation by the pilot.

Unintentional jettisoning of the drag chute has occurred because of lack of a sufficiently positive detent or catch to hold the drag chute control handle in the "DEPLOY" position. The drag chute control handle mechanism should be modified to insure positive engagement and retention in the aft position.

Arrangement of the instrument panel is considered satisfactory and the layout appears optimized for the limited space available. There are several instrument deficiencies which should be corrected, however.

The turn and slip indicator gives erroneous indication when the aircraft is being rotated in a roll maneuver, or when rolling into or out of a turn. The indications are often opposite to the direction of the turn being initiated. The indications are correct only after a steady state turn has been established and is being maintained. The installation should be changed so that the instrument is oriented along the flight [illegible] placed so [illegible] panel.

A [illegible] accelerometer should be in[illegible] F-102 aircraft.

The primary heading reference of the F-102A is a system composed of a J-2 gyro magnetic compass and the radio magnetic indicator. Indications are extremely poor because of error at the beginning, throughout, and upon completion of the turn. Errors as large as 20 degrees have been noted and it is always difficult to arrive at an exact heading.

The present altimeter has been misread by as much as 10,000 feet during high performance descent. This should be alleviated by installing an altimeter which provides a "barber pole" indication of aircraft altitude below 15,000 feet.

The present selsyn indicator must be read to determine afterburner nozzle position, since the position indications look alike. The indicator could be used more rapidly if the "OPEN" indication were black letters on a white background, and "CLOSE" were white letters on a black background.

The K4B-MM-2 remote attitude indicator has an "OFF" flag which indicates complete AC or DC power failure. Electrical failures in pitch or roll components will give erroneous readings on the instrument but no warning is provided. The attitude indicating system should be modified so that loss of either roll pitch signals will cause the "OFF" flag to appear.

taxiing and ground handling

The taxiing and ground handling characteristics of the F-102A are generally satisfactory. Visibility to the front and sides is satisfactory, but the area aft of the wing tips cannot be seen due to the canopy design. Idle or near idle power is adequate for all taxi operations except initial movement from stand still. Nose wheel steering response is rapid and smooth, except that the high rudder force causes the aircraft to straighten out more quickly than desired during recovery from a short radius turn. The configuration of the nose wheel steering switch and the force required to keep the switch in the "ENGAGE" position result in hand fatigue. A flat-top switch similar to the ignition switch with minimum spring tension would be a better installation.

Braking action during taxi operation is unsatisfactory in that there is little relationship between brake pedal force and braking obtained. Large pedal forces are required to initiate braking action, and a slight increase in force will lock the wheel.

take-off

Initial run-up and take-off are accompanied by a surge of ventilation air, which blows dust in the pilot's face. Since this impairs vision, it is a definite hazard to safe flight and should be corrected as soon as possible.

The controls are trimmed automatically prior to take-off by actuating a push button in the cockpit. This sets the rudder and aileron trim at zero degrees and the elevator to 5 degrees up. The aircraft is aligned with the runway using nose wheel steering. After the engine check at military power is completed the brakes are released with the nose wheel steering engaged. After the first 1 or 2 seconds of acceleration are completed and it is assured that directional control is being maintained, the afterburner is lighted. A distinct increase in acceleration will indicate a successful afterburner light.

It is recommended that nose wheel steering be engaged during the afterburner light. A severe directional change when the afterburner is lit indicates a malfunction of the afterburner eyelids and requires the afterburner to be shut off, power reduced to idle, and the take-off aborted.

Nose wheel steering should be discontinued when the rudder becomes effective at approximately 70 knots IAS. Differential braking should not be required but could be used cautiously if necessary. The aircraft is easily rotated to take-off attitude at 125 knots IAS using light stick forces with either a forward or aft center of gravity, and becomes airborne at 150 knots.

The elevons should be left near the trim position until 125 knots IAS is obtained, then sufficient force should be applied to rotate the aircraft to the desired attitude. No advantage is obtained by applying up elevon at low speeds. Figure 1, Appendix I shows a time history of a take-off wherein pull forces were maintained throughout most of the take-off roll. Figures 2 and 3 illustrate a take-off where back pressure was applied at the optimum nose wheel lift-off speed. In each instance, the aircraft responded at nearly the same speed. With an aft cg, full up elevon could cause rates of rotation at nose wheel lift-off that would be difficult to control.

The rapid acceleration after becoming airborne requires that a climb be maintained so that the landing gear may be fully retracted prior to reaching the gear limit speed of 220 knots IAS. Trim changes with gear retraction are minor.

Take-off using military power is similar, except for the longer take-off roll, shallower climb angle, and slower acceleration.

A time lag of as much as 2 seconds occurs between release of the pedals and actual release of the brakes. This sometimes results in non-simultaneous release of the brakes, impairing directional control in the take-off roll. The time lag between release of the brake pedals and the actual release of the brakes should be eliminated.

dynamic longitudinal stability

With the pitch damper off the aircraft does not meet MIL-F-8785(ASG) Specifications, which require that oscillations damp to 0.1 amplitude in one cycle. The damping is lowest at supersonic speeds, where oscillations require two to three cycles to damp to 0.1 amplitude and a small residual oscillation persists for a considerable length of time (Figure 7, Appendix I). Small oscillations which damp very slowly are easily induced in normal flying. Damping is much improved at subsonic Mach numbers but is still not considered satisfactory.

With the pitch damper operating longitudinal oscillations damp to 0.1 amplitude in one cycle and residual oscillations are eliminated. The pitch damper is necessary for high speed radar tracking.

A sharp movement of the longitudinal control causes a momentary change in normal acceleration opposite to the expected direction. This is caused by a change in lift coefficient due to elevon deflection before the aircraft responds to the pitching moment created.

Time histories of dynamic longitudinal oscillations presented in Figures 4 through 17 of Appendix I are summarized in the following table.

DYNAMIC LONGITUDINAL STABILITY

Fig.	Altitude	Mach No.	[illegible]	Pitch Damper	Yaw Damper	No. Cycles to Damp to 0.1 Amp.	Residual Oscillation
4	45,000	1.07	Combat	on	on	1.5	no
5	45,000	0.9	Cruise	on	on	1.0	no
6	35,000	1.1	Combat	on	on	1.0	no
7	35,000	1.1	Combat	off	on	2.5	yes
8	35,000	1.1	Combat	on	off	1.0	no
9	35,000	1.1	Combat	off	off	2.5	yes
10	35,000	0.9	Cruise	on	on	1.0	no
11	35,000	0.9	Cruise	off	off	1.5	no
12	35,000	0.7	Cruise	on	on	1.0	no
13	35,000	0.7	Cruise	off	off	2.0	no
14	10,000	0.7	Cruise	on	on	0.5	no
15	10,000	0.5	Cruise	on	on	0.5	no
16	10,000	0.33	Pwr Approach	on	on	0.5	no
17	10,000	0.33	Pwr Approach	off	on	1.0	no

static longitudinal stability

The static longitudinal characteristics of the F-102A are similar to those of other transonic and supersonic aircraft. Stability is positive at all altitudes in the low subsonic speed region, but is nearly neutral between 0.75 and 0.9 Mach number, as shown in Figure 24. The trim change associated with transonic flight begins at approximately 0.75 Mach number at low altitudes, and at 0.85 to 0.9 Mach number at high altitudes. A pull force of approximately 8 pounds is required to maintain level flight while accelerating from 0.9 to 1.1 Mach number at 35,000 feet with a center of gravity of 26.9 percent MAC. With an aft center of gravity (30.3 percent MAC), the pull force is decreased to 5 pounds, as shown in Figure 22, Appendix I.

The aircraft is equipped with an automatic trim servo to compensate for the trim change in the transonic and supersonic region. The trim servo automatically retrims the elevons up from 0.85 to 1.1 Mach number, which completely eliminates the stick force reversal in this speed region. As shown in Figures 19 and 21, a push force of approximately 12 pounds is required to accelerate from 0.9 to 1.1 Mach number with the trim servo operating.

Although trim servo operation is satisfactory, one minor deficiency was noted. When the trim servo system is engaged supersonically and the aircraft is slowed to minimum flying speed without manually retrimming, the pull forces required decrease at speeds below 0.85 Mach number, (Figure 21, Appendix I). This is not an unsatisfactory condition since the minimum pull forces are in excess of 10 pounds, and in ordinary operation the aircraft would be manually retrimmed. It should also be noted that the operating limits of the trim servo had not been finalized at the time of the Phase IV tests.

The low speed stick-fixed neutral points were found at lift coefficients of 0.20, 0.30, and 0.40. The neutral point was found to vary from 32.8 percent MAC at a lift coefficient of 0.40 to 37.6 percent MAC at a lift coefficient of 0.20. The neutral point at high lift coefficients or low speed appears from extrapolated data to be approximately 2.5 percent MAC aft of the present flight limit of 30.5 percent MAC.

trim changes

The longitudinal trim changes resulting from speed brake operation, afterburner operation, and landing gear extension are small, and the forces required are generally within the control friction band. Lighting the afterburner causes a dynamic oscillation, which damps out quickly without pilot effort. (Figure 26, Appendix I.) No trim change is evident after the oscillation. This oscillation could be caused by non-simultaneous opening of the afterburner eyelids.

Trim changes were determined by trimming the aircraft for level flight, introducing the variable, and maintaining level flight. The change in forces and positions are summarized in the following table.

LONGITUDINAL TRIM CHANGES

Calibrated Airspeed Knots	Alt-Ft.	Landing Gear	Speed Brake	Variable	Elevon Defl.	Long. Stick Force Change
325	35,000	up	In	Sp Brk out	0.7 dn	Negligible
225	20,000	up	In	Gear dn	1.3 dn	5 Lbs. Forward
240	35,000	up	In	A/B on	None	Negligible

maneuvering flight

The maneuvering force gradients are unsatisfactory at the extreme limits of the operational flight envelope.

The force gradient is excessive above 0.9 Mach number at 50,000 feet. As shown in Figure 27, the gradients of 30 pounds per g at 26.9 percent MAC

[illegible] Fig. 27

and 25 pounds per g at 30.1 percent MAC, obtained at 0.97 Mach number greatly exceed the 9 pounds per g maximum permitted by Air Force Specifications. The gradient at supersonic speeds at 35,000 feet is also excessive as may be seen in Figure 29.

The force gradient of only 2.5 pounds per g obtained at 0.72 Mach number and 35,000 feet, as shown in Figure 32, is less than the 3.5 pounds per g minimum allowed by the specification. Maneuvering gradients are satisfactory throughout the remainder of the subsonic flight regime, with the exception of the power approach configuration with a forward cg where the force gradient is excessive.

Force gradient characteristics during sudden pull ups are inconsistent and undesirable under certain conditions. If the stick is displaced and returned to neutral in from 1 to 2 seconds, the stick force per g is sometimes less than that required for steady maneuvering flight, which is contrary to MIL-F-8785 (ASG) requirements. However, if the stick is displaced and returned to neutral in 1 second or less, stick forces exceed steady maneuvering flight gradients. This characteristic is not objectionable. Pull-up and return rates of 3 seconds or over usually result in gradients similar to those obtained in steady accelerations.

A stick-free pitch-up is encountered when making decelerating turns from supersonic trim speeds. Decreasing back forces are required above 2.5g and the aircraft will maintain 3g with the stick free. This condition is encountered with the trim servo on or off, as may be seen in Figures 37 and 38, Appendix I, and results from the combined effect of decelerating through the trim change and the bob weight effect of the yoke stick.

dynamic lateral-directional stability

The airplane meets Air Force requirements in dynamic lateral-directional damping, since oscillations damp to 0.10 amplitude or less in 1 cycle with the yaw damper operating. No significant difference in the damping is noted with addition of either the turn coordinator or sideslip feedback. A barely perceptible high frequency oscillation accompanied by light buffet persists in the speed region from .96 to 1.0 true Mach number with the yaw damper operating. This is annoying but is not considered unsatisfactory since it occurs in a seldom-used region of flight.

Little damping occurs with the yaw damper off. As shown in Figures 46 and 50, the aircraft oscillates at a frequency of approximately ½ cycle per second, requiring 1½ cycles to damp to ½ amplitude.

The production yoke stick produces an undesirable effect on the dynamic lateral-directional characteristics. A sharp lateral disturbance of the control stick causes a chatter in the control system, and an elevon oscillation of approximately 3 cycles per second. This persists for 3 to 4 cycles, and is followed by a slower oscillation in the natural frequency of the aircraft. This is caused by the bob weight effect of the stick coupled with the dynamics of the control system and the airplane. At 1.1 Mach number the mass of the stick is sufficiently affected by the lateral loads to produce a continuous lateral-directional oscillation with the stick free. This effect may be seen in Figure 43, Appendix I.

At Mach numbers above 0.9 the aircraft exhibits a small amplitude directional oscillation in response to a longitudinal pulse. As shown on the dynamic longitudinal time histories, Figures 4 through 17, Appendix I, these oscillations are eliminated by the yaw damper.

static directional stability

Although the aircraft exhibits positive static directional stability in all configurations and all altitudes, directional control does not meet the specification requirements since the rudder force gradient becomes neutral and even negative at large deflections such as that required for 10 degrees of sideslip in the power approach configuration. This is illustrated in Figure 65, Appendix I. Control is satisfactory to the pilot; however, since positive gradients are encountered in all realistic realms of flight.

The rudder force required for a given rudder deflection rises rapidly with Mach number due to a Mach sensing device in the rudder feel system.

The aircraft meets the Air Force Specification under all conditions with the sideslip feedback operating. Nearly twice the force is required for a given rudder deflection with the feedback operating as would be required without it. (Figure 54.)

lateral control

The lateral control system is unsatisfactory in that wings-level flight is difficult to maintain. With the aircraft trimmed for level flight a single maneuver requiring lateral control will cause the aircraft to be out of trim upon completion of the maneuver. This problem is caused by a combination of high sensitivity of the control near trim and high breakout forces from trim. The breakout friction and sensitivity at trim should be reduced, and in addition positive centering must be provided for the yoke stick. ✓

The roll rate with maximum aileron deflection meets Air Force Specifications in the cruise configuration. Roll rates above 200 degrees per second are attainable at 10,000 feet with a helix angle of 0.086.

The full lateral control capabilities of the aircraft cannot easily be realized, however, because the control stick hits the pilot's knees at three-fourths deflection (10.5 to 12 degrees). This reduces the roll capability at 10,000 feet to only 160 degrees per second with a helix angle of 0.066. At 45,000 feet the helix angle is further reduced to .06. The aircraft is still capable of meeting the requirement calling for 100 degrees of bank change in 1 second after control initiation at altitudes below 20,000 feet.

At speeds below 300 knots, adverse yaw tends to slow the roll, and at speeds above 300 knots complimentary yaw accelerates the maneuver. The sideslip feedback and turn coordinator decrease the yaw in all cases. The yaw damper, with the other stability augmentation devices off, initially tends to create adverse yaw in all rolls. The roll capabilities shown in Figures 66 through 69 are those obtainable within 120 degrees of bank change with turn coordinator and sideslip feedback operating. Rolls through 360 degrees with half aileron showed no increase in rate beyond 120 degrees of bank change.

The lateral control does not meet Air Force Specifications in the power approach configuration. The helix angle of .03 obtainable is 40 percent below the required .05, although the peak roll rates of 35 degrees per second commensurate with the .03 helix angle obtainable at 180 knots IAS are adequate under adverse landing conditions. The principle reason for the low helix angle is a 0.3 second delay between control initiation and response of the aircraft. Only about 12 degrees of aileron travel, rather than the full 14 degree capability of the control system, is available because of insufficient room for full stick throw. al

The forces required to roll the aircraft are acceptable in all configurations. The bungee system used in the lateral control system is unaffected by inflight air loads and produces a force gradient of approximately 1 pound per degree of total differential elevon.

inertial coupling

Figs 71 & 72

Limited tests were performed to determine the inertial coupling tendencies within the limits specified by the manufacturer. The aircraft was rolled through 360 degrees at 80 degrees per second with the sideslip feedback off and 120 degrees per second with the feedback on.

with the s/s feedback off

At airspeeds above 305 knots CAS and at all supersonic speeds the aircraft exhibits complimentary yaw during elevator fixed rolls. The complimentary yaw is increased by down elevator. An up elevator input causes decreased complimentary yaw, or in some cases, adverse yaw. At speeds below 305 knots, the aircraft exhibits adverse yaw when rolled without elevator input. With a positive elevator input, the adverse yaw is increased; with a negative elevator input it is decreased. In no case were the yaw angles encountered dangerous. The roll characteristics at all speeds with the sideslip feedback operating are similar to those above 305 knots with the feedback off.

This [illegible] down.

Further discussion is included in AFFTC-TN-55-24, "Roll Evaluation of the F-102A" and AFFTC-TN-56-15, "Roll Evaluation of the F-102A with an Enlarged Vertical Stabilizer."

trim rates and trim control

The longitudinal and directional trim rates and response are satisfactory but the effectiveness of the lateral trim control is considered unsatisfactory because of the excessive delay between initiation of lateral trimming action and the accomplishment of the desired amount of trim. This problem is amplified by the high sensitivity to lateral control movement. The steady state lateral trim rate in flight is approximately 40 percent higher than either the directional or longitudinal trim rate.

The trim rates were determined on the ground and inflight by trimming the aircraft through the maximum range available. In every case, the values shown are average values over the entire range. Both the longitudinal and directional controls have feel force systems that are functions of airspeed. Since they affect the forces being trimmed out, they also affect the trim rate on these controls. The lateral control system feel force is obtained from a bungee system only and is not affected by airspeed.

Control	Rate of Trim-deg/sec Ground	Rate of Trim-deg/sec In flight	In-flight Airspeeds Knots, IAS
Directional	1.55	0.72	200
Lateral	1.45	1.05	300
Longitudinal	1.32	0.85 & 0.75	200 & 265

stalls and low speed handling characteristics

The low speed handling characteristics are good. The aircraft exhibits buffet at speeds well above the point where dangerous sink rates develop. As the angle of attack is increased, directional control is gradually lost as the rudder is blanked out by the wing and fuselage. Continuous use of rudder and aileron is required to maintain straight and level flight below 120-125 knots IAS. Longitudinal control remains good at speeds near stall and recovery is easily effected by reducing the back pressure on the stick. The aircraft was not flown at speeds below 110 knots since spin tests had not been demonstrated.

approach and landing

The recommended initial approach speed is 300 to 350 knots IAS. The speed brake is extended and power reduced during the break turn to arrive on the downwind leg at 220 knots. Landing gear is lowered prior to turning on the base leg. After the gear is lowered the speed brake may be closed to reduce buffeting. A speed of 200 knots is maintained on the base leg, and the turn to final is started at 190 knots and completed at 180 knots. The optimum approach speed is 160 knots with power being used to maintain airspeed, the over-the-fence speed is 150 knots, and the touchdown is made at 135 knots.

The drag chute is deployed immediately after touchdown while the nose wheel is still off the ground. The drag chute is in line with the cg so that no pitching moment is created by deployment. Directional control remains good upon deployment, and yaw due to crosswind is slight. The nose wheel is lowered at 100 to 110 knots IAS by use of the longitudinal control.

Braking action during landing is unsatisfactory. Caution must be used when braking due to the heavy pedal forces and lack of feel in the brake system. Heavy braking will cause skidding, and with the excessive lag in brake release a blown tire will usually result before the skid can be stopped. Positive and effective braking action requires that anti-skid devices be incorporated in the brake system.

Seems to have become a habit too.

Drag chute deployment is not reliable. Fifteen failures occurred in 100 deployments accomplished during Phase IV and Phase VI testing. The drag chute deployment time is satisfactory, but an additional second is required for deployment if the landing is made with the speed brakes closed.

engine operation

Continuous engine stalls occur with a combination of low indicated airspeed and low power setting at high altitude. During one flight at 45,000 feet altitude the throttle was slowly retarded to idle, causing steady compressor stalls. Throttle movement had no effect upon recovery and the aircraft was dived at 0.9 Mach number to prevent the exhaust temperature from exceeding the maximum limits. Dive recovery was made just below 30,000 feet and, as airspeed decreased, slow throttle advancement permitted engine acceleration through the stall region. This incident is similar to several encountered during the Phase VI F-102A program. One of the Phase VI aircraft was provided with a modified bleed valve schedule that eliminated this difficulty. All aircraft should be equipped with an optimum bleed valve schedule as soon as possible.

On several high final approach landings the throttle was kept in idle. When attempting to add power for the last part of the final approach several compressor stalls were experienced which delayed engine acceleration approximately 5 seconds. This was accompanied by a loss in airspeed of 5 to 10 knots. The need for stall-free engine operation at low altitude, low airspeed and low power setting is mandatory from a safety standpoint.

The test aircraft did not have the extended ramp on the inlet duct, and very annoying duct rumble and buzz were experienced above 1.2 Mach number.

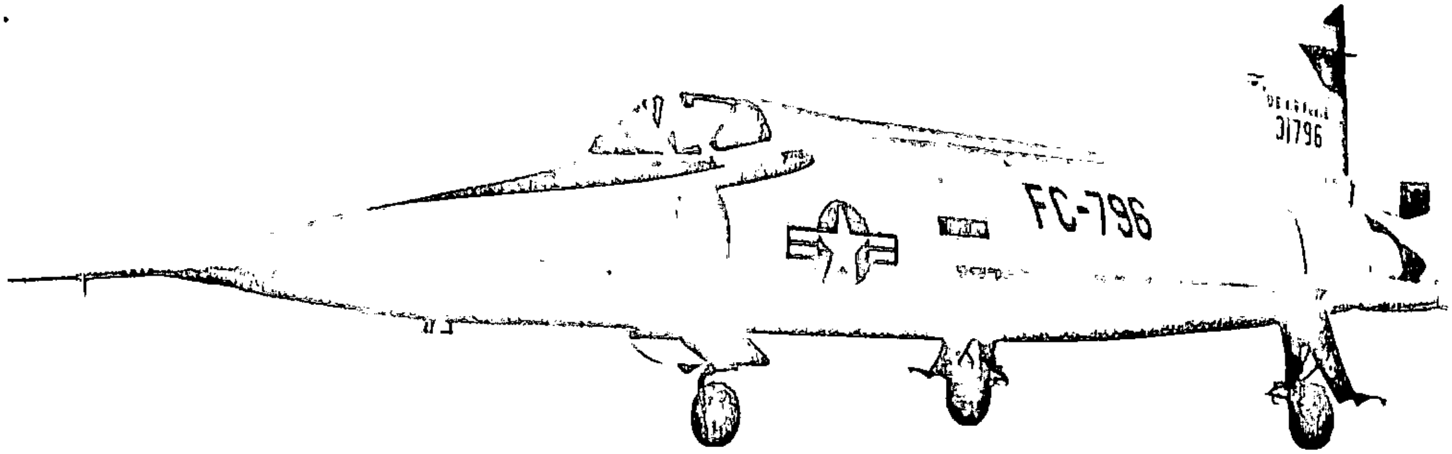
control friction

The aileron and rudder break-out forces are excessive and exceed the requirements of Air Force Specification MIL-F-8785 (ASG). The static break-out friction for the lateral control system at the trim position exceeds the specification by 1 pound, or 50 percent. Both lateral and directional break-out friction are much less at other than trim positions.

The static break-out forces as measured after flight No. 7 are as follows:

Control	Break-out Force-lbs.	Maximum Alowable Break-out Force MIL-F-8785 (ASG)
Longitudinal	3	3
Lateral	3	2
Directional	11	7

The static friction calibrations were obtained by moving the controls in slow, short movements so that the break-out force was measured with each movement (see Figures 75 through 77). No in flight control friction data was obtained.



CONCLUSIONS

The stability and control characteristics of the F-102A with stability augmentors operating are considered satisfactory for an all-weather interceptor. Without the stability augmentors operating the aircraft could be flown safely, but would probably be unable to successfully accomplish an intercept mission.

Several deficiencies which are annoying but do not seriously decrease the combat effectiveness were encountered.

1. The yoke stick prevents effective use of full aileron.
2. The bob weight effect of the yoke stick produces undesirable dynamic effects.
3. Lateral and directional control break-out friction is excessive and lateral control sensitivity near trim is too high.
4. Longitudinal control forces are excessive above 50,000 feet at Mach numbers greater than 0.9 and in the power approach configuration with a forward cg, but forces are too light below 35,000 feet at Mach numbers below 0.7 in the cruise configuration.

5. The maneuvering force gradients are unsatisfactory at the extreme limits of the operational flight envelope.
6. Forces required to obtain a given value of normal acceleration during abrupt pull-ups vary with the speed of the control input and return to neutral.

RECOMMENDATIONS

It is recommended that the following items be corrected as soon as possible:

1. The throw of the lateral control system be modified to allow the pilot to obtain full aileron.
2. The rudder trim switch be relocated to the left console for easier operation.
3. The belt length or its installation be changed to insure sufficient adjustment range for all pilots.
4. A priority system pneumatic pressure gage be installed in the cockpit.
5. The lateral trim control be modified to give less lag between control initiation and aircraft response.
6. The lateral break-out friction and control sensitivity near trim be reduced.
7. A positive detent or catch be provided to hold the emergency gear extension handle in place when activated.
8. Canopy cinch down operation be made more reliable.
9. The fuel quantity indication be modified to allow the pilot to read both the total fuel and the quantity in each wing.
10. The time delay between release of the brake pedal and actual release of the brakes be eliminated.
11. The present altimeter be replaced with an altimeter which provides a "barber pole" indication of aircraft altitude below 15,000 feet.
12. Steps be taken to eliminate engine compressor stalls.
13. The extended ramp on the inlet duct be installed on all aircraft.
14. An anti-skid device be installed in the brake system.
15. The attitude indicating system be modified so that loss of either roll or pitch signals will cause the "OFF" flag to appear.

The following items are less critical but should be accomplished as soon as feasible.

1. A bob weight be added to the longitudinal control system to offset the weight of the yoke stick.
2. The oxygen hose length be increased 6 inches.

3. The drag chute control mechanism should be modified to insure positive engagement and retention in the aft "DEPLOY" position.

4. A plainly visible accelerometer should be installed in all F-102 aircraft.

5. The surge of ventilation air on initial engine run-up and take-off should be eliminated.

6. The angle at which the ladder for cockpit entrance hangs be modified so that the pilot moves forward toward the airplane.

7. A means be provided to positively lock the canopy in any position during taxi operations.

8. The "T" on the canopy locking handle be oriented horizontally.

9. The control handles for the drag chute, emergency gear extension and canopy locking be identified by placarding directly on the handles.

10. The flight mode switch be called "STABILITY AUGMENTATORS" and the labeling be changed from "DIRECT-MANUAL" and "MANUAL" to "ON" and "OFF."

11. The anti-ice switch be replaced by one similar to the cabin pressure switch in this aircraft.

12. The number of rheostat switches used to control cockpit lighting be reduced from five to three.

13. Small indicator lights should be installed on the fuel selector panel to show open positions of the fuel shut-off valves.

14. The windshield anti-reflection curtain should be replaced with a durable fixed divider.

15. The landing gear warning light should be relocated with a red light located on the instrument panel.

16. Sensing should be changed on the landing gear warning light so that it operates at altitudes below 10,000 feet and airspeeds below 200 knots IAS.

17. The present primary heading reference should be replaced with a system which provides accurate heading indications during turns.

18. The turn and slip indicator should be aligned so the instrument longitudinal axis is parallel to the aircraft flight path longitudinal axis. ?

19. The afterburner selsyn indicator be changed so the "OPEN" indication will employ black letters on a white background and the "CLOSE" position white letters on a black background.

20. The present nose wheel steering switch be replaced with a flat-top switch similar to the ignition switch.

21. The rudder break-out friction be reduced.

APPENDIX II

general aircraft information

WING:

area	695.05 sq. ft.
span	38 ft. 1.6 in.
MAC	23 ft. 9.1 in.
aspect ratio	2.189
airfoil section	NACA 0004-65 (mod)
incidence	0°
dihedral	0°
leading edge sweepback	60°
trailing edge sweep forward	5°
camber	7.5 percent C

VERTICAL TAIL: ([illegible])

area	68.33 sq. ft.
span	8.67 ft.
MAC	10.513 ft.
aspect ratio	1.10
airfoil	NACA 0004-65 (mod)
leading edge sweepback	60°
trailing edge sweep forward	5.0°

FUSELAGE:

length (including boom)	68 ft. 3.3 in.
wheel base	23 ft. 3.9 in.
tread (main wheels)	14 ft. 6.0 in.

ELEVONS:

area	68 sq. ft.
span	12 ft. 10.75 in.
deflection	34° −19°

RUDDER:

area	7.43 sq. ft.
deflection	25°

FIN:

area	60.90 sq. ft.

flight limitations

ENGINE:

Maximum power	1 minute on ground 15 minutes inflight
Military power	5 minutes on ground 30 minutes inflight
exhaust gas temperature	630 °C below 35,000 ft.
limit	660 °C above 35,000 ft.

AIRSPEED:

clean	510 knots 1.25 Mach number
gear extended	220 knots
drag chute	160 knots
canopy open	30 knots
ram air turbine	345 knots

LOAD FACTORS:

normal	5.5 g −2.4 g
lateral	1.3 g

Aircraft is restricted from inverted flight and to 3 seconds of zero or negative g flight.

ROLL LIMITATIONS:

Maximum roll rate of 80° second for continuous roll maneuvers with sideslip feedback off.

Maximum rate of 120° second for continuous roll maneuvers with sideslip feedback on.

Unrestricted in roll rate in the top two quadrants.

Uncoordinated maneuvers are prohibited.

miscellaneous

Armament bay doors closed in flight.
Pilot assist system deactivated.
Inlet duct ramps not installed.
Pitot heater inoperative.
No instrument panel lighting.

flight control systems

Longitudinal and lateral control is accomplished by movement of the elevons; in parallel motion as elevators, differentially as ailerons or in any combination of the two motions. Conventional stick movements are employed for elevon control. These control surfaces are actuated by irreversible full hydraulic power control systems which include artificial feel, gyro-stabilized yaw and pitch damper systems and trimming devices. Pilot control motion mechanically meters hydraulic pressure to the control surface actuating cylinders. The control surfaces move only while the controls are being moved. When pilot control motion stops, a mechanical follow-up mechanism shuts off the control valve.

A conventional rudder affords directional control. The rudder surface is actuated by an irreversible full hydraulic power system including a gyro stabilized yaw damper system, a trimming device, a turn coordinator and sideslip feedback. The hydraulic power system functions in the same manner as described above.

Elevator feel force is made proportional to control displacement by a variable linkage. A pneumatic cylinder, modulated by dynamic pressure, provides a variation with airspeed. A trim servo automatic trimming device that senses Mach number provides positive control in the unstable transonic speed range. Rudder feel is obtained from the combined action of an overload control spring assembly and a piston type feel cylinder which is loaded by amplified dynamic pressure. This system limits the lateral acceleration to a safe maximum, yet provides adequate travel for maneuverability. Aileron feel is made directly proportional to control displacement by a simple spring device.

Trim is accomplished electrically by displacing the control surface positions and in turn repositioning the neutral point of the cockpit controls.

fuel system

All fuel is carried in right and left wing tank systems. Each system consists of three integral tanks. Except for a common, single-point refuel-defuel fitting the two wing systems are independent of each other, and feed fuel to the engine through a common supply line either simultaneously or separately. Tank pressurization forces fuel from the aft outboard tank to the forward tank and then to the aft inboard tank where the booster pump is located. Total fuel capacity is 1,110 gallons, 1,084 gallons of which is usable fuel.

power plant

A Pratt and Whitney J57-P-23 turbojet engine with afterburner (flap type nozzle, single cylinder, sheet metal flap) is installed in this aircraft. The engine is rated at 16,000 pounds sea level static thrust with afterburning and 9,880 pounds non-afterburning.

weight and balance

The aircraft was ballasted prior to initiation of the program to a forward center of gravity at combat weight. The gross weight at engine start was 27,500 pounds including pilot, fuel and oil. The center of gravity varied with fuel consumption from 28.65 percent MAC at engine start to 23.5 percent MAC when the fuel load was expended. During the test program the aircraft was reballasted to 27,400 pounds with a center of gravity of 31.7 percent MAC at engine start. With this loading it was necessary to burn 1200 pounds of fuel to arrive at a center of gravity of 30.5 percent at take-off. The fuel expended center of gravity was 27.5 percent MAC. The aircraft was finally ballasted to a gross weight of 27,600 pounds with a mid center of gravity. The center of gravity was 29.8 percent MAC at engine start and 25.1 percent MAC with the fuel expended.

test instrumentation

Pilot's Panel:
- Angle of Attack
- Angle of Yaw
- Airplane Attitude
- Indicated Airspeed
- Indicated Mach Number
- Pressure Altitude
- Rudder Surface Position
- Left Elevon Position
- Right Elevon Position
- Normal Acceleration
- Rate of Roll
- Total Fuel
- Engine rpm (High
  Pressure Rotor)
- Turbine Exit Temperature
- Afterburner Nozzle Position
- Correlation Counter
- Outside Air Temperature
- Time of Day
- Turn and Bank Indicator
- Directional Heading
- Fuel Flow Totalizer

Photo Panel:
- Indicated Airspeed
- Pressure Altitude
- Airplane Attitude
- Outside Air Temperature
- Normal Acceleration
- High Pressure Rotor rpm
- Low Pressure Rotor rpm
- Turbine Discharge
  Total Temperature
- Time of Day
- Correlation Counter
- Lateral Trim Position
- Longitudinal Trim Position
- Directional Trim Position
- Fuel Flow Totalizer
- Afterburner Nozzle Position
- Turbine Discharge Pressure
- Directional Heading

Oscillograph:
- Angle of Attack
- Angle of Yaw
- Longitudinal Stick Force
- Longitudinal Stick Position
- Lateral Stick Force
- Lateral Stick Position
- Rudder Surface Position
- Rudder Pedal Force
- Left Elevon Position
- Right Elevon Position
- Normal Acceleration
- Lateral Acceleration
- Free Stream
  Differential Pressure
- Free Stream
  Static Pressure
- Rate of Roll
- Rate of Yaw
- Rate of Pitch
- Angle of Bank
- Angle of Pitch
- Instrumentation
  Voltage Reference
- Camera Pulse Interval

APPENDIX I

The elevon data presented in these plots may be in error by as much as plus or minus one-half degree when compared with data from a different flight. These errors result from zero shifts, which were impossible to detect due to the differential action of the elevons. ? The data on any single maneuver should have an accuracy of approximately plus or minus one-tenth degree.

Elevator angles are taken as the average of the two elevon positions. The terms elevator and average elevon are used interchangeably throughout the report.

Aileron angles are taken as the difference between the two elevon positions. The terms aileron and differential elevon are used interchangeably throughout the report.

stability curves

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FIGURE NO. 1

TAKE-OFF TIME HISTORY

F 102A USAF NO. 53-1796

WEIGHT 27500 LBS.

C.G. 28.5 PERCENT M.A.C.

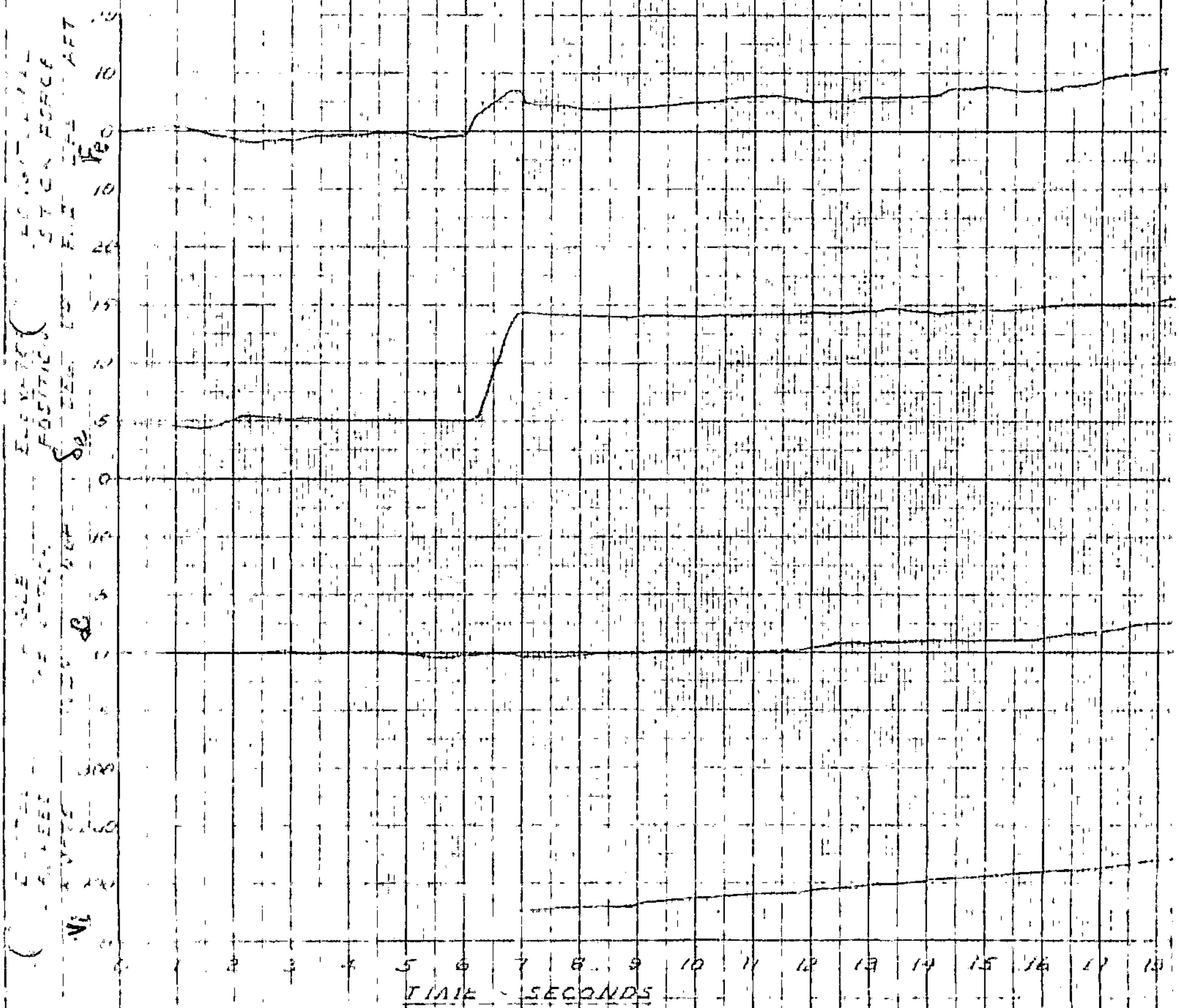


Fig No. 1 (cont.)

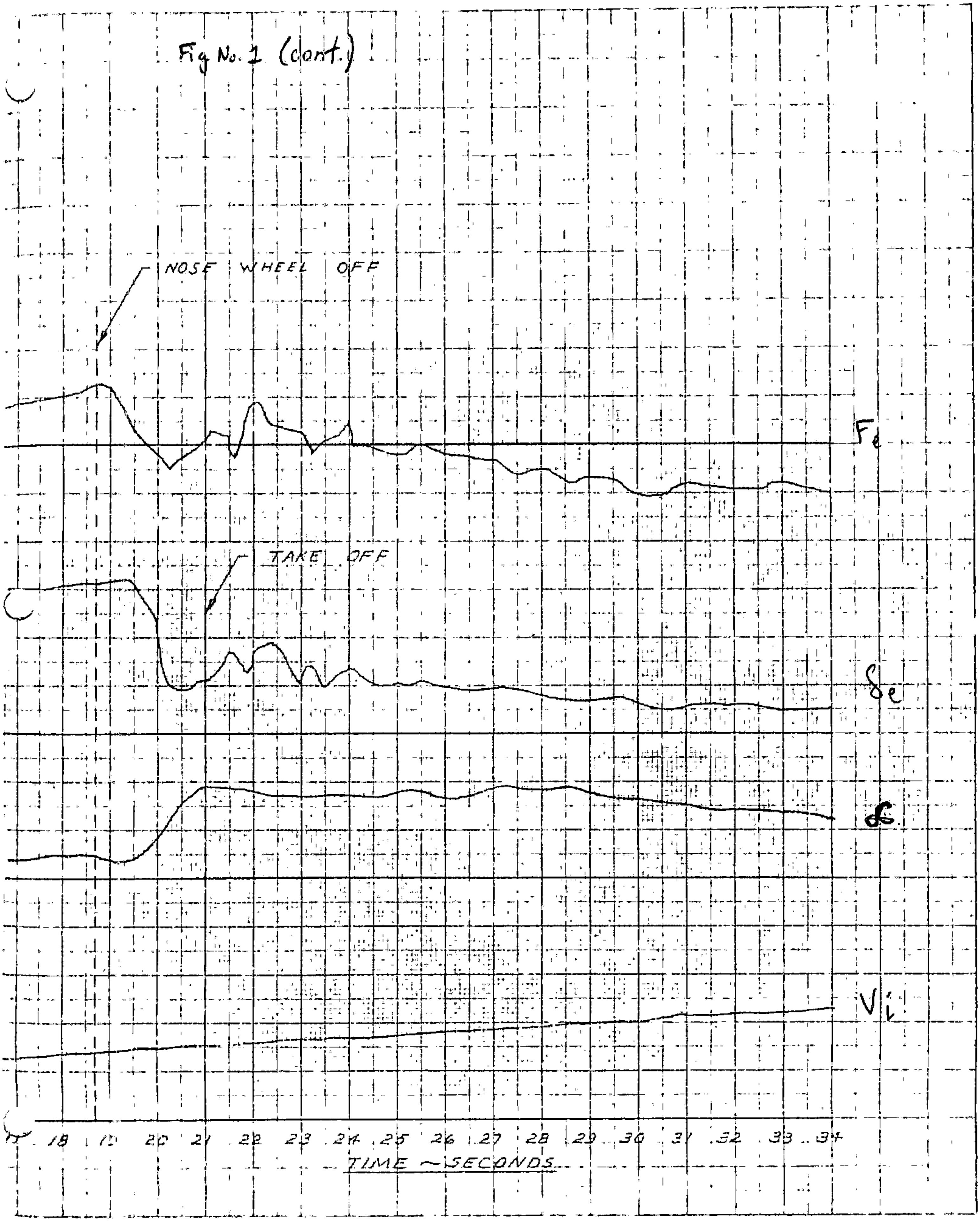
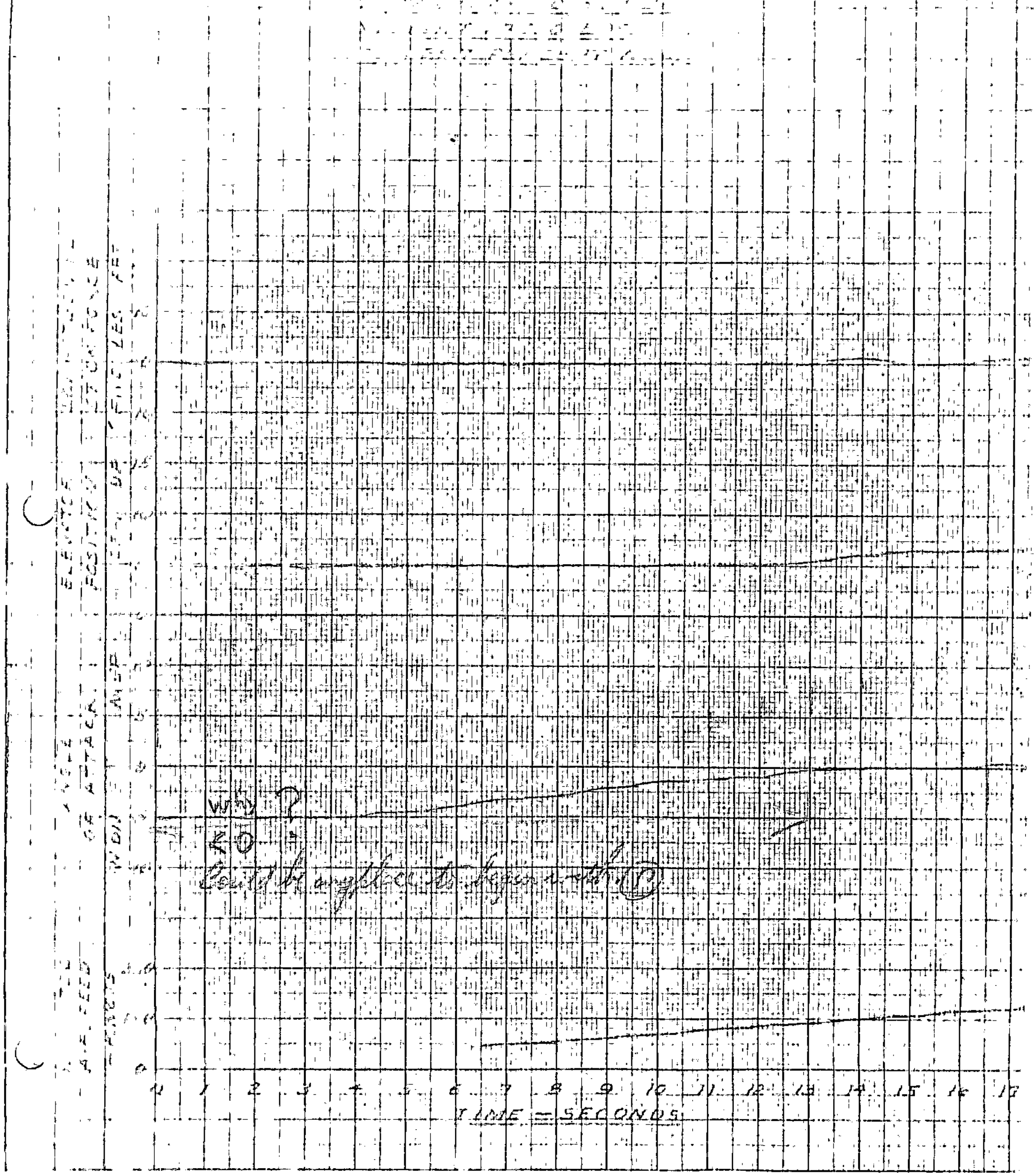

NOSE WHEEL OFF
TAKE OFF
F_e
δ_e
α
V_i
17
18
19
20
21
22
23
24
25
26
27
28
29
30
31
32
33
34
TIME ~ SECONDS

FIGURE NO. 2



why ?

← 0

Could be [illegible] to begin with

Fig. No. 2 (cont.)

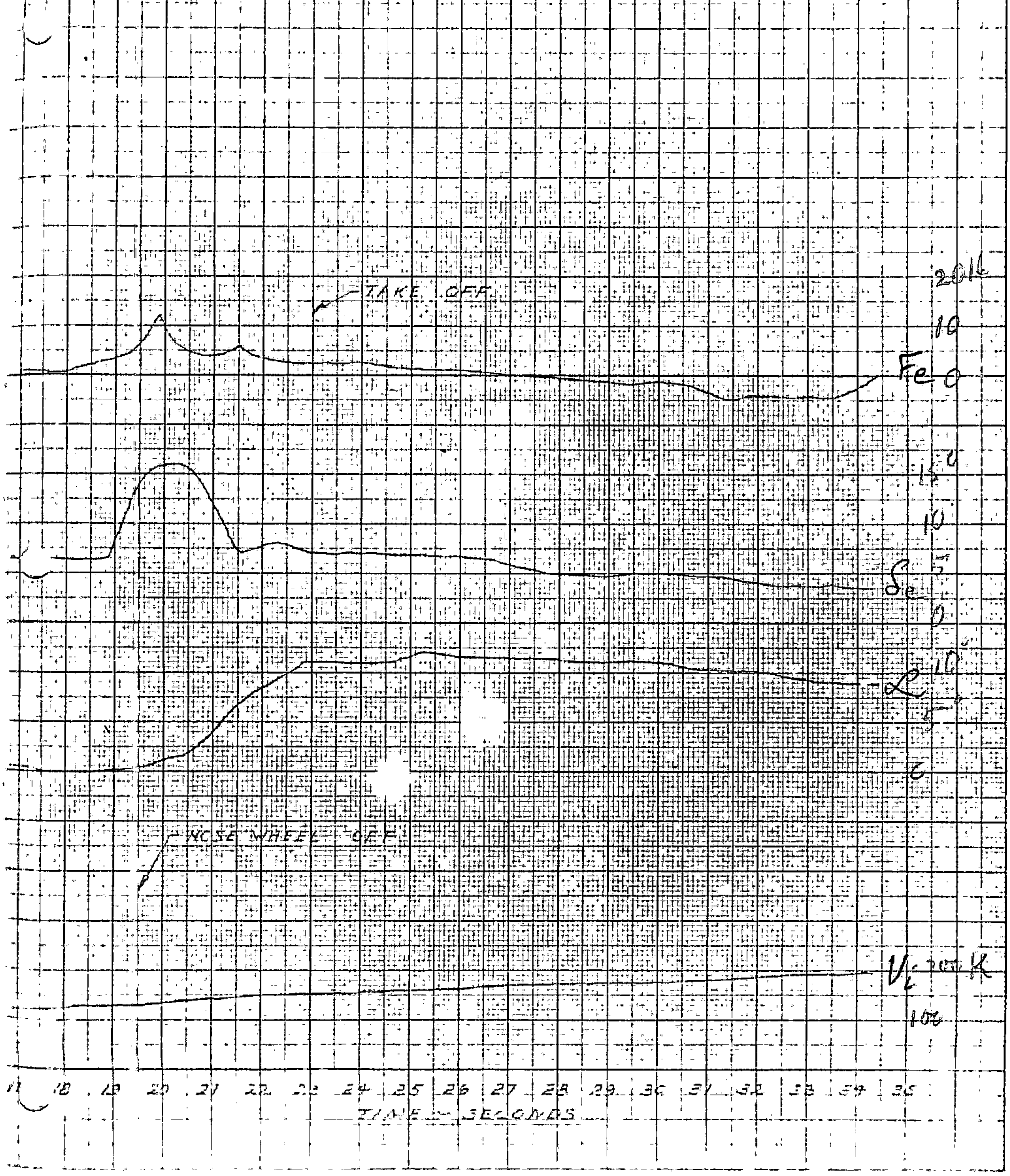

TAKE OFF
20 lb
10
Fe 0
15°
10
5
δ_e
0
10°
α
5
0
NOSE WHEEL OFF
V_i 200 K
100
17
18
19
20
21
22
23
24
25
26
27
28
29
30
31
32
33
34
35
TIME — SECONDS

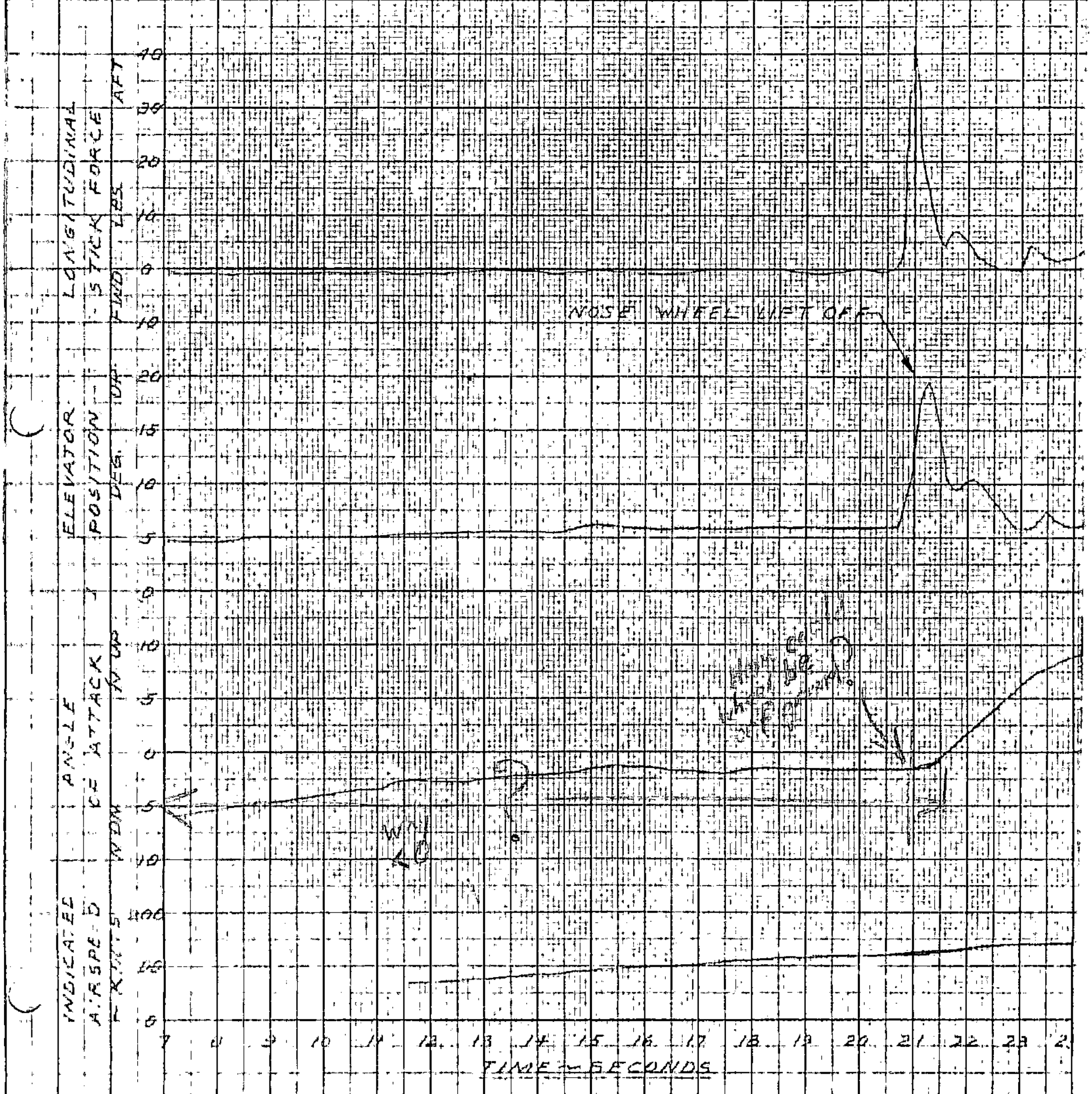
FIGURE NO. 3

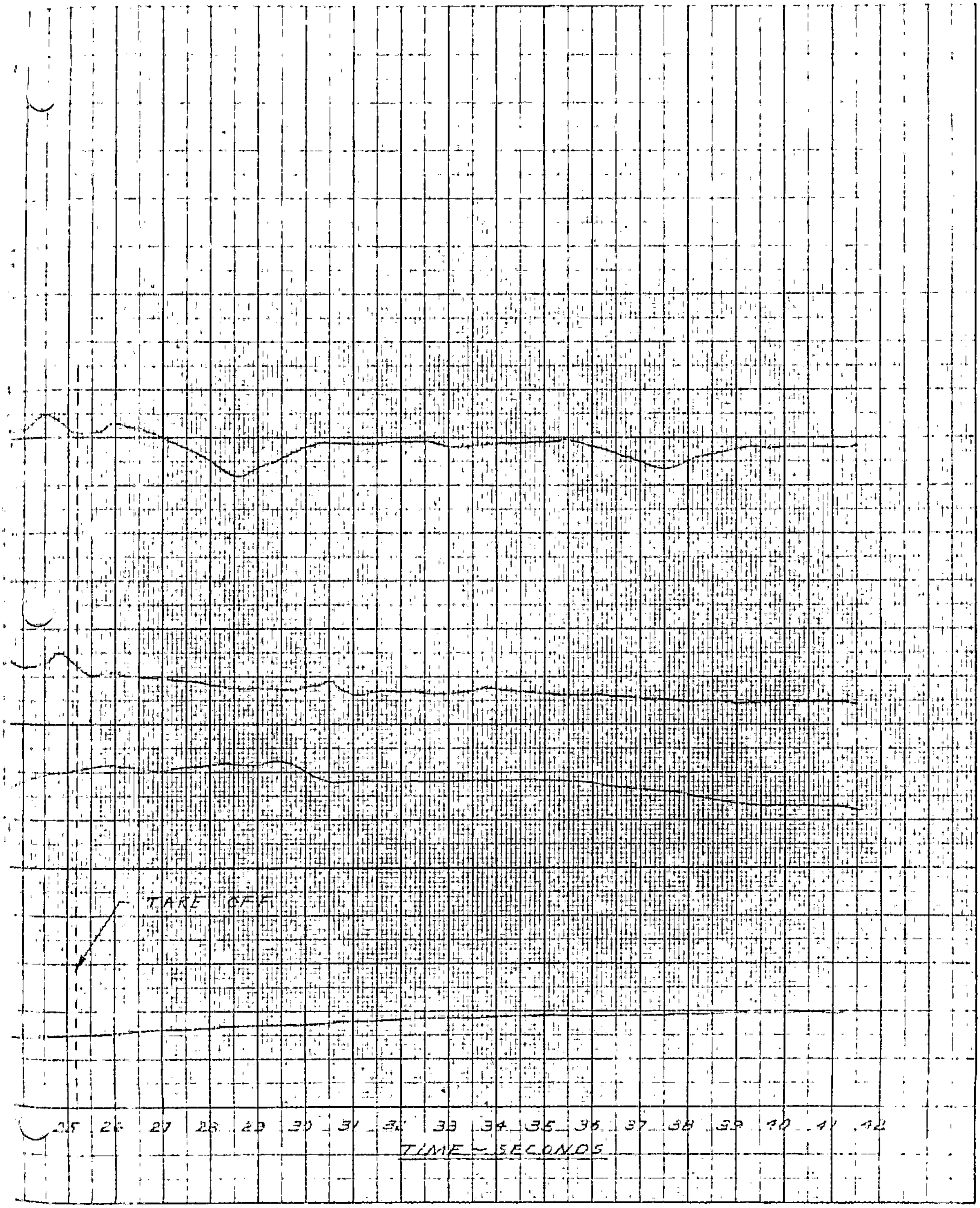
TAKE-OFF TIME HISTORY

F-102A USAF NO. 53-1796

WEIGHT 26200 LBS.

C.G. 30.5 PERCENT MAC.




TAKE OFF
25
26
27
28
29
30
31
32
33
34
35
36
37
38
39
40
41
42
TIME ~ SECONDS

Dynamic Long Stab Config

Alt	C.G	Mach	Pitch Damper	Yaw Damper
45	27.0	1.07	On	On
45	27.1	0.9	On	On
35	27.1	1.1	On	On
35	27.1	1.1	off	On
35	26.9	1.1	On	off
35	26.9	1.1	off	off
35	27.0	0.9	On	On
35	27.0	0.9	off	off
35	27.0	0.7	On	On
35	27.0	0.7	off	off
10	27.0	0.7	On	On
~~10~~	~~27.0~~	~~0.7~~	~~On~~	~~On~~
10	25.7	0.33	On	On
10	25.7	0.33	off	On

Figure No. 4

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration — Pitch Damper On

45,000 Altitude — Mach No. 1.07

24200 Lbs. Weight — Yaw Damper On

C. G. 27.0 percent MAC

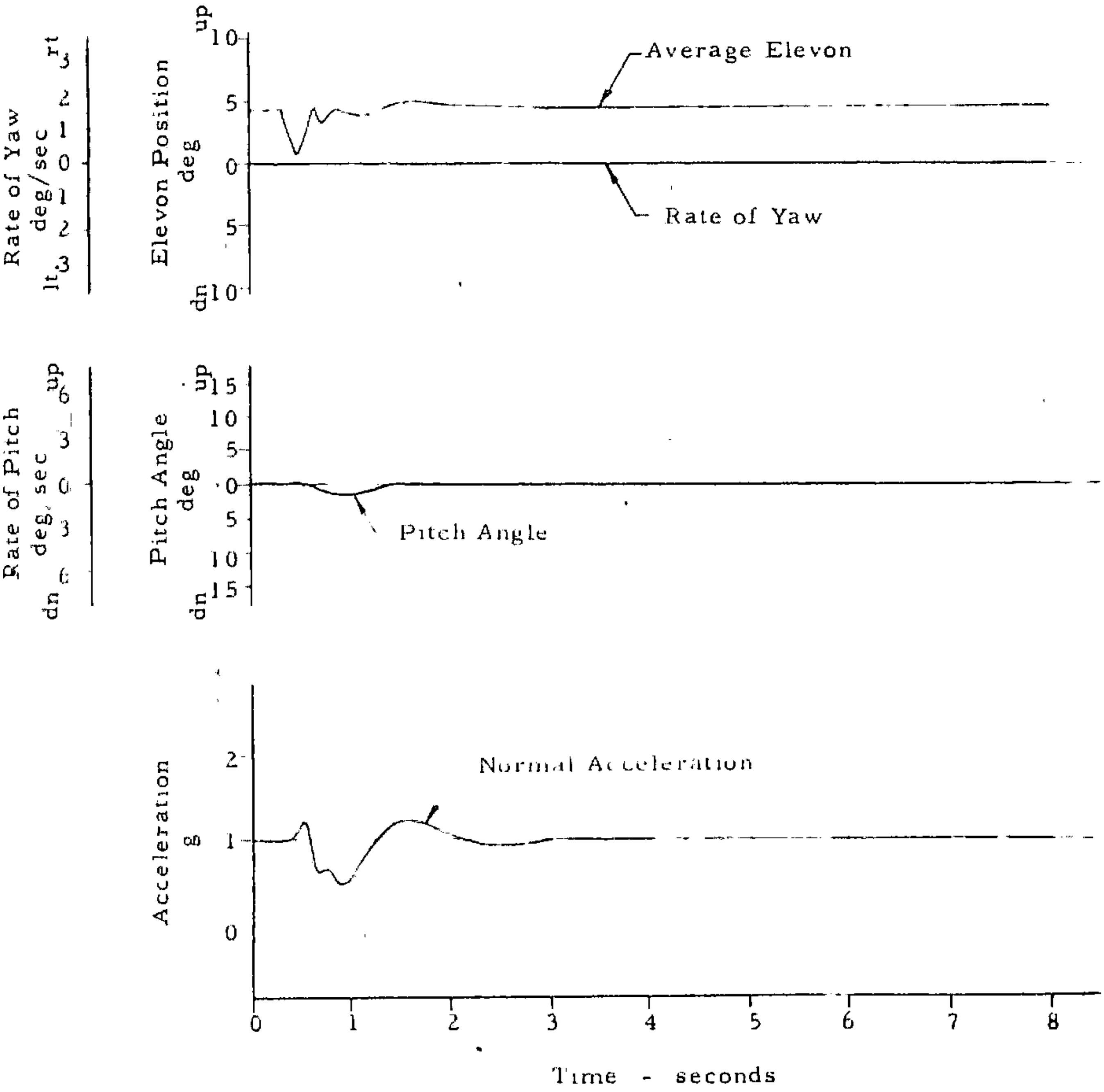


Figure No. 5

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
45,000 Altitude
23600 Lbs. Weight

Pitch Damper On
Mach No. 0.9
Yaw Damper On

C. G. 27.1 percent MAC

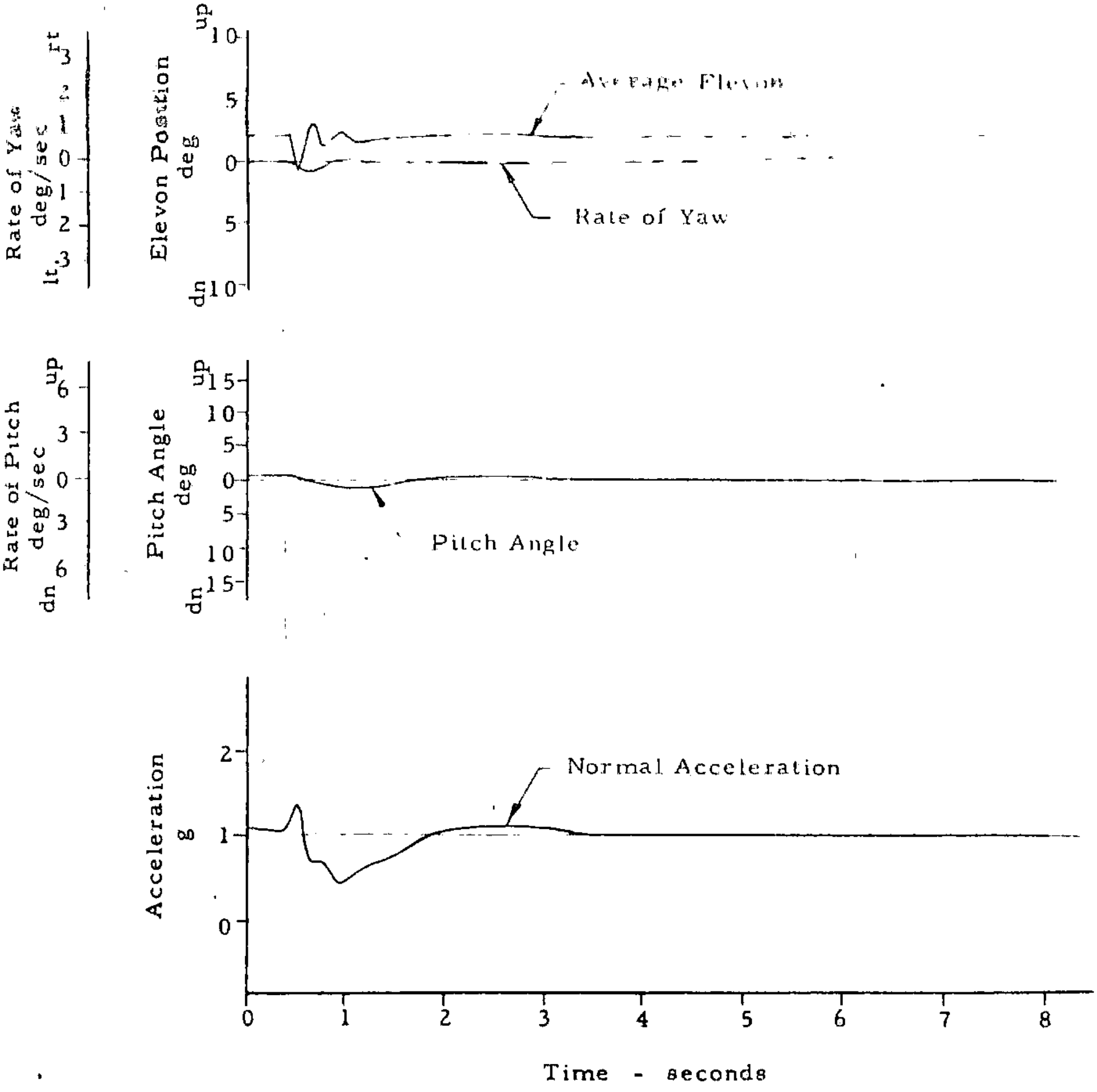


Figure No. 6

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
23400 Lbs. Weight

Pitch Damper On
Mach No. 1.1
Yaw Damper On

C.G. 27.1 percent MAC

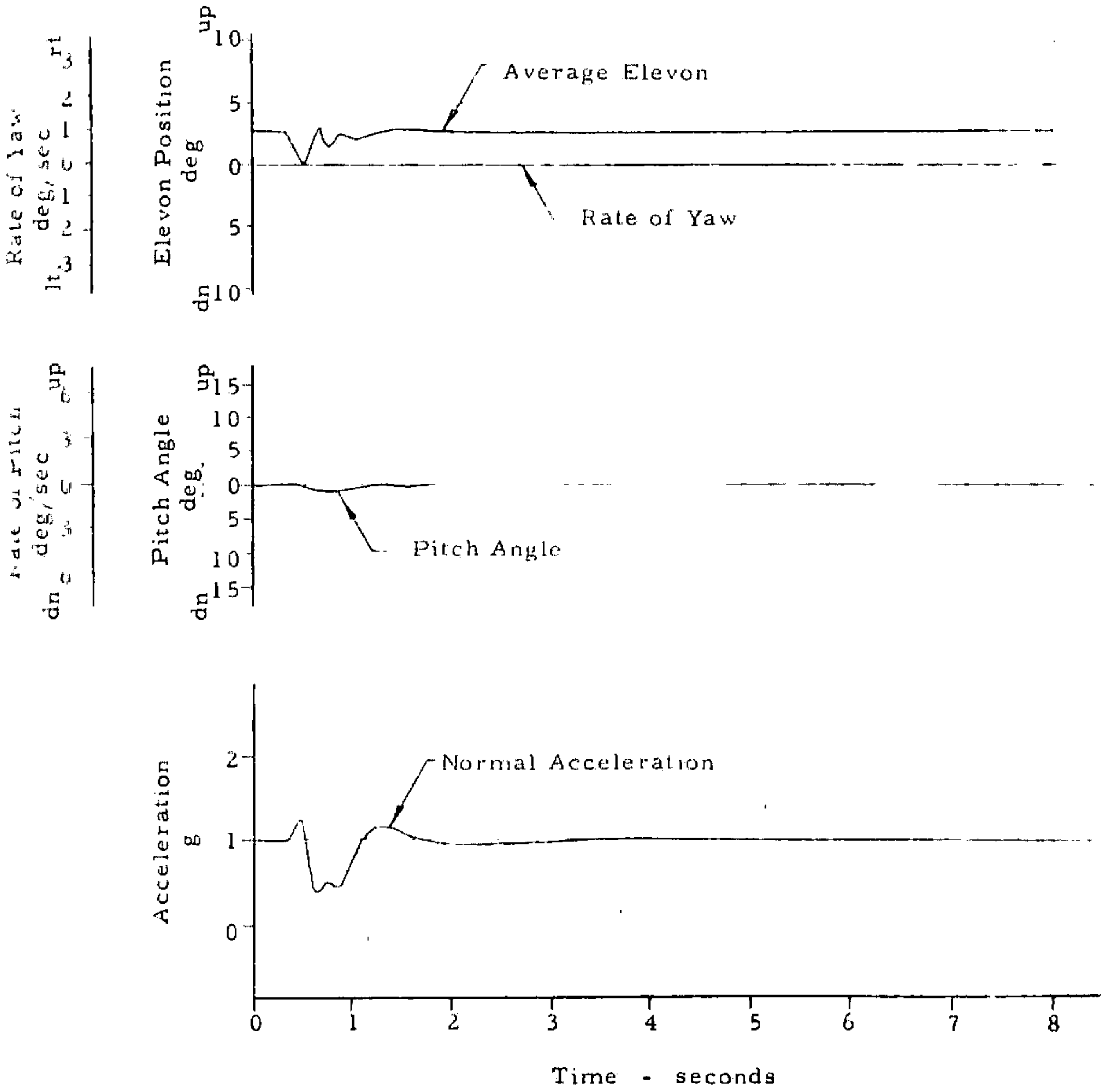


Figure No. 7

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration Pitch Damper Off
35,000 Altitude Mach No. 1.1
23400 Lbs. Weight Yaw Damper On
C.G. 27.1 percent MAC

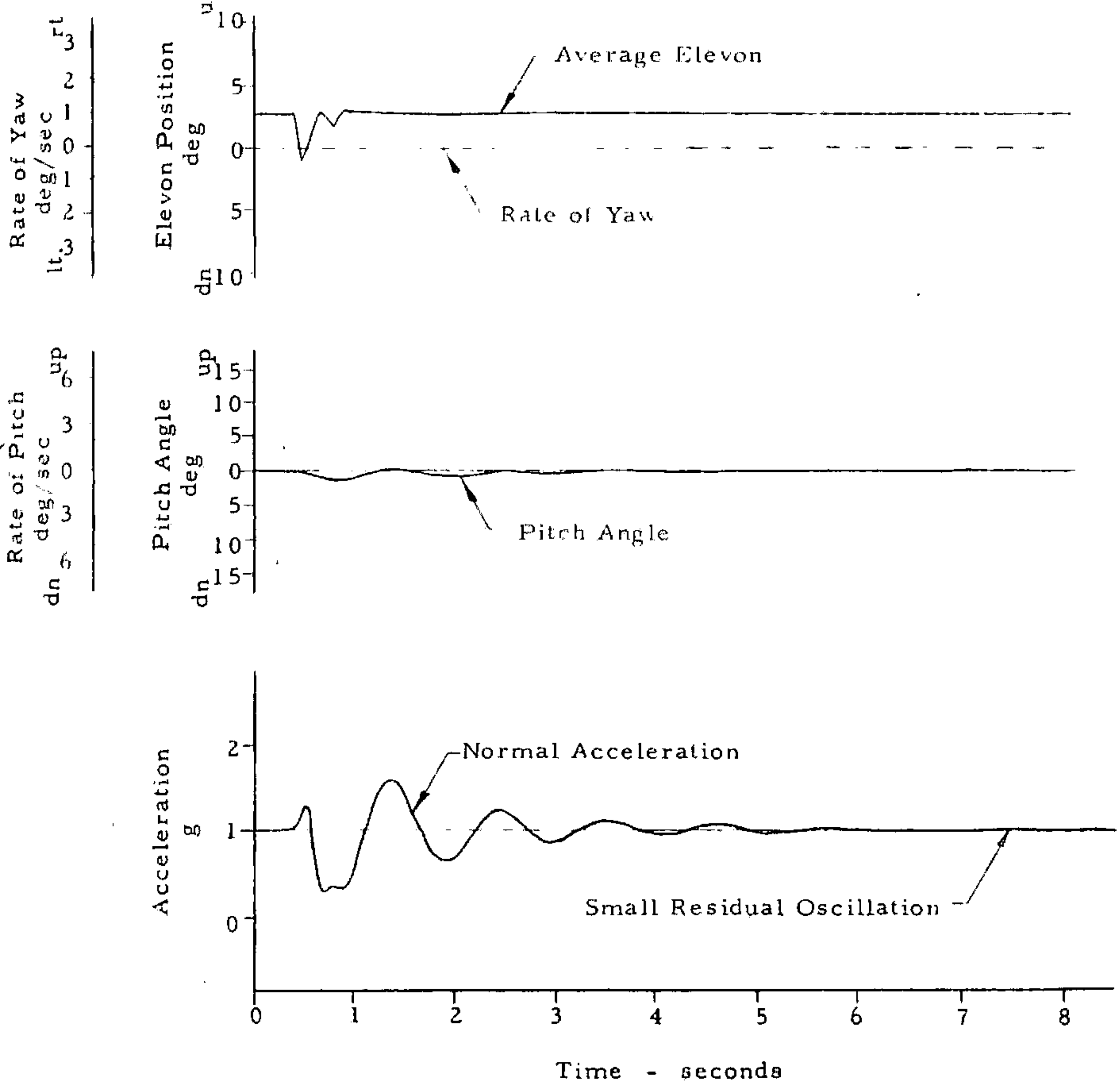
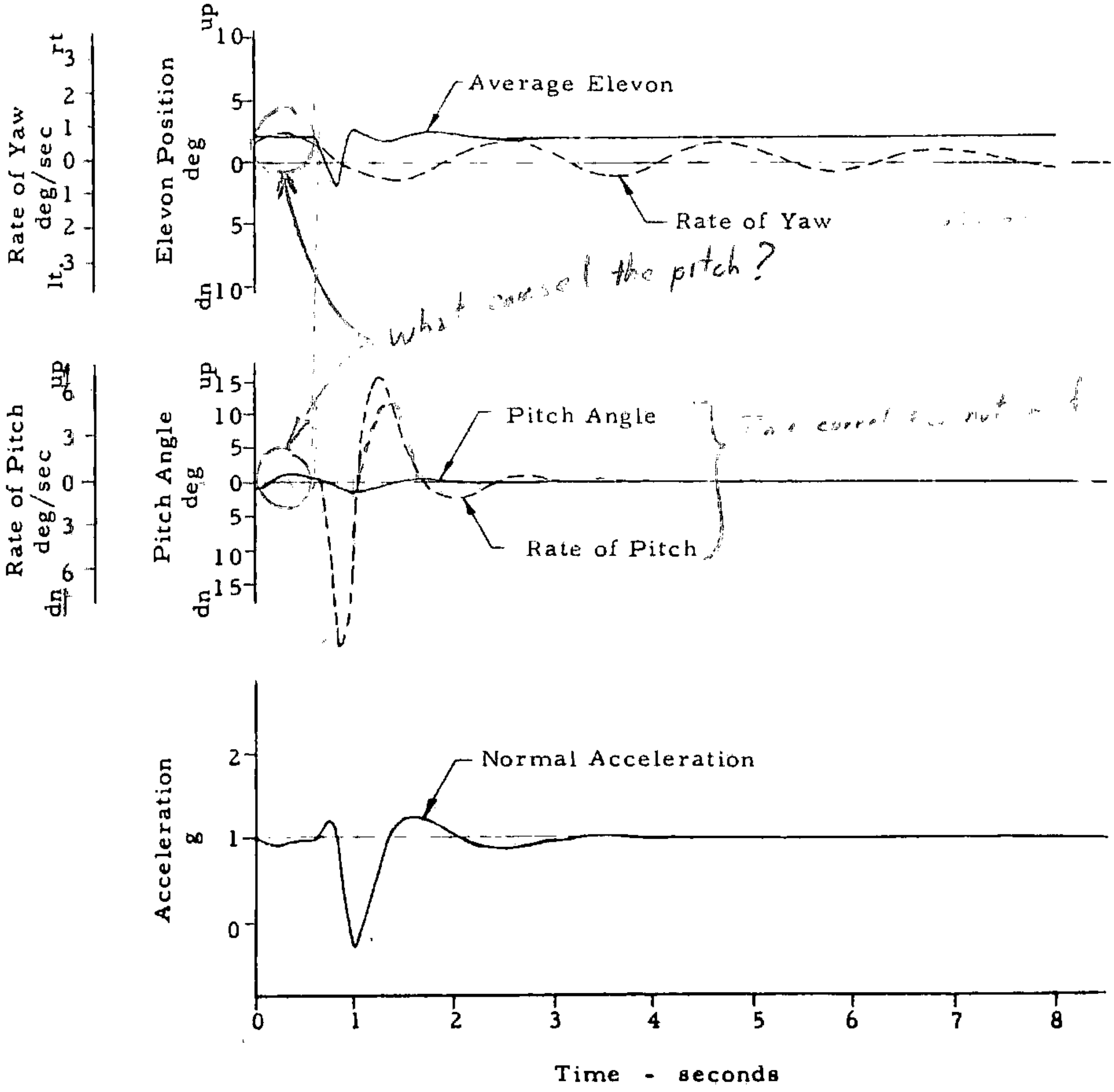


Figure No. 8

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
24600 Lbs. Weight
Pitch Damper On
Mach No. 1.1
Yaw Damper Off
C.G. 26.9 percent MAC



what caused the pitch?

[illegible] correl [illegible] not [illegible]

Figure No. 9

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Combat	Configuration	Pitch Damper	Off
35,000	Altitude	Mach No.	1.1
24600	Lbs. Weight	Yaw Damper	Off

C.G. 26.9 percent MAC

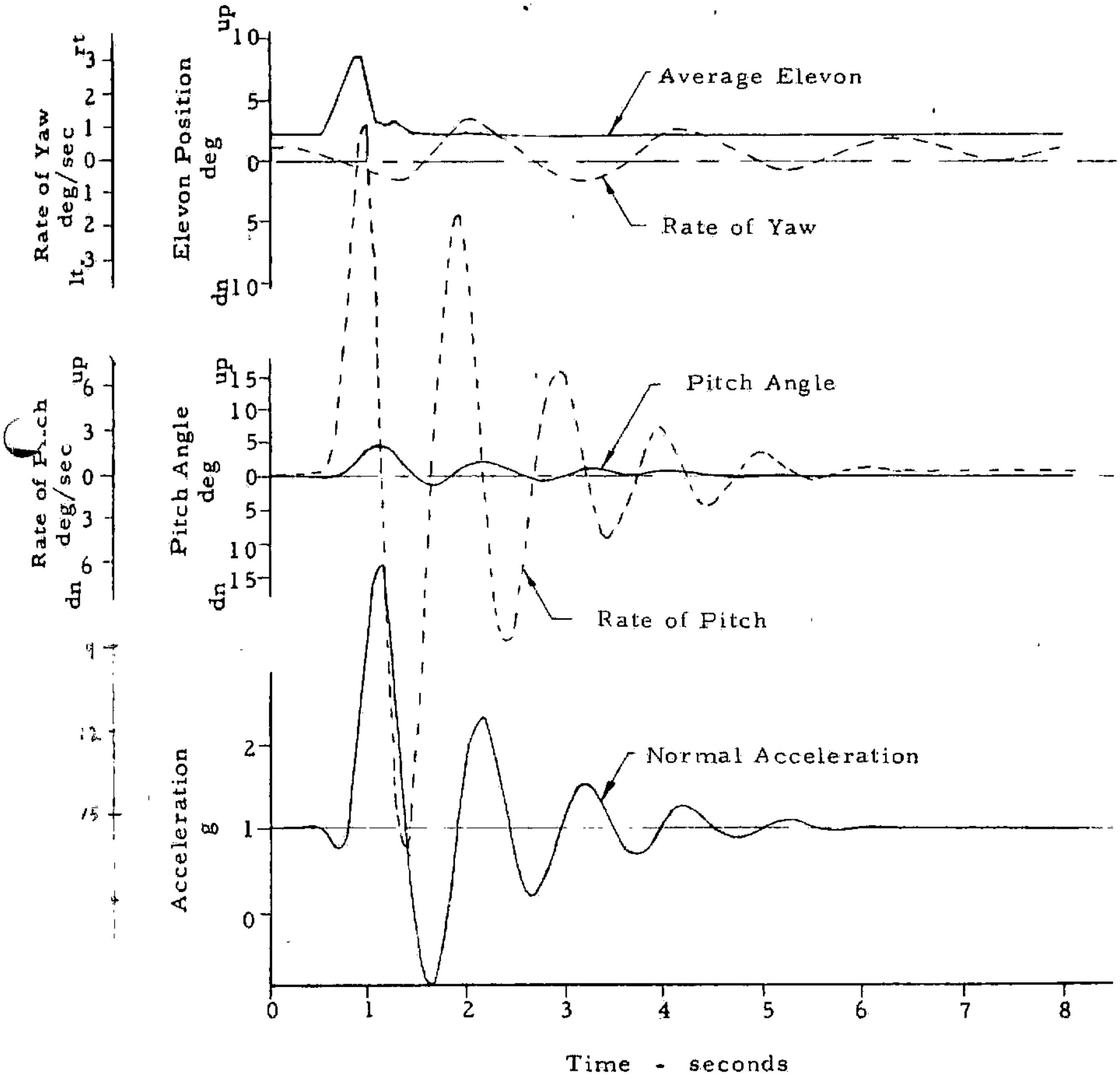


Figure No. 10

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration

35,000 Altitude

24000 Lbs. Weight

Pitch Damper On

Mach No. 0.9

Yaw Damper On

C.G. 27.0 percent MAC

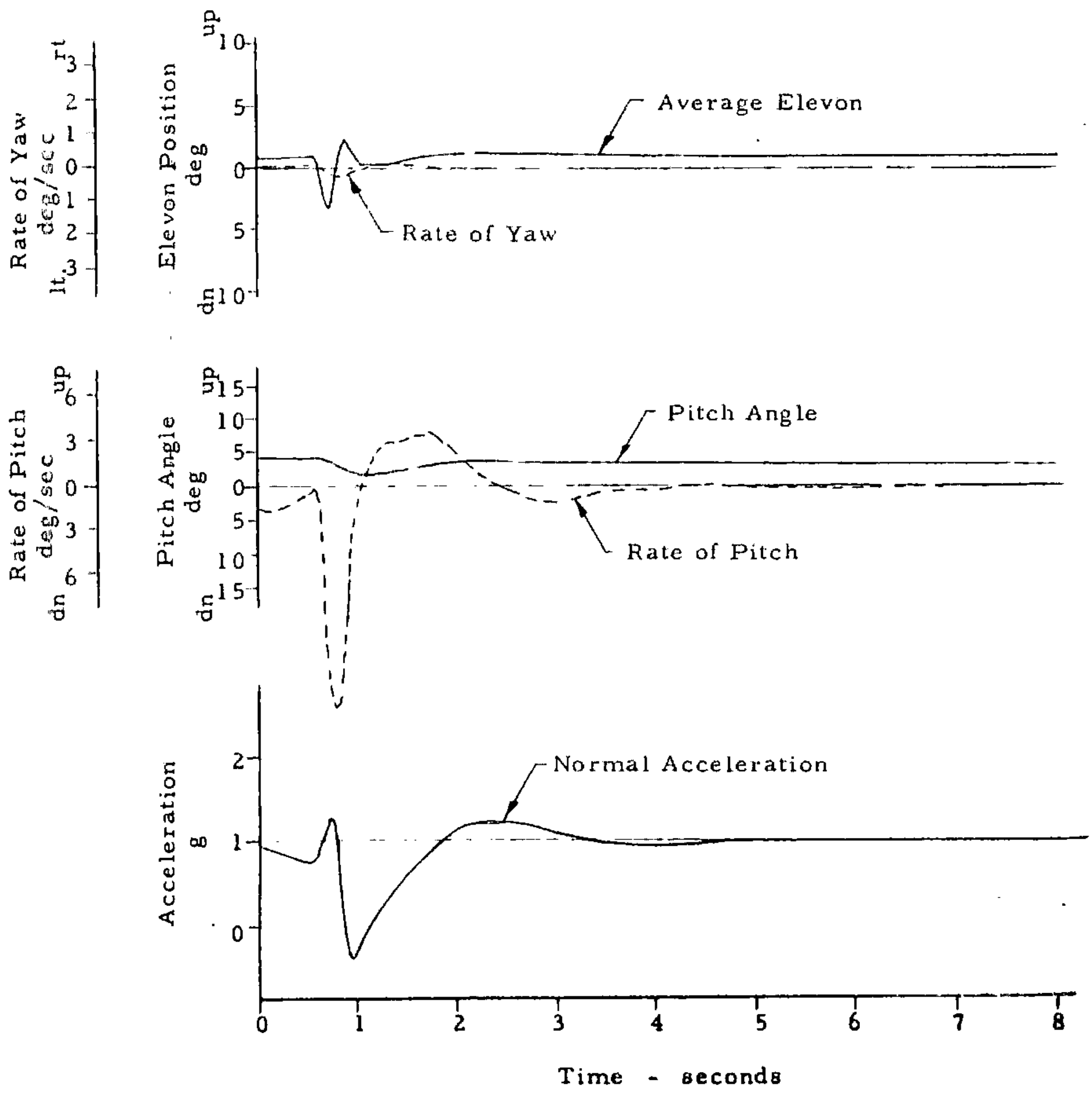


Figure No. 11

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
24200 Lbs. Weight

Pitch Damper Off
Mach No. 0.9
Yaw Damper Off

C.G. 27.0 percent MAC

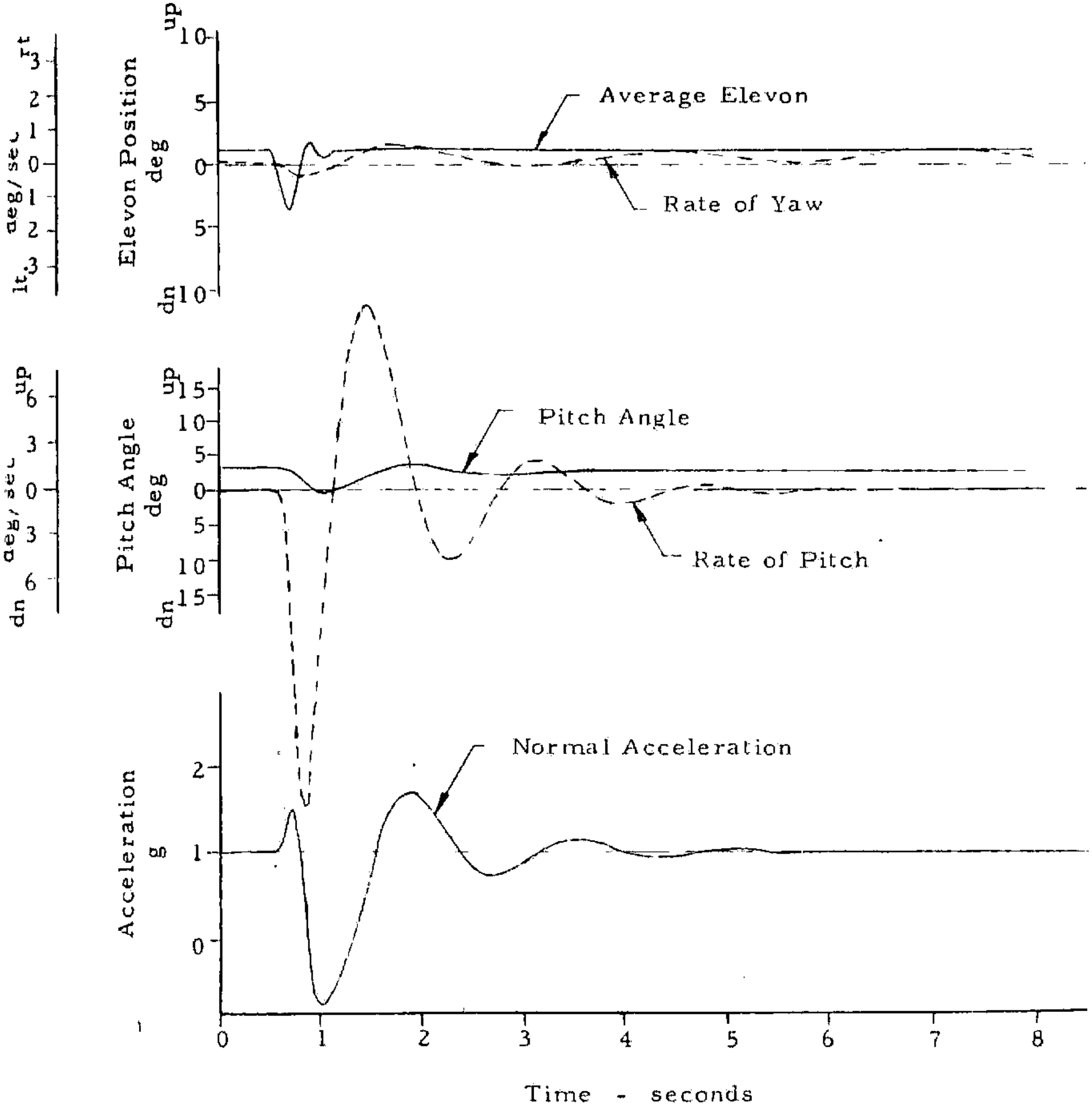


Figure No. 12

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
24000 Lbs. Weight

Pitch Damper On
Mach No. 0.7
Yaw Damper On

C.G. 27.0 percent MAC

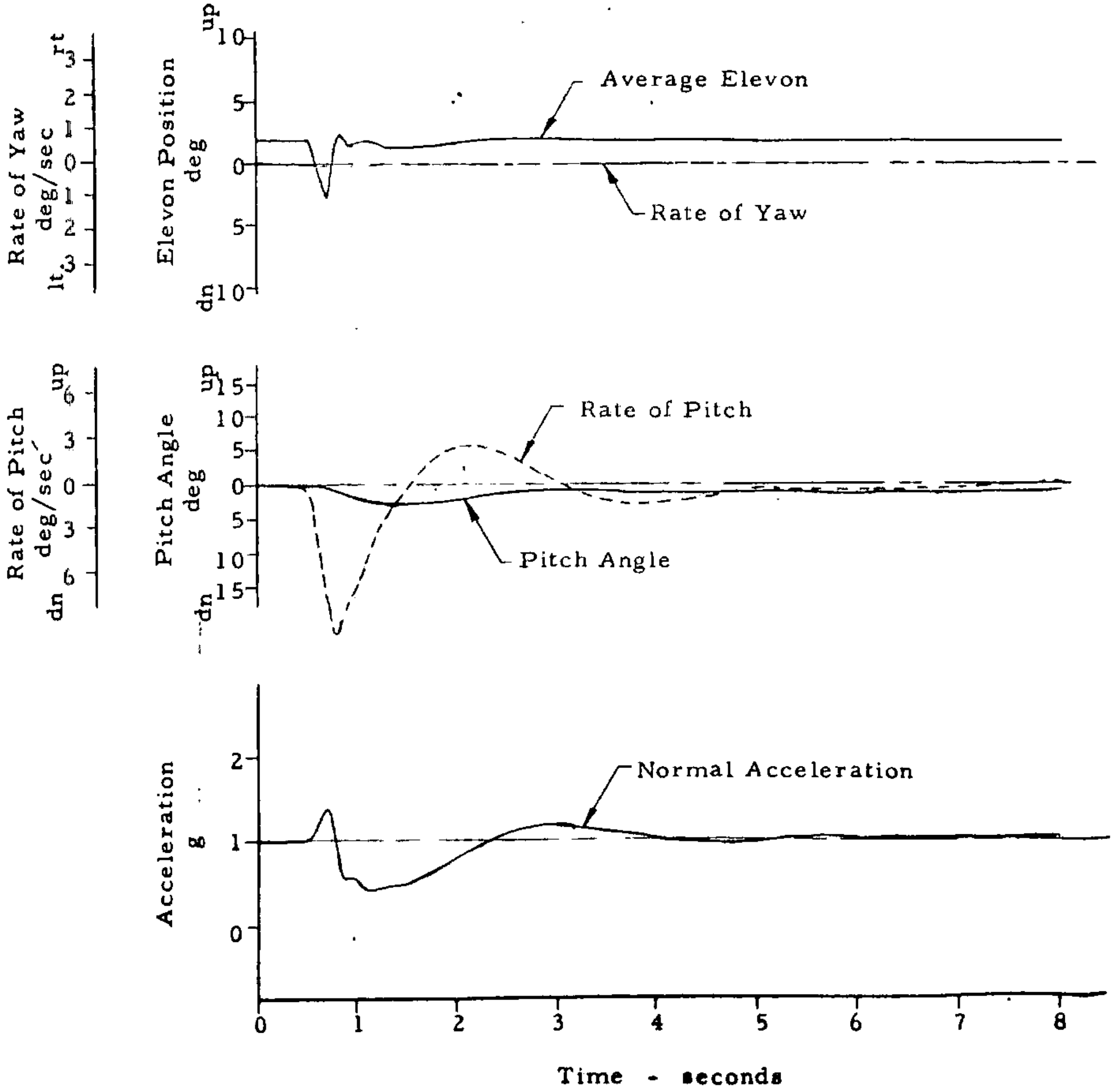


Figure No. 13

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
24000 Lbs. Weight

Pitch Damper Off
Mach No. 0.7
Yaw Damper Off

C.G. 27.0 percent MAC

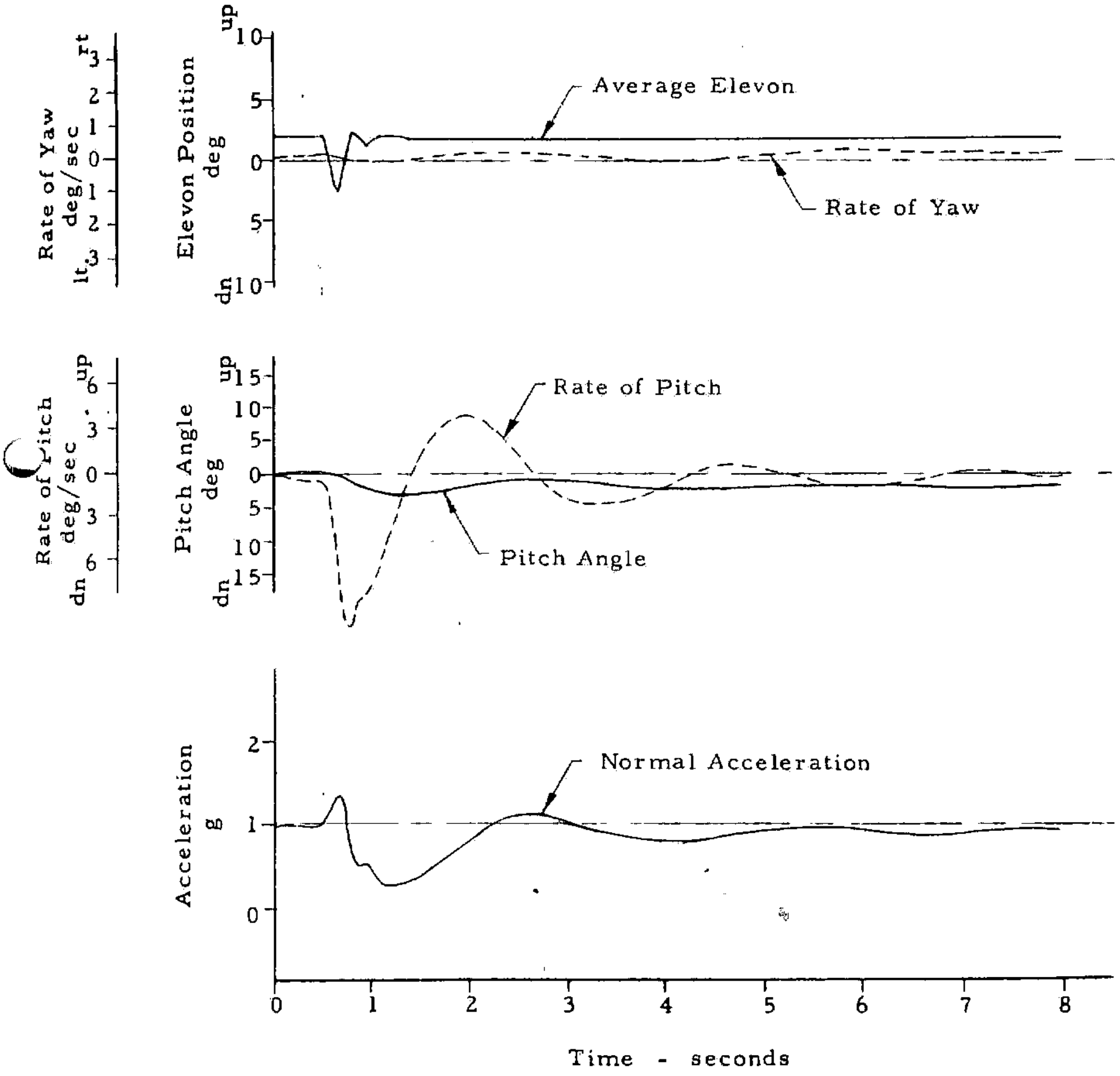
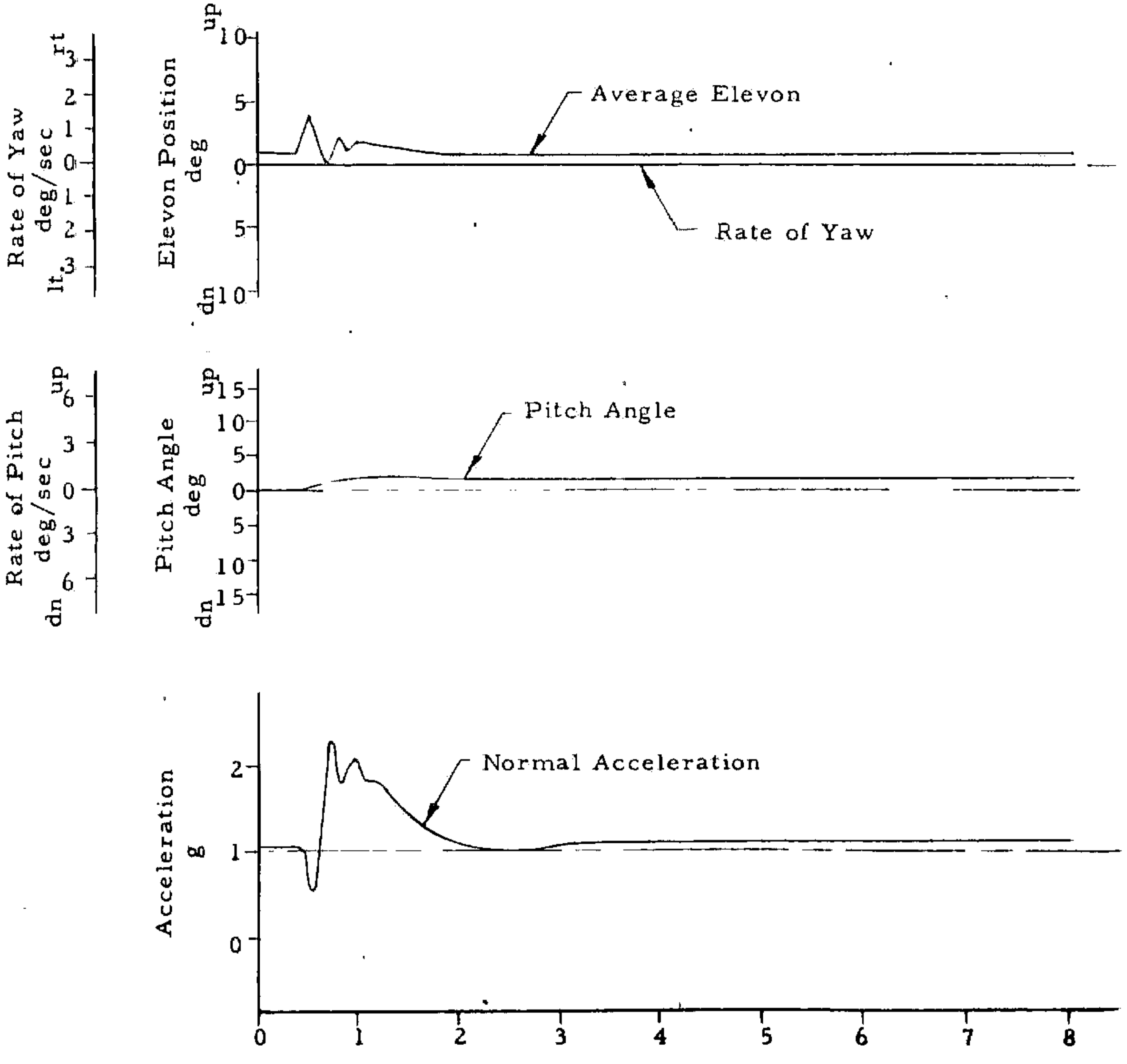


Figure No. 14

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
10,000 Altitude
22800 Lbs. Weight
Pitch Damper On
Mach No. 0.70
Yaw Damper On

C.G. 27.0 percent MAC



Time - seconds

APPENDIX I

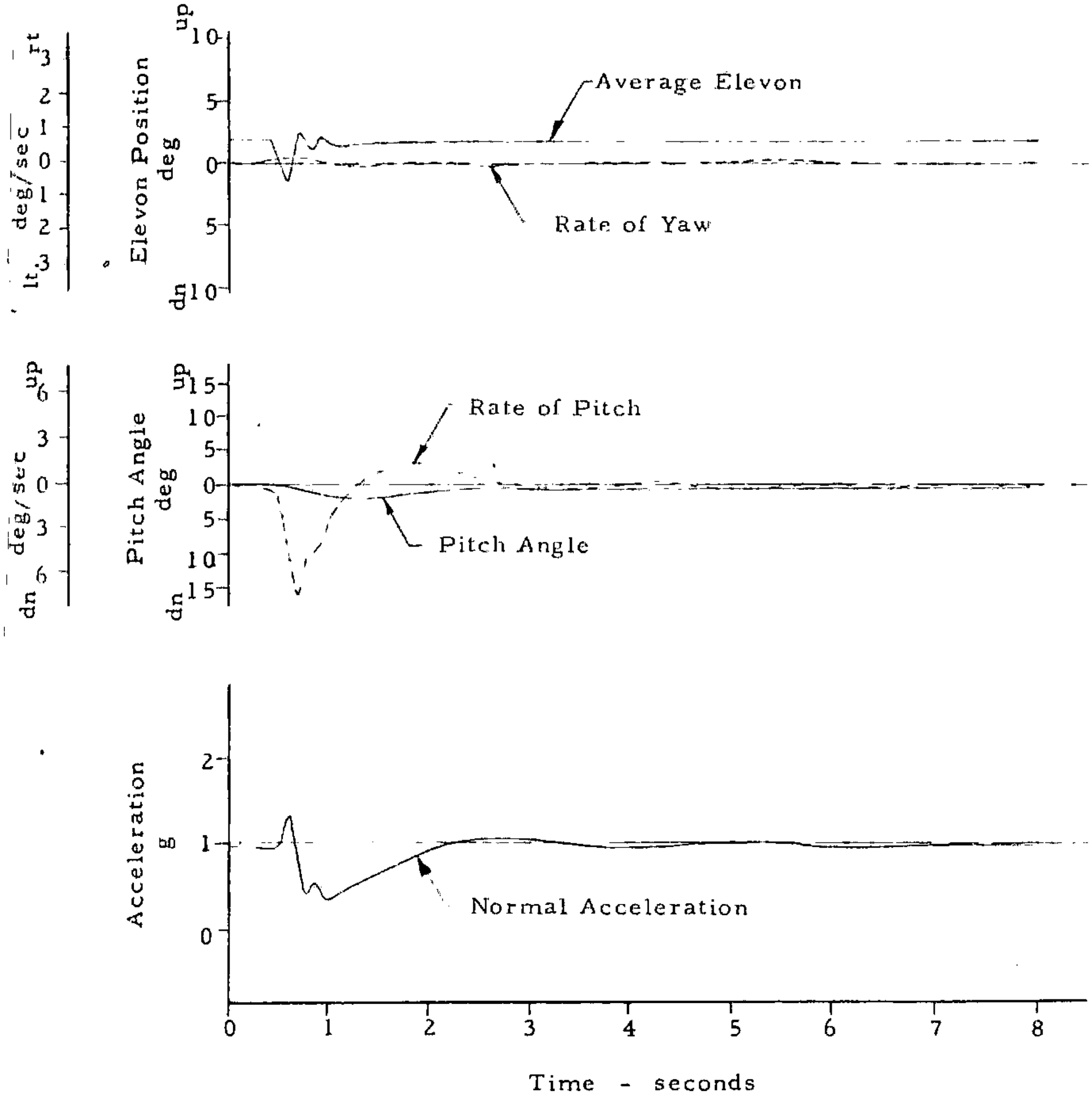
Figure No. 15

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
10,000 Altitude
22700 Lbs. Weight

Pitch Damper On
Mach No. 0.5
Yaw Damper On

C.G. 27.0 percent MAC



APPENDIX I

Figure No. 10

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Power Approach Configuration   Pitch Damper On

10,000 Altitude   Mach No. 0.327

21700 Lbs. Weight   Yaw Damper On

C.G. 25.7 percent MAC

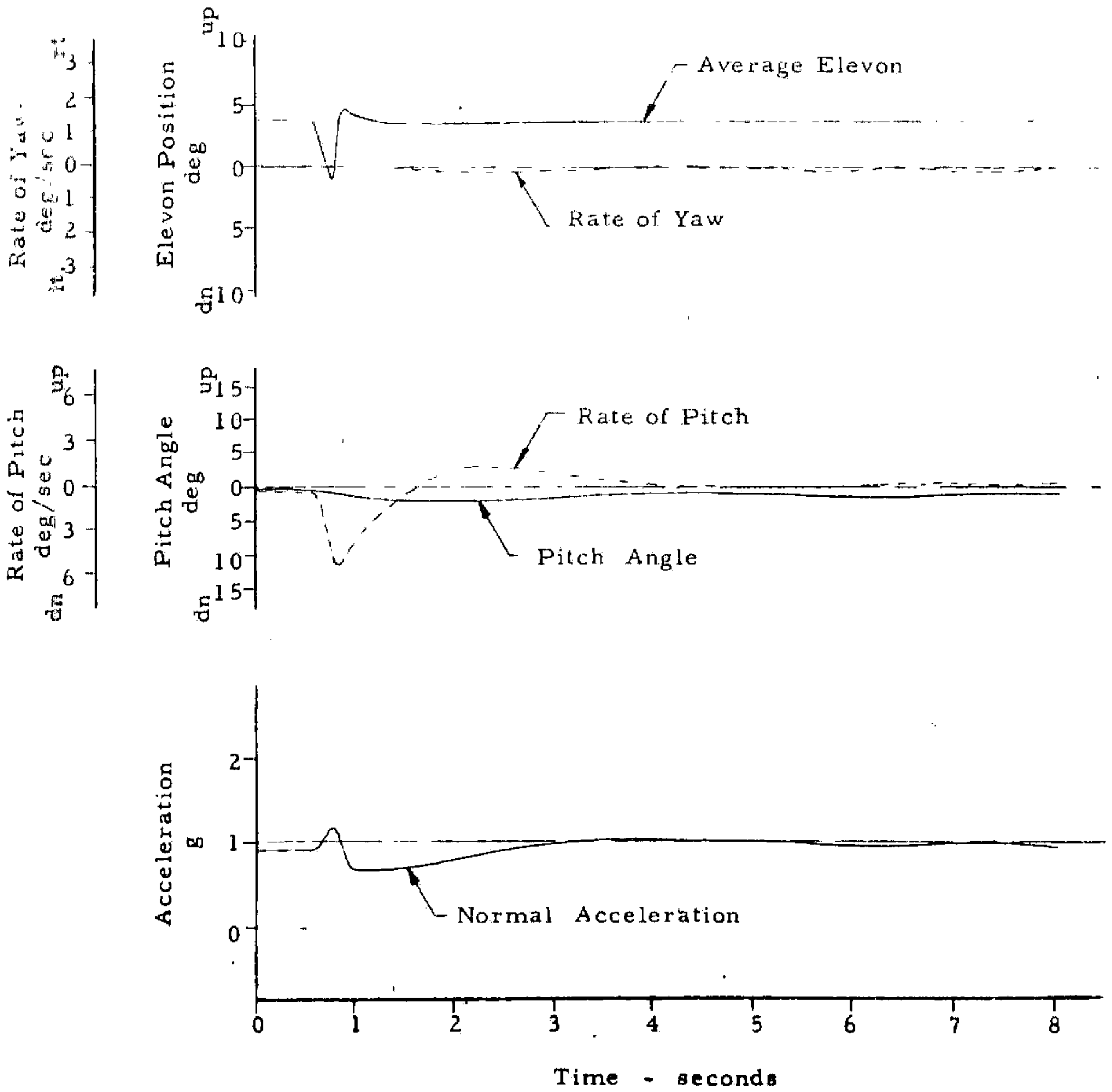


Figure No. 17

DYNAMIC LONGITUDINAL STABILITY
F-102A USAF No. 53-1796

Power Approach Configuration Pitch Damper Off
10,000 Altitude Mach No. 0.327
21700 Lbs. Weight Yaw Damper On

C.G. 25.7 percent MAC

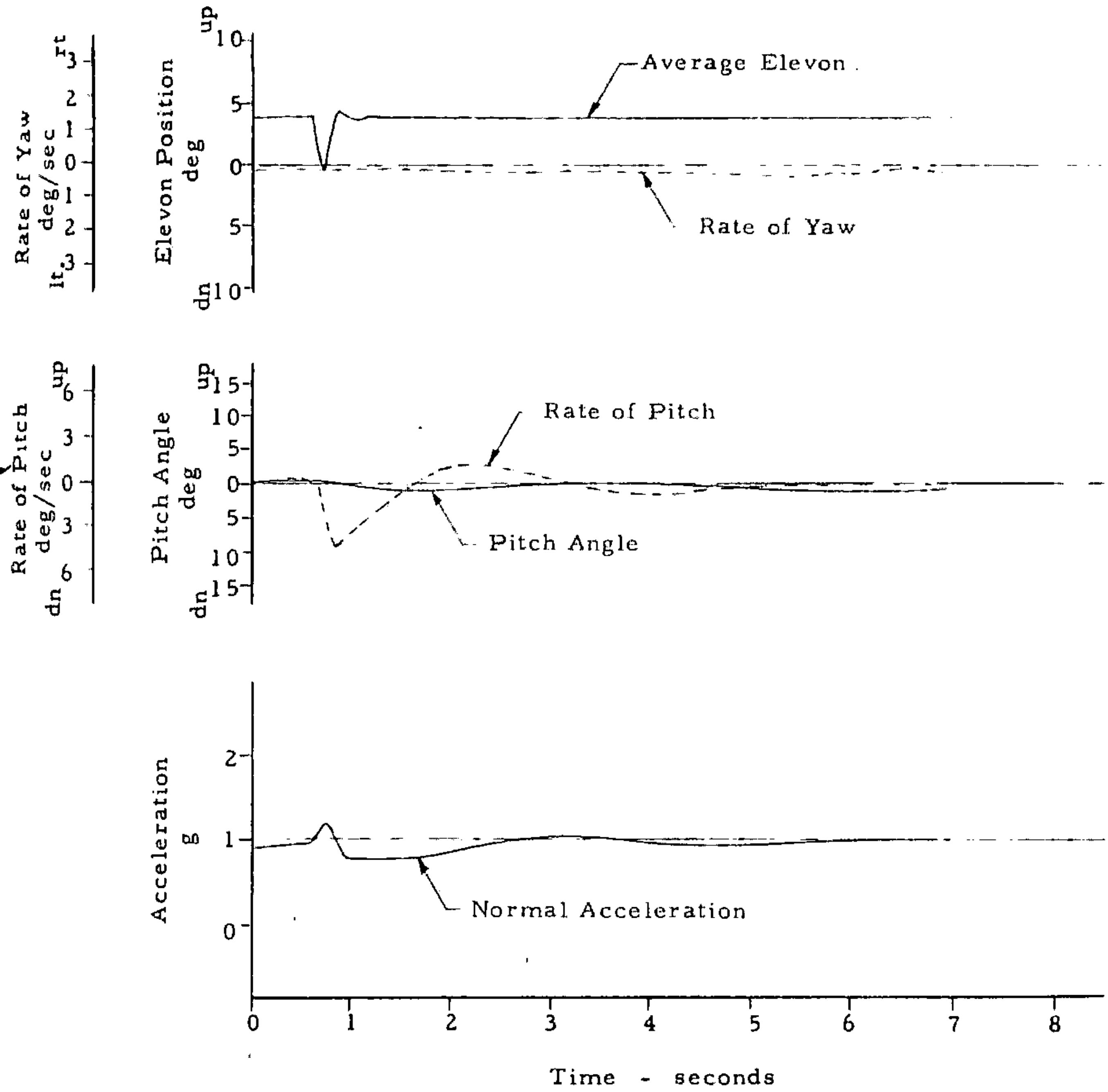


FIGURE NO. 18

STATIC LONGITUDINAL STABILITY

F-102A USAF NO. 53-1796

35000 FT. ALTITUDE

TRIM SERVO ON AND OFF

CRUISE CONFIGURATION

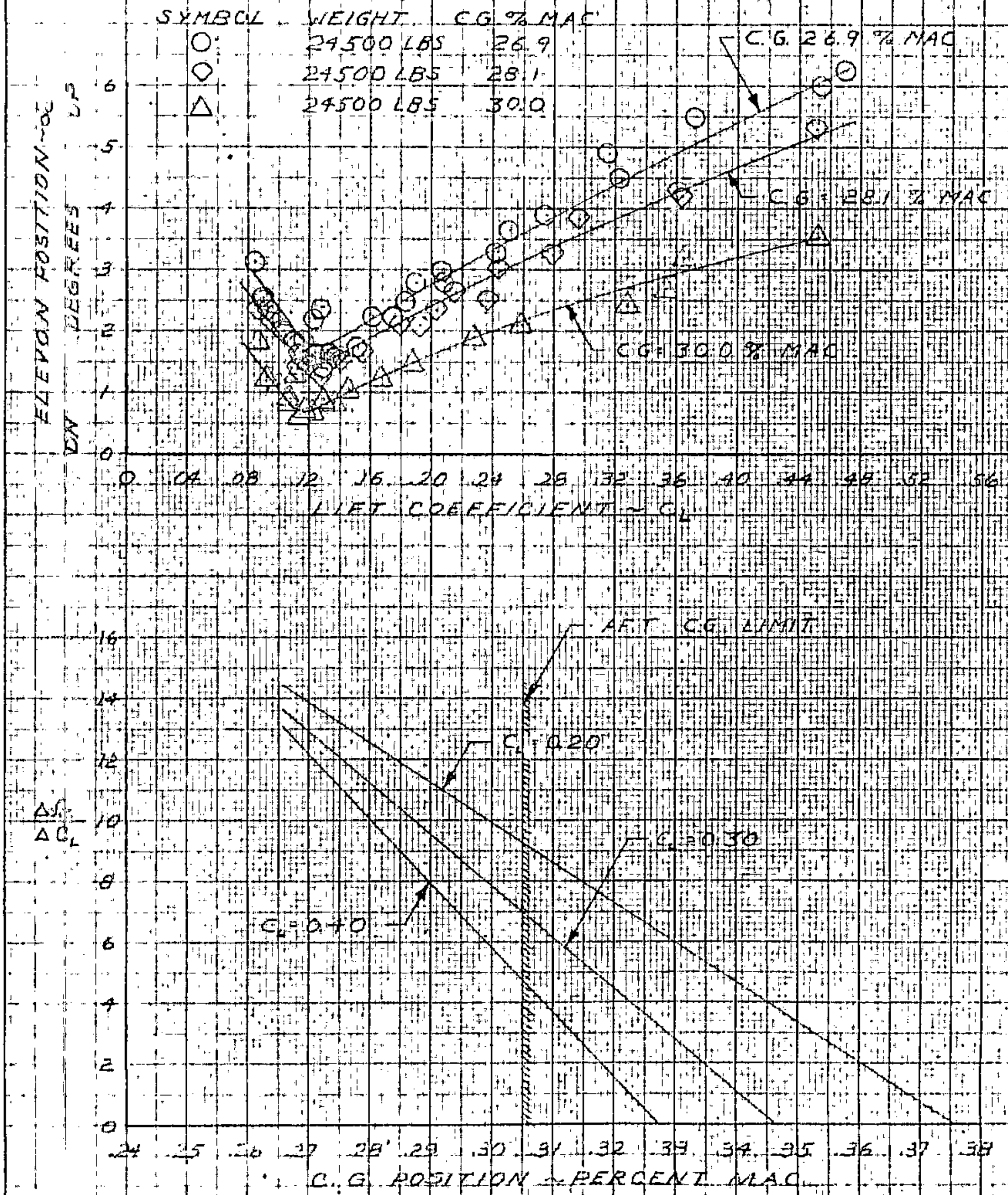


FIGURE NO. 17

STATIC LONGITUDINAL STABILITY

F-102A USAF NO. 53-1796

45,000 FT. ALTITUDE

TRIM SERVO ON

CRUISE CONFIGURATION

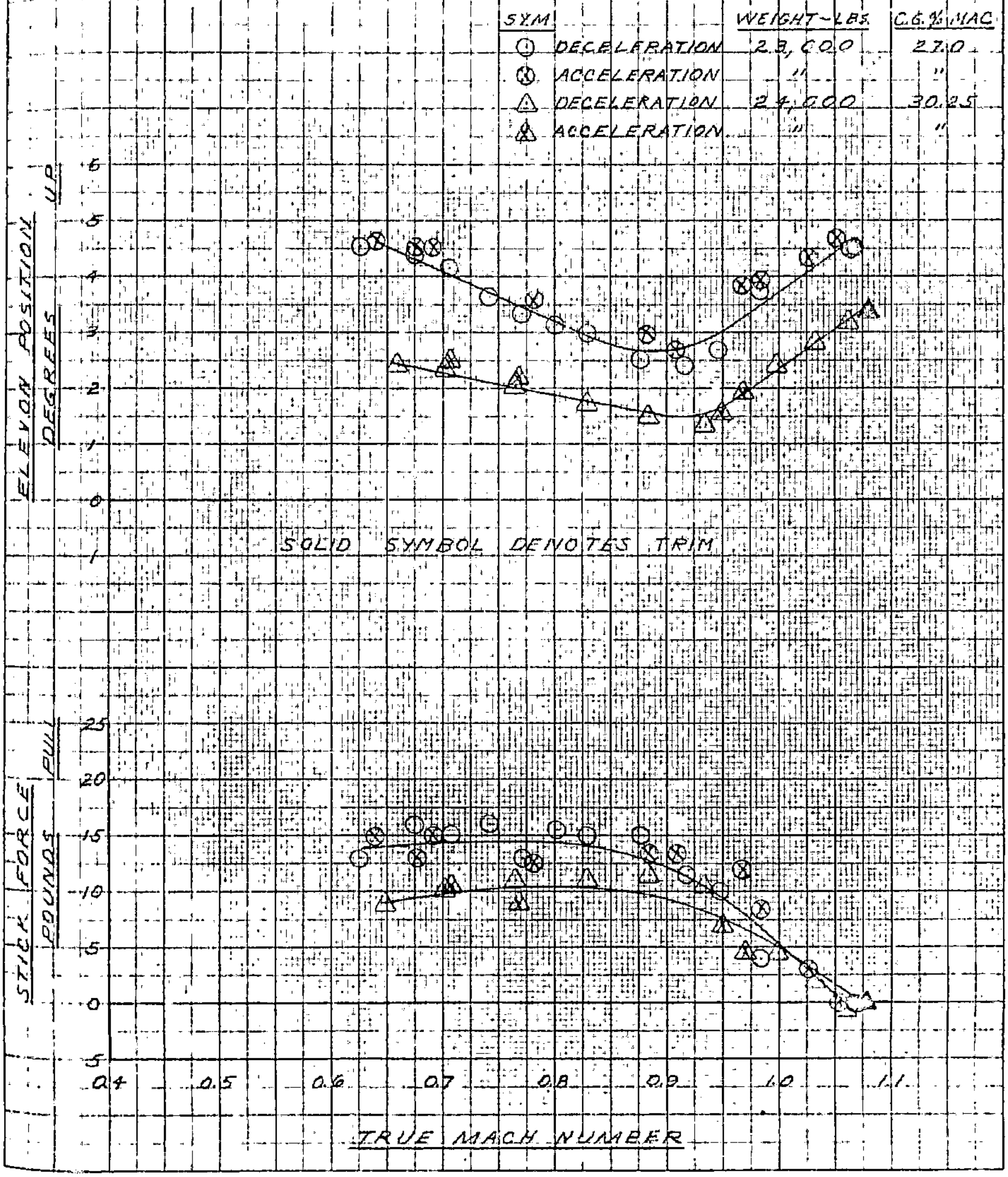


FIGURE NO. 20

STATIC LONGITUDINAL STABILITY

F-102A USAF NO. 53-1796

45,000 FT. ALTITUDE

TRIM SERVO OFF

CRUISE CONFIGURATION

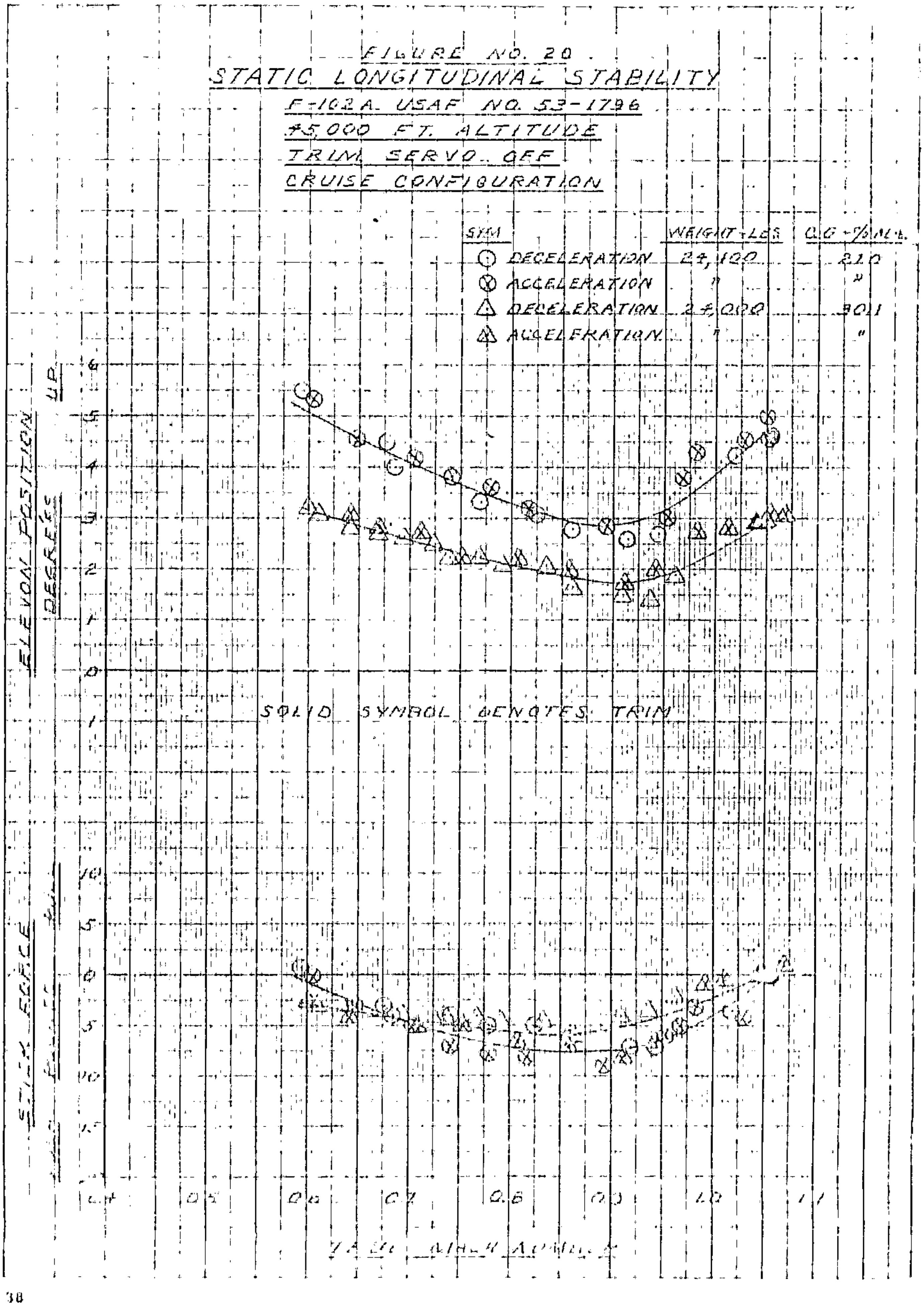
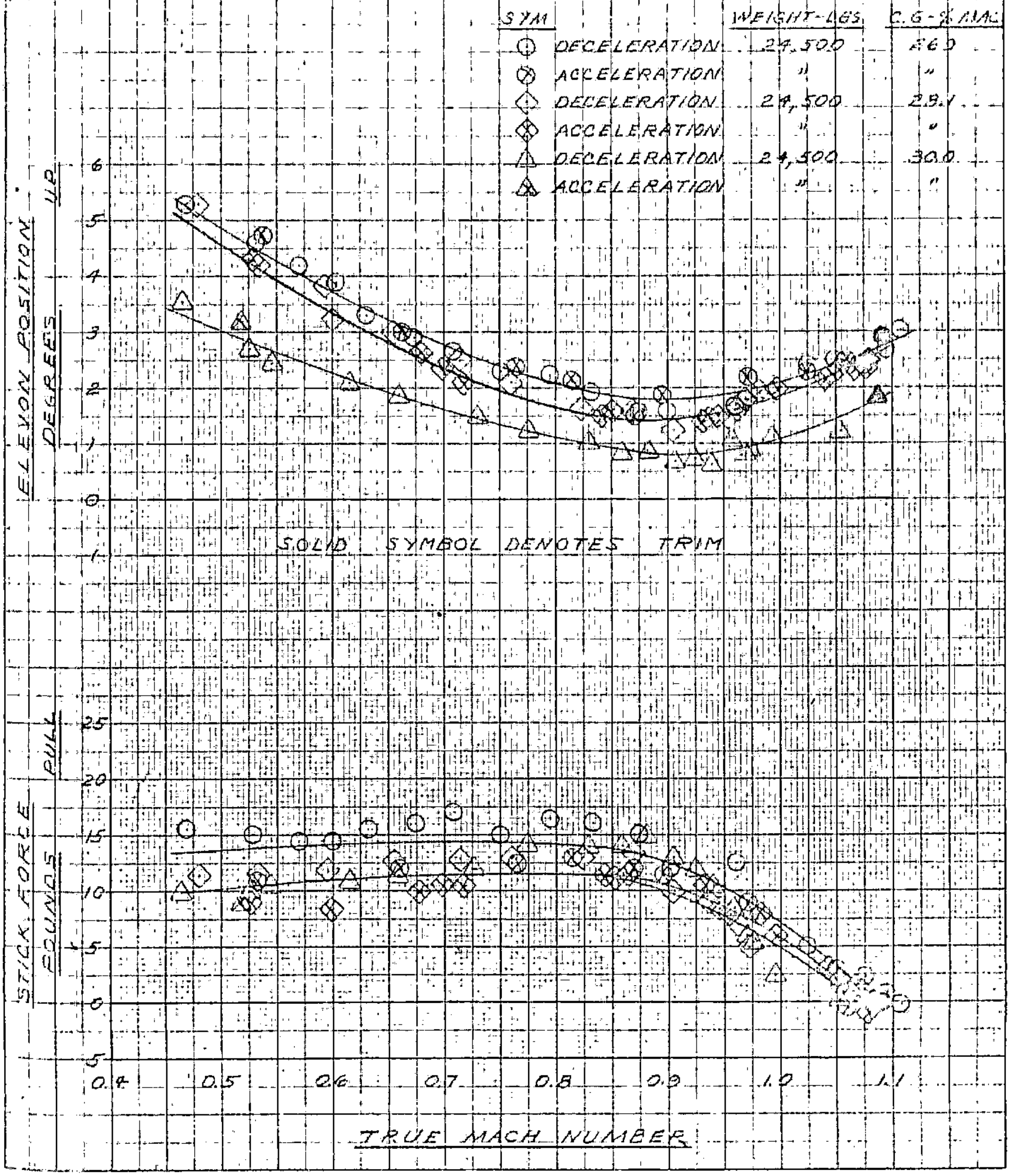


FIGURE NO. 21

STATIC LONGITUDINAL STABILITY

F-102A. USAF NO. 53-1796.
35,000 FT. ALTITUDE
TRIM SERVO ON
CRUISE CONFIGURATION



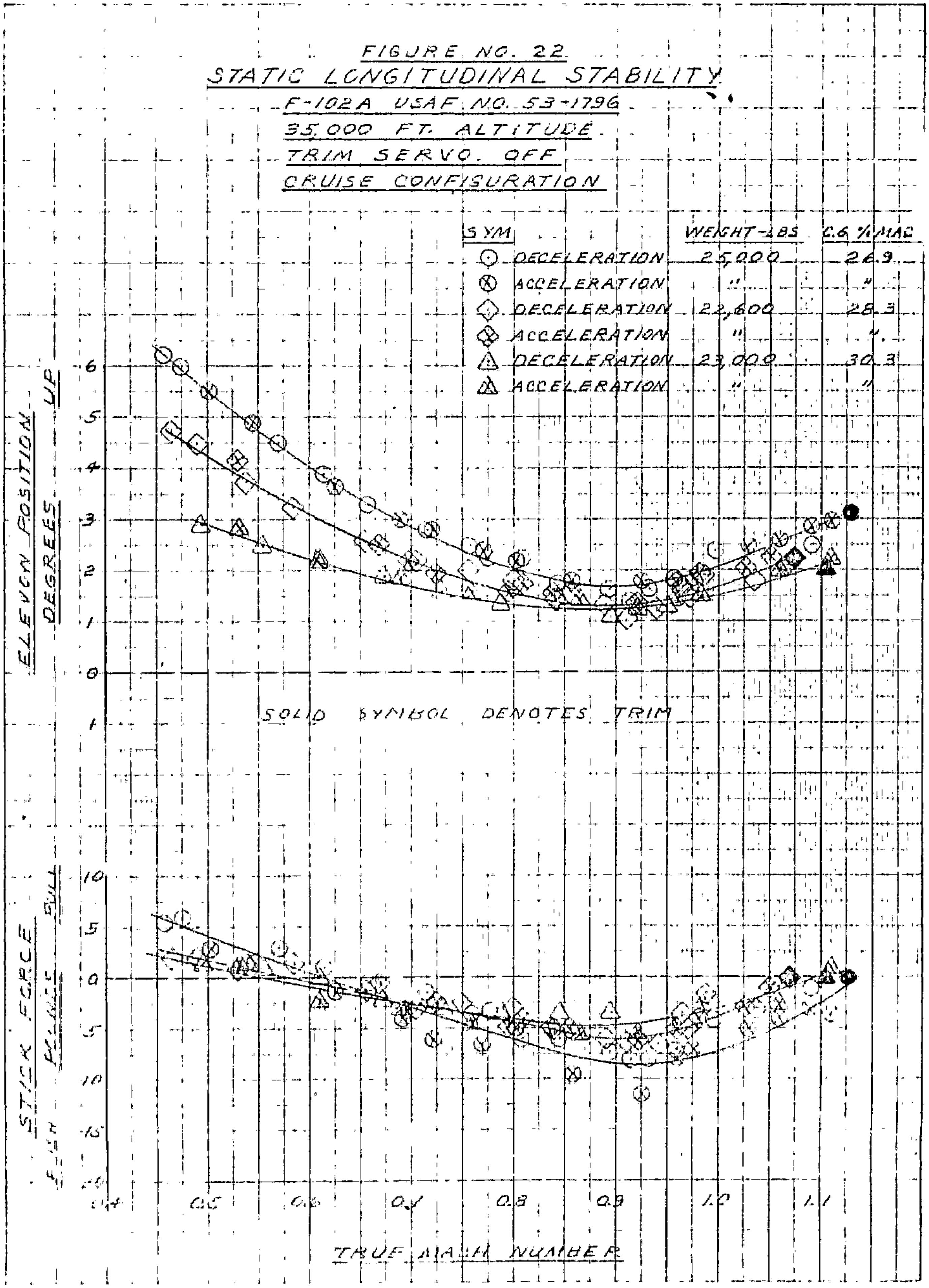


FIGURE NO. 22

STATIC LONGITUDINAL STABILITY

F-102A USAF NO. 53-1796

35,000 FT. ALTITUDE

TRIM SERVO. OFF

CRUISE CONFIGURATION

FIGURE NO. 23

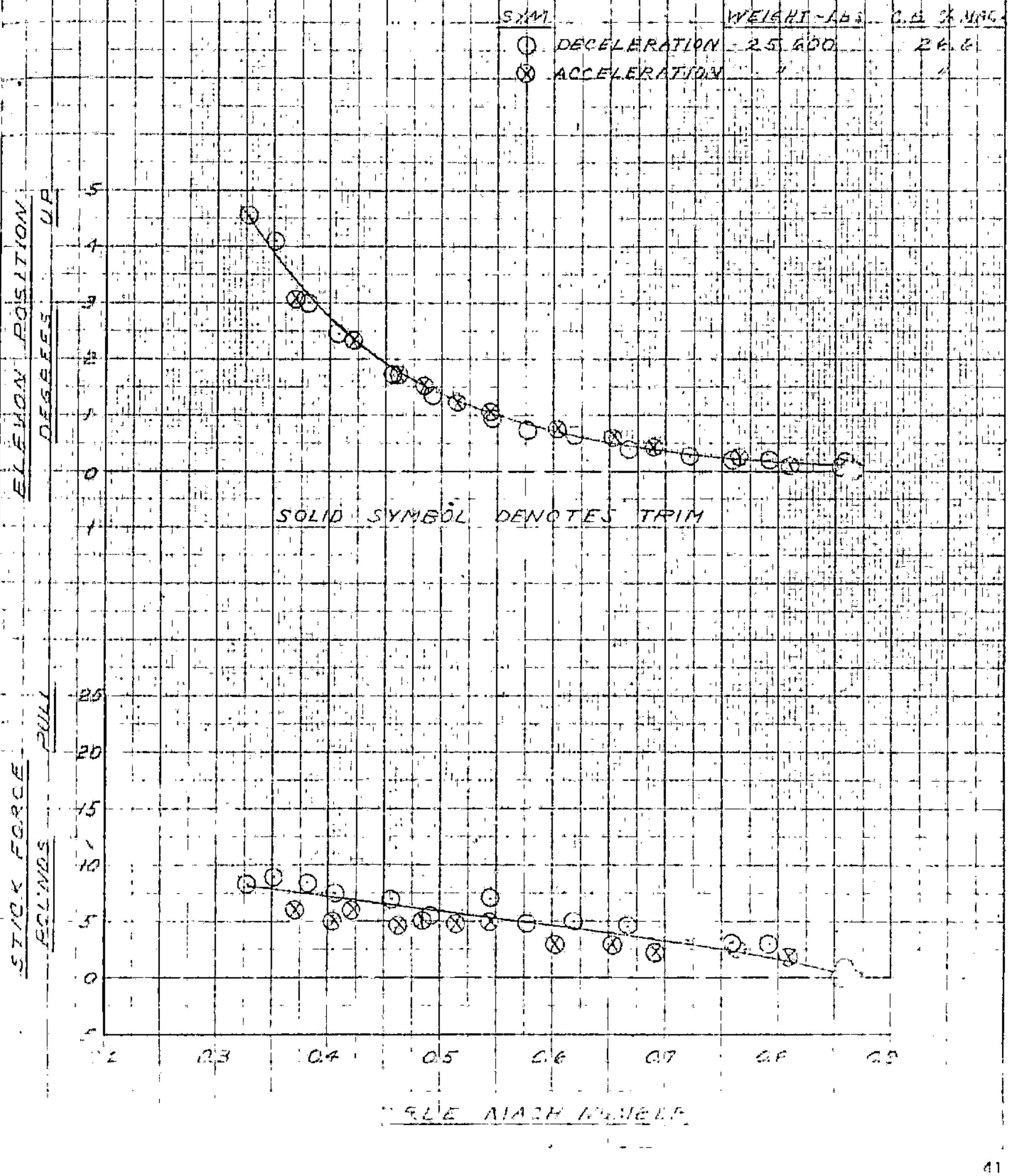
STATIC LONGITUDINAL STABILITY

F-102A USAF NO 53-1796

10000 FT. ALTITUDE

TRIM SERVO ON

CRUISE CONFIGURATION



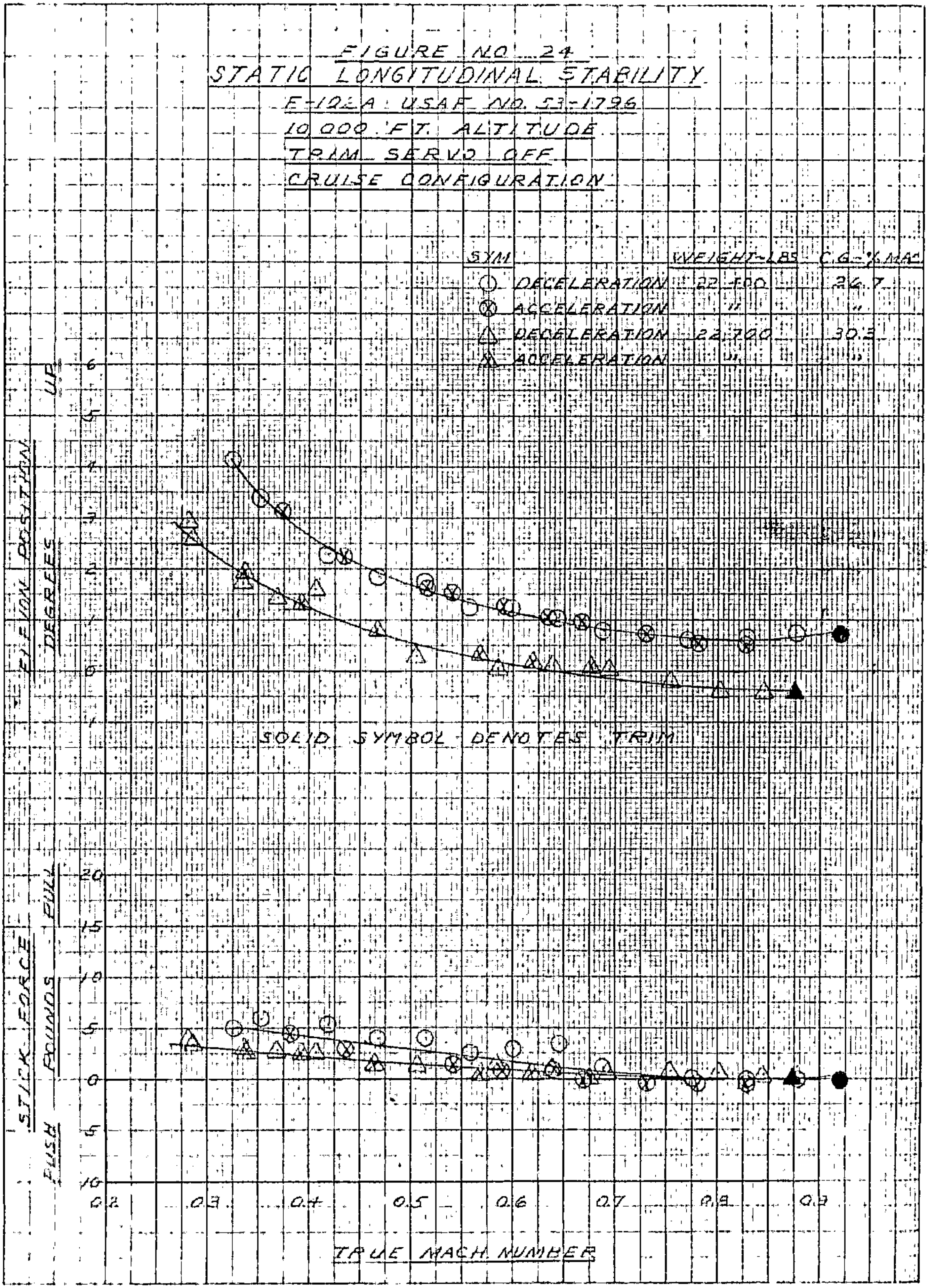


FIGURE NO. 24
STATIC LONGITUDINAL STABILITY
F-102A USAF NO. 53-1796
10,000 FT. ALTITUDE
TRIM SERVO OFF
CRUISE CONFIGURATION

FIGURE NO. 25
STATIC LONGITUDINAL STABILITY
F-102A USAF NO. 53-1796
10,000 FT ALTITUDE
TRIM SERVO OFF
POWER APPROACH CONFIGURATION
LANDING GEAR EXTENDED
SPEED BRAKES IN

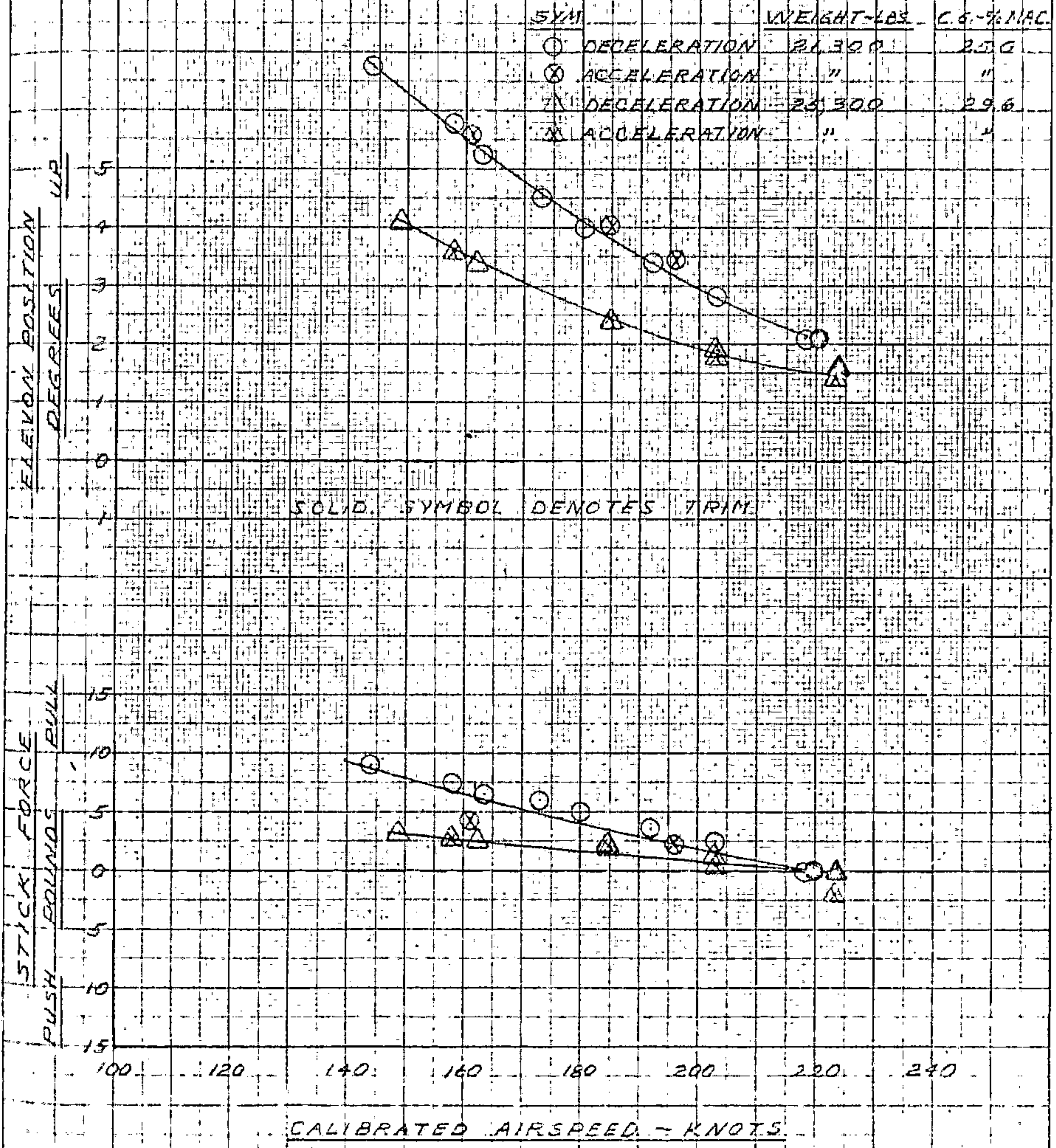
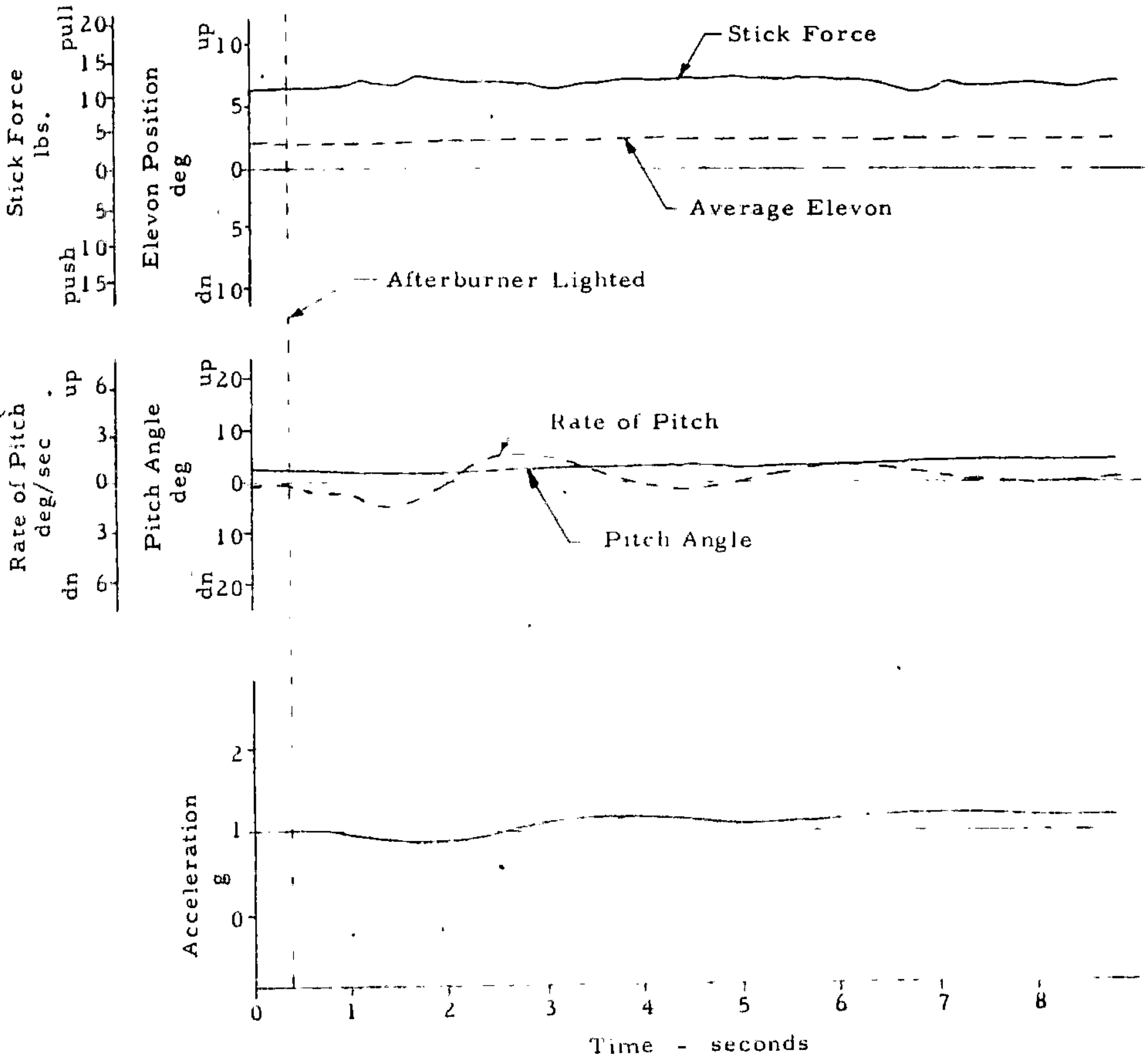


Figure No. 26

TIME HISTORY OF AFTERBURNER LIGHT
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
24,500 Weight

Pitch Damper On
Mach No. .72
C.G. 28.1 °/oMAC



$C_N =$

$N_0 = x_{cg} + \left(\frac{d\delta}{dC_N}\right) C_{m_\delta}$

$x_{cg} + \frac{dC_M}{dC_N}$

$-\frac{dC_M}{dC_N} = -N_0 + x_{cg}$

FIGURE NO 27

MANEUVERING FLIGHT

F-102A USAF NO 53-1736

50500 FT. ALTITUDE

CRUISE CONFIGURATION

0.970 MACH NUMBER

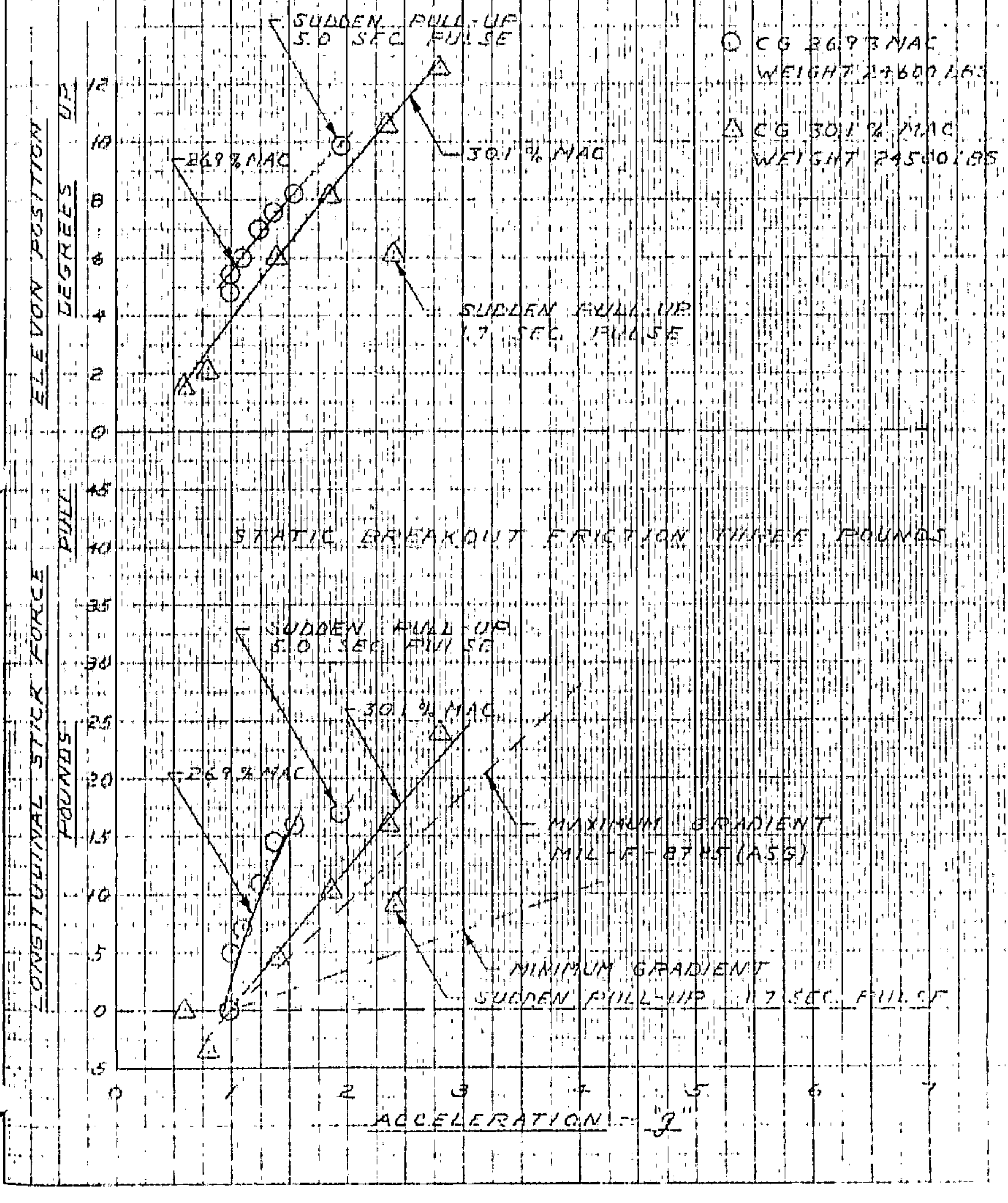
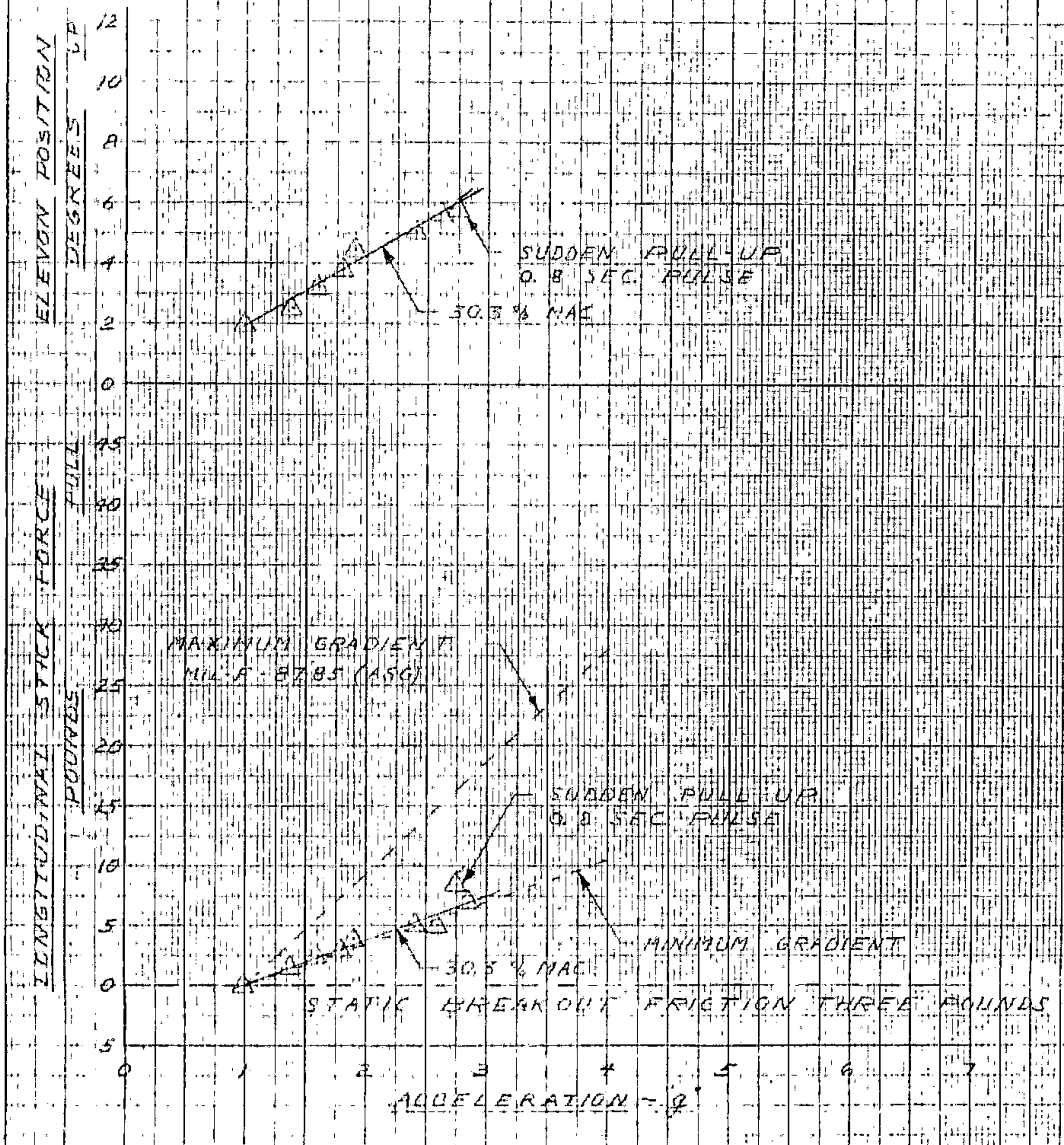


FIGURE NO. 28
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
50500 FT. ALTITUDE
CRUISE CONFIGURATION
0.890 MACH NUMBER

C. G. 30.3 % MAC
WEIGHT 23900LBS



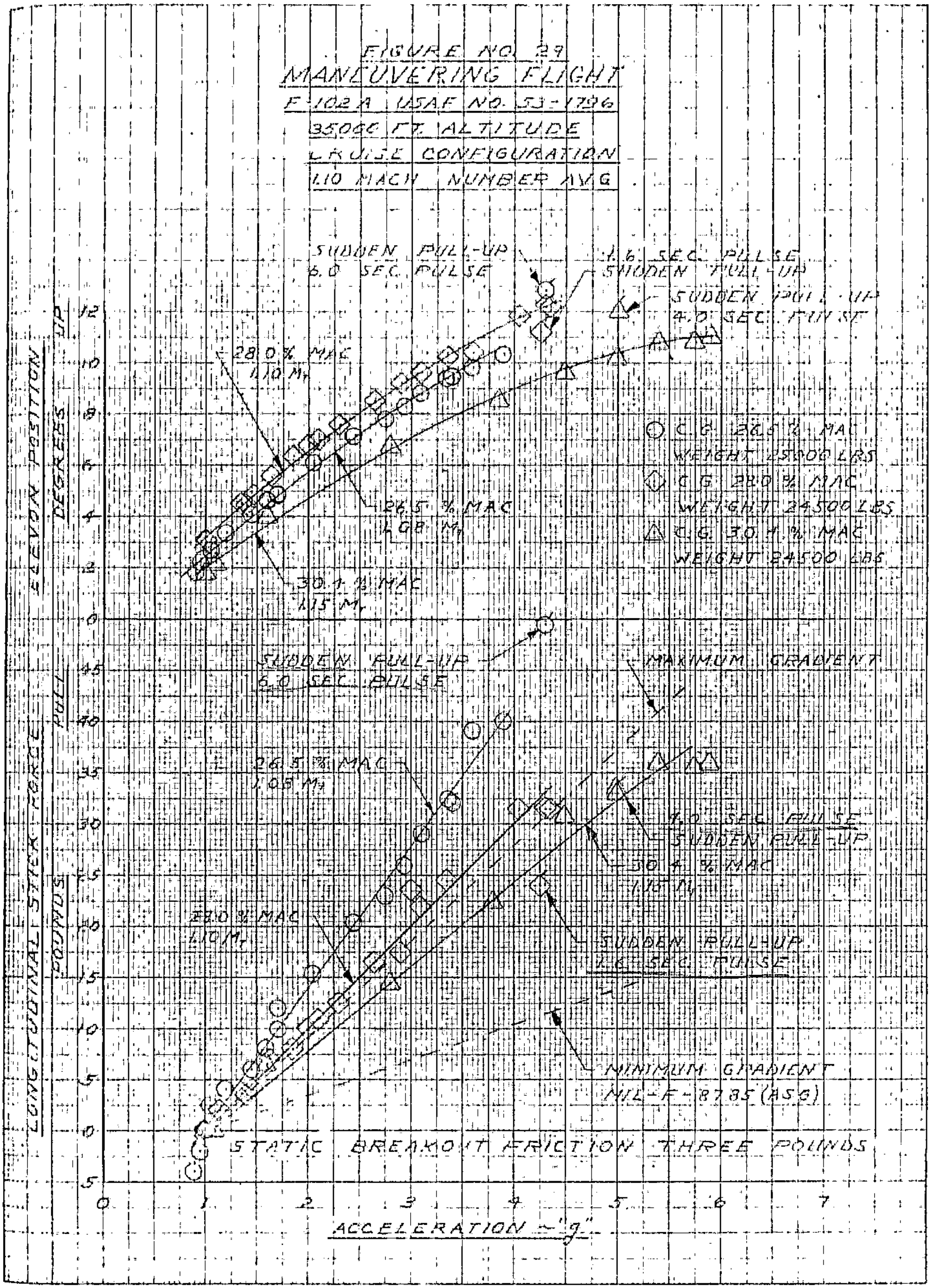


FIGURE NO. 29

MANEUVERING FLIGHT

F-102A USAF NO. 53-1796

35000 FT. ALTITUDE

CRUISE CONFIGURATION

1.10 MACH NUMBER AVG

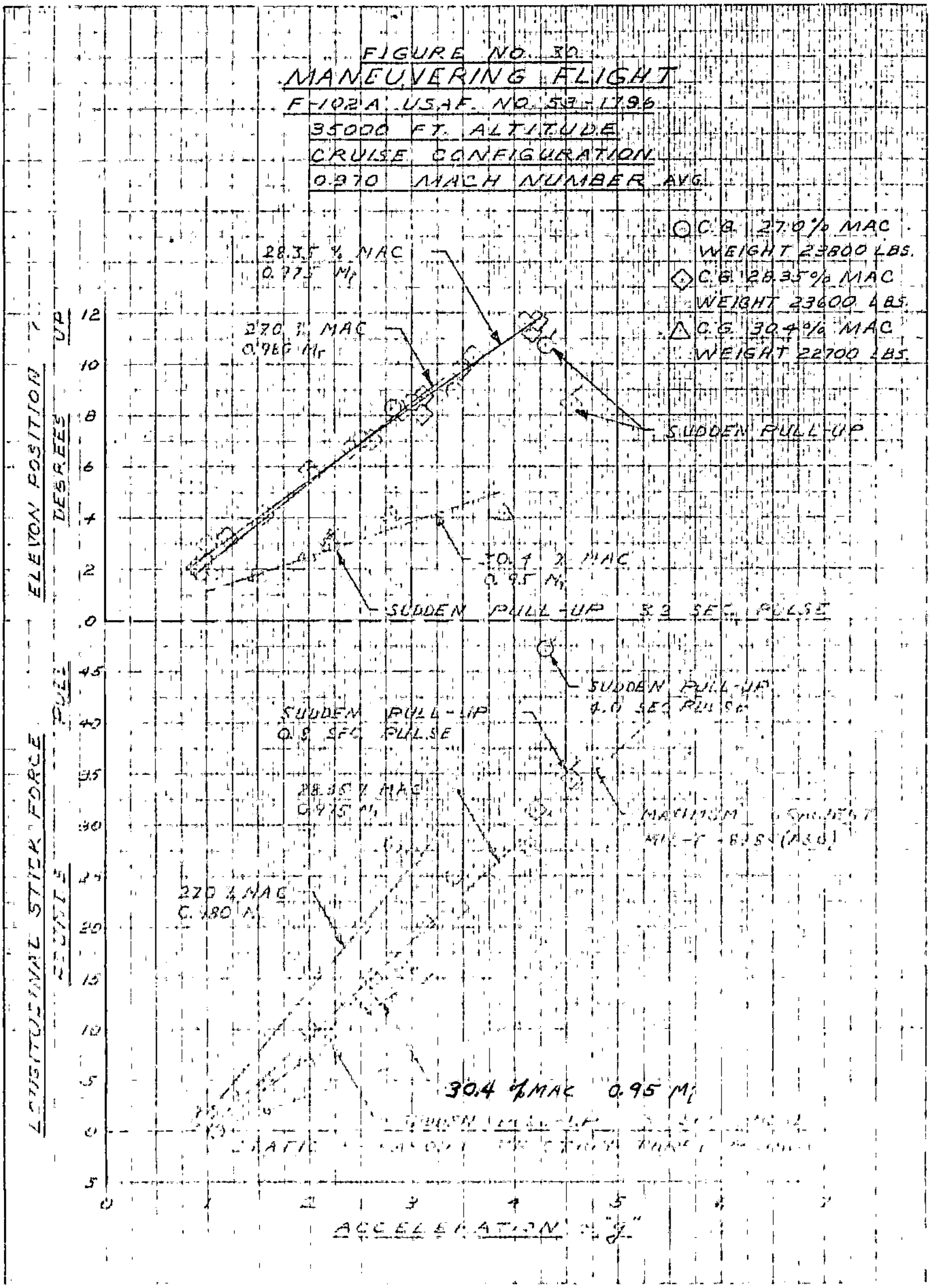
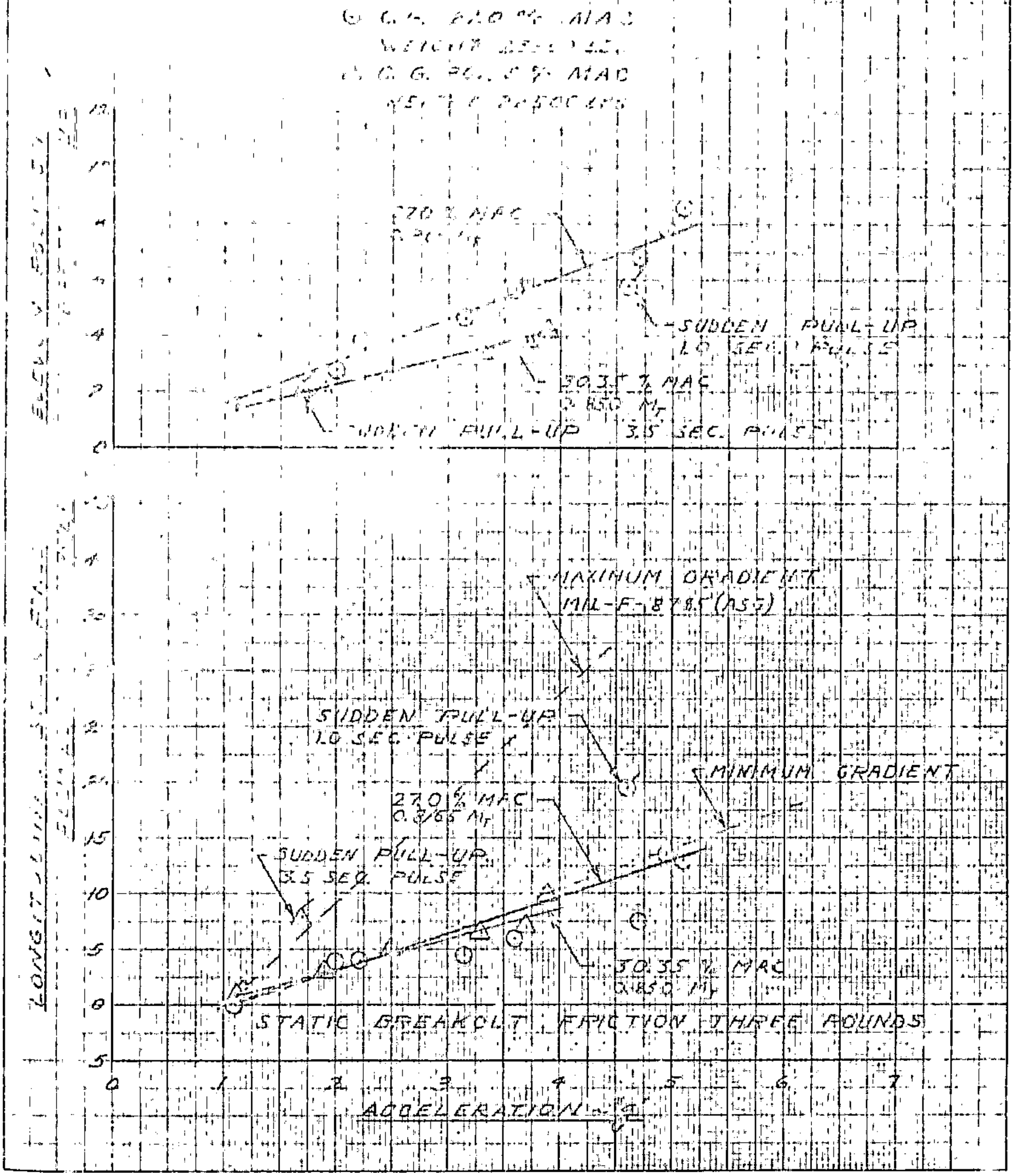


FIGURE NO. 30
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
35000 FT. ALTITUDE
CRUISE CONFIGURATION
0.970 MACH NUMBER AVG.

FIGURE NO. 31
MANEUVERING FLIGHT
F-104A USAF NO. 53-1796
35000 FT ALTITUDE
CRUISE CONFIGURATION
0.855 MACH NUMBER AVG



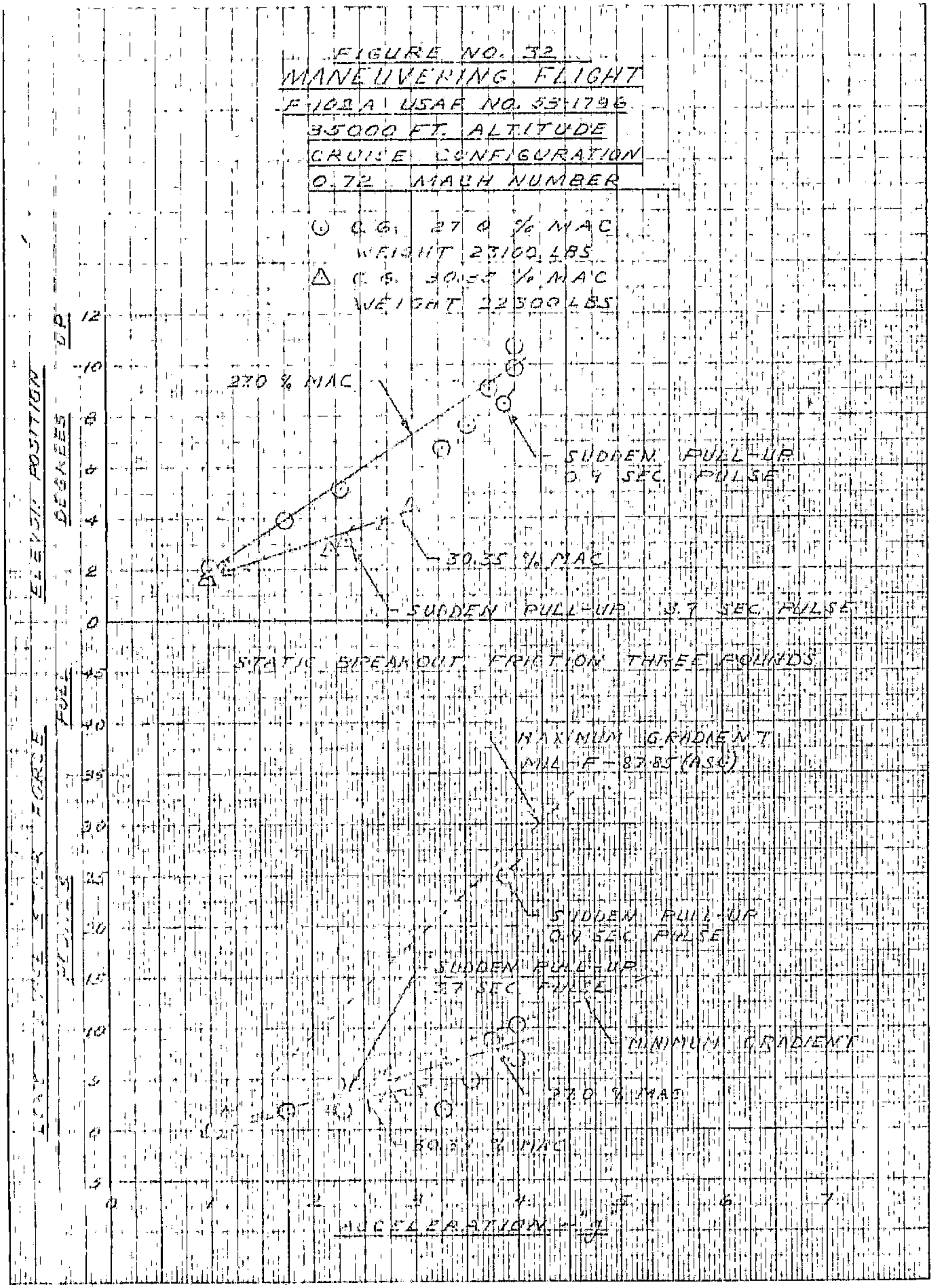
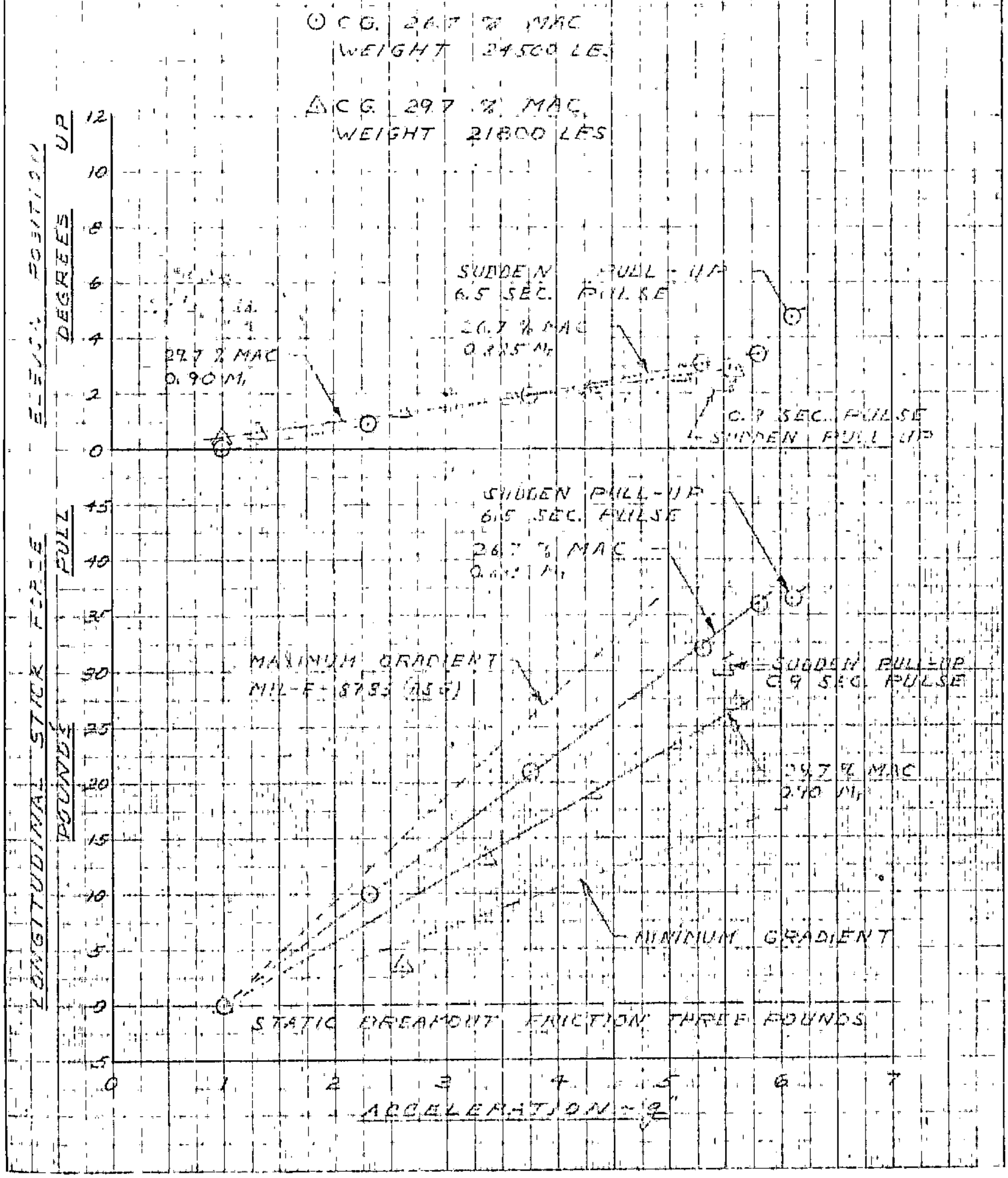


FIGURE NO. 32
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
35000 FT. ALTITUDE
CRUISE CONFIGURATION
0.72 MACH NUMBER

FIGURE NO. 33
MANEUVERING FLIGHT
F-102A USAF NO 53-1796
10000 FT. ALTITUDE
CRUISE CONFIGURATION
0.90 MACH NUMBER AVG



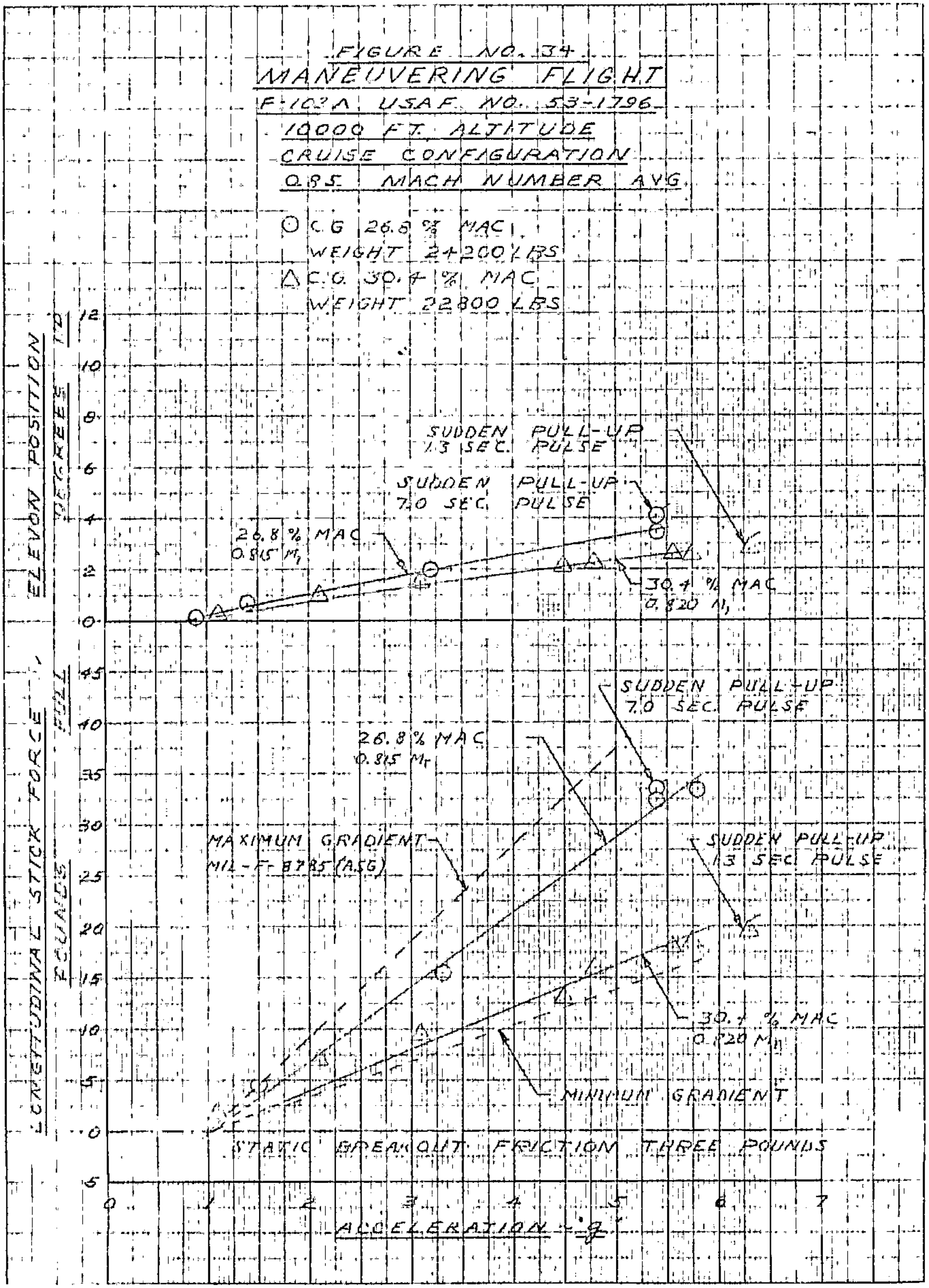


FIGURE NO. 34

MANEUVERING FLIGHT

F-102A USAF NO. 53-1796

10000 FT. ALTITUDE

CRUISE CONFIGURATION

0.85 MACH NUMBER AVG.

○ C.G 26.8 % MAC

WEIGHT 24200 LBS

△ C.G. 30.4 % MAC

WEIGHT 22800 LBS

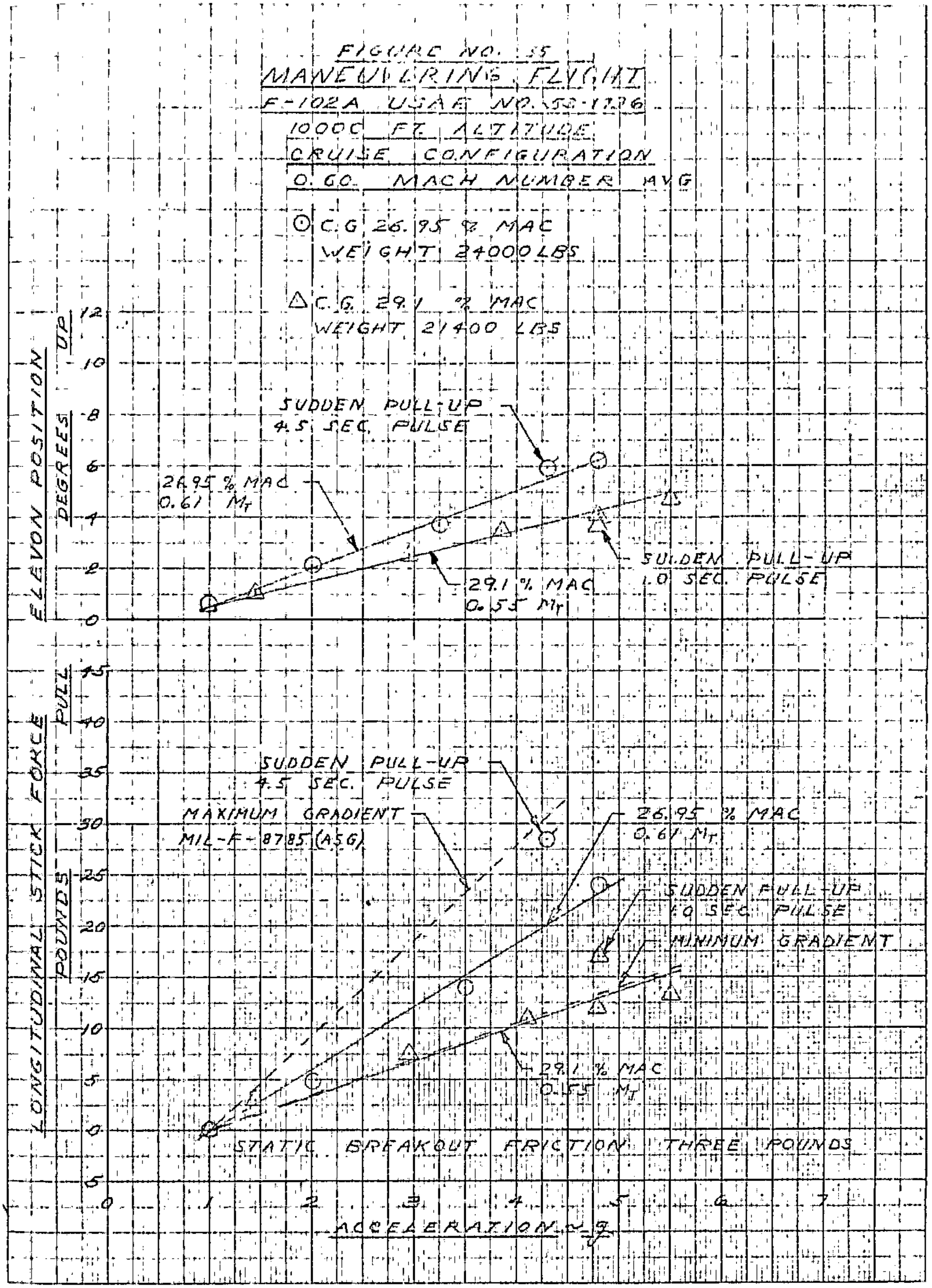


FIGURE NO. 55
MANEUVERING FLIGHT
F-102A USAF NO. 53-1776
10000 FT. ALTITUDE
CRUISE CONFIGURATION
0.60 MACH NUMBER AVG

FIGURE NO. 36

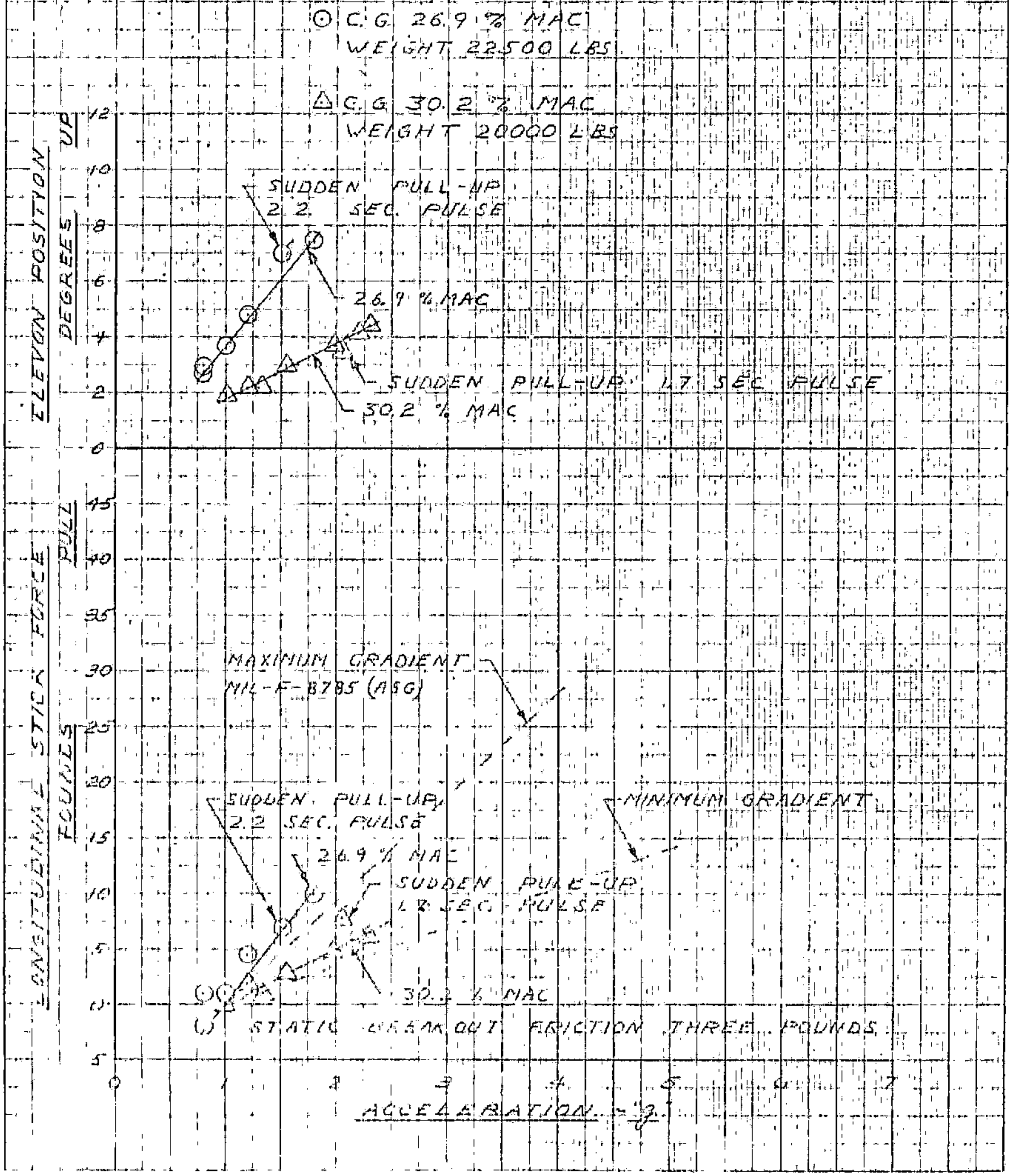
MANEUVERING FLIGHT

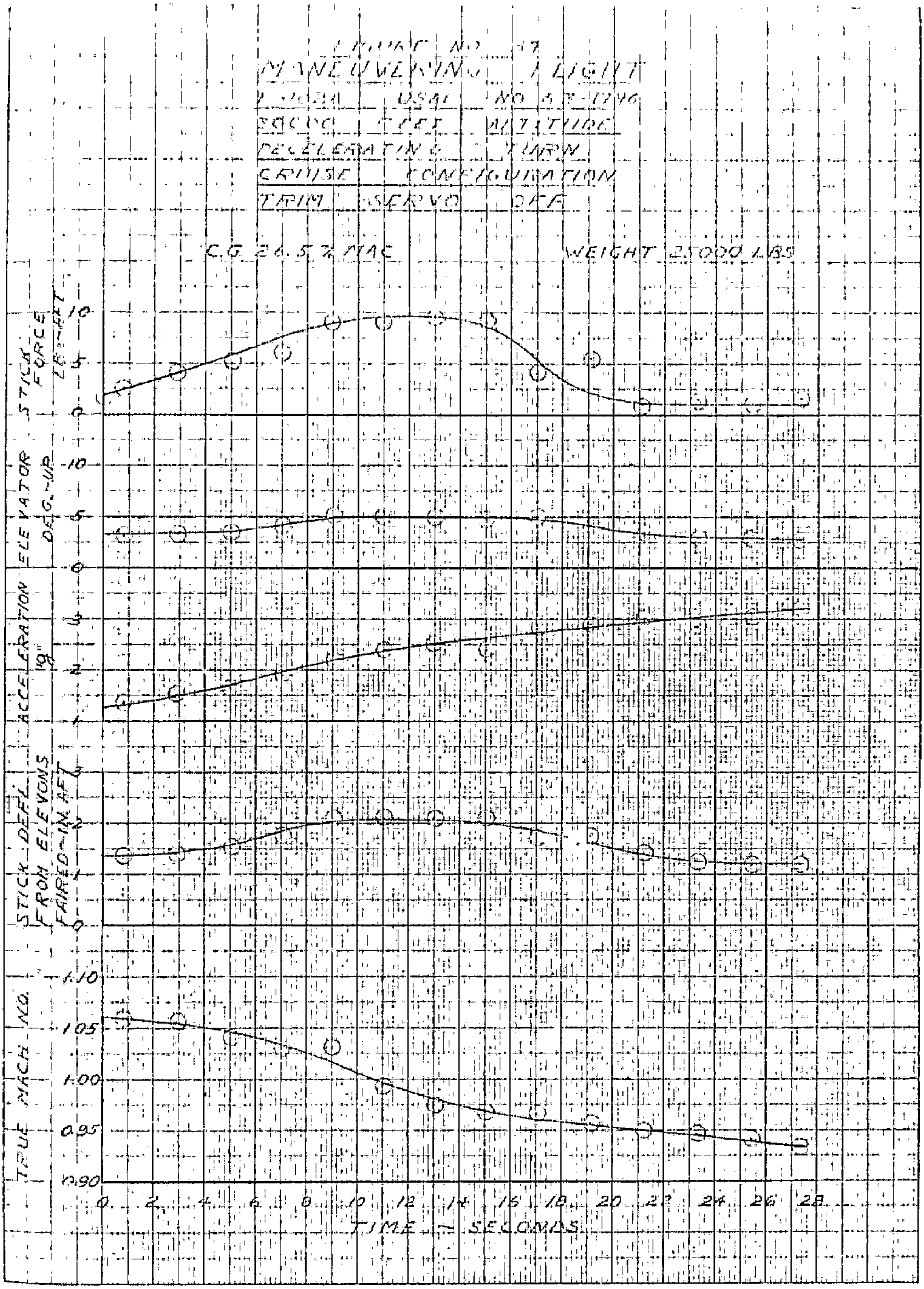
F-102A USAF NO. 53-1706

20000 FT. ALTITUDE

POWER APPROACH CONFIGURATION

180 KNOTS CALIBRATED AIRSPEED





See Fig 30
Forces much lighter here.

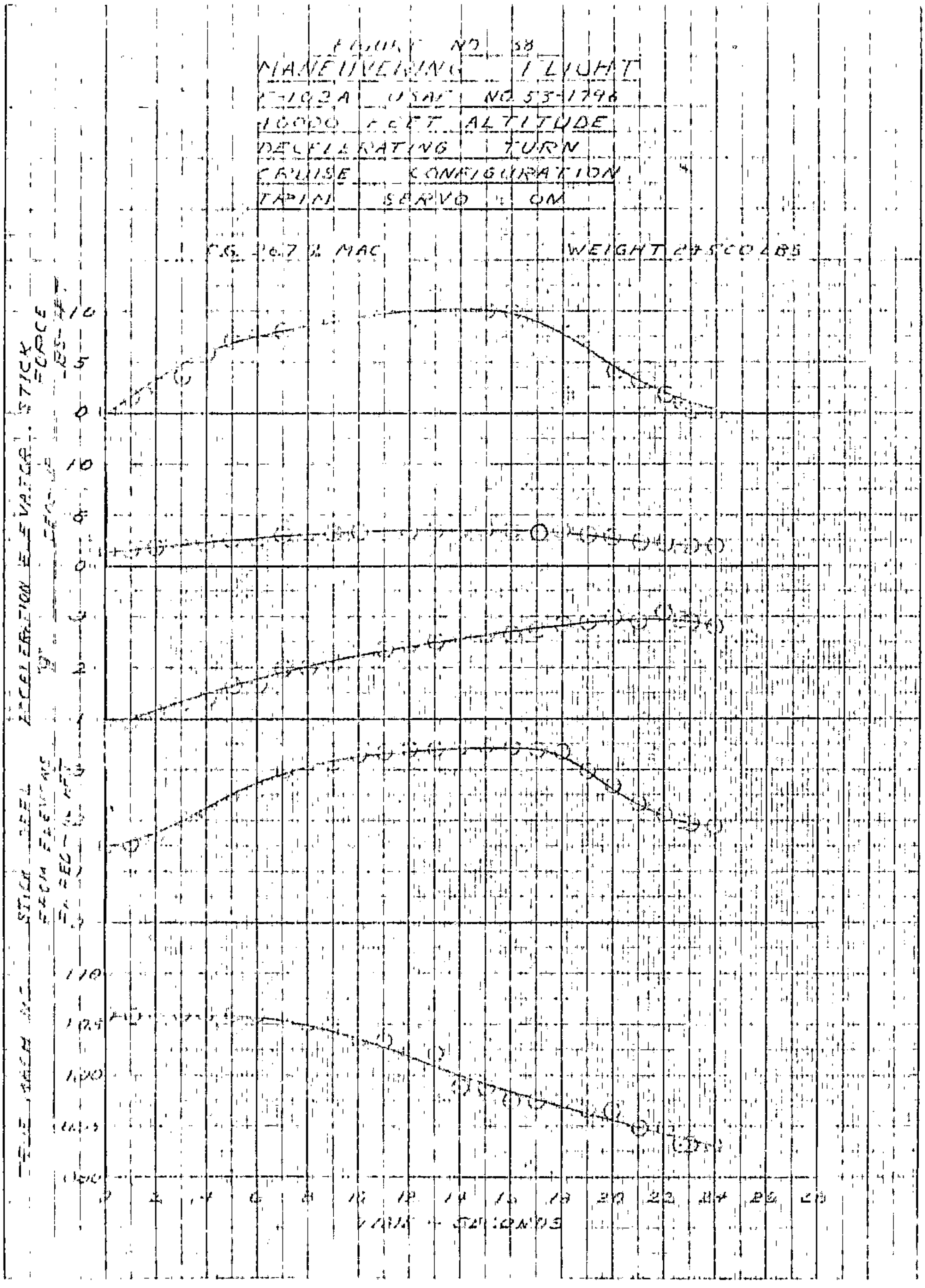


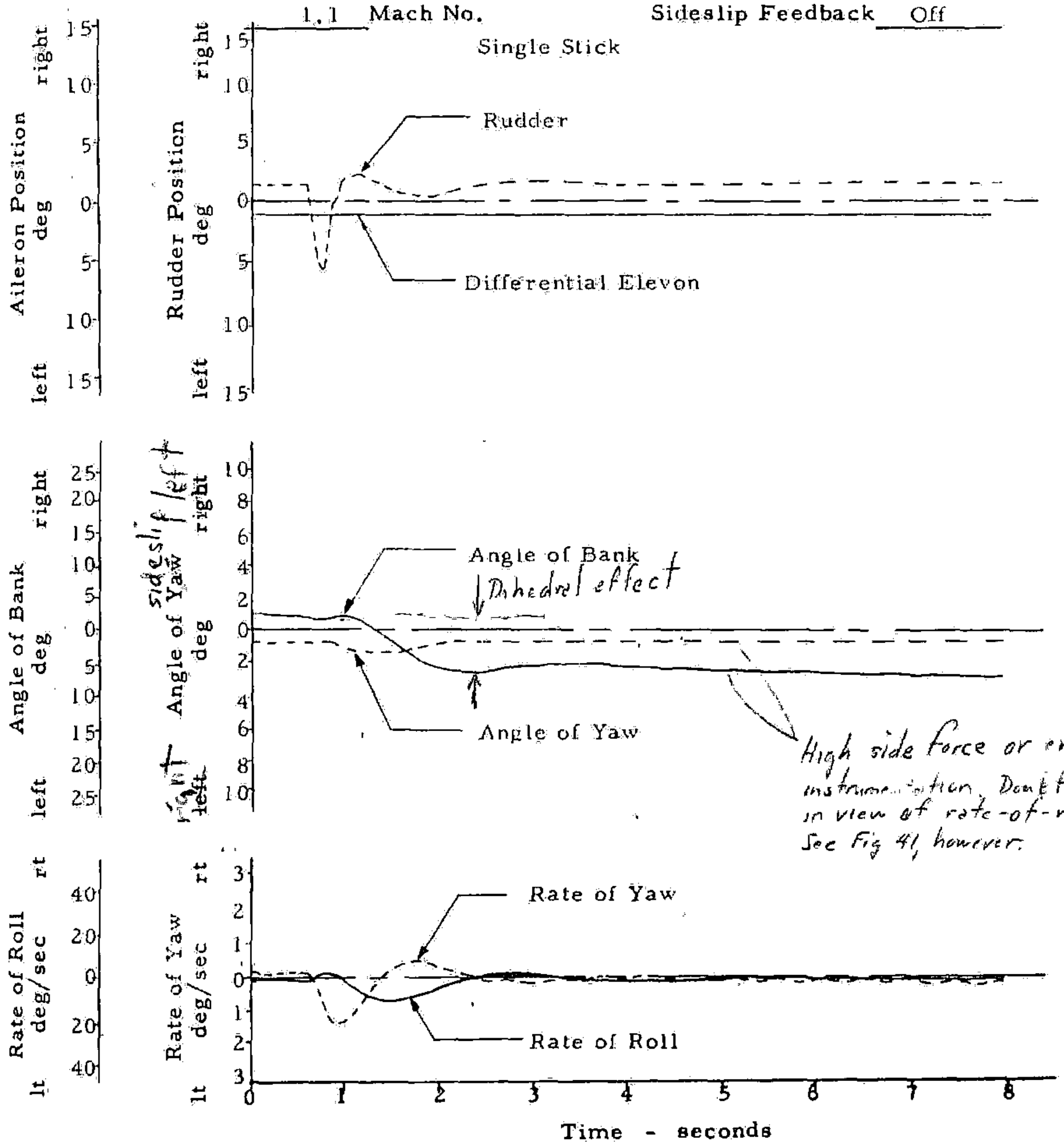
FIGURE NO. 58
MANEUVERING FLIGHT
F-102A USAF NO. 53-1796
10000 FEET ALTITUDE
DECELERATING TURN
CRUISE CONFIGURATION
TAPIN SERVO ON

Figure No. 39

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
45,000 Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off



High side force or erroneous instrumentation. Don't letter in view of rate-of-roll below. See Fig 41, however.

APPENDIX I

Figure No. 40

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
45,000 Altitude
0.77 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

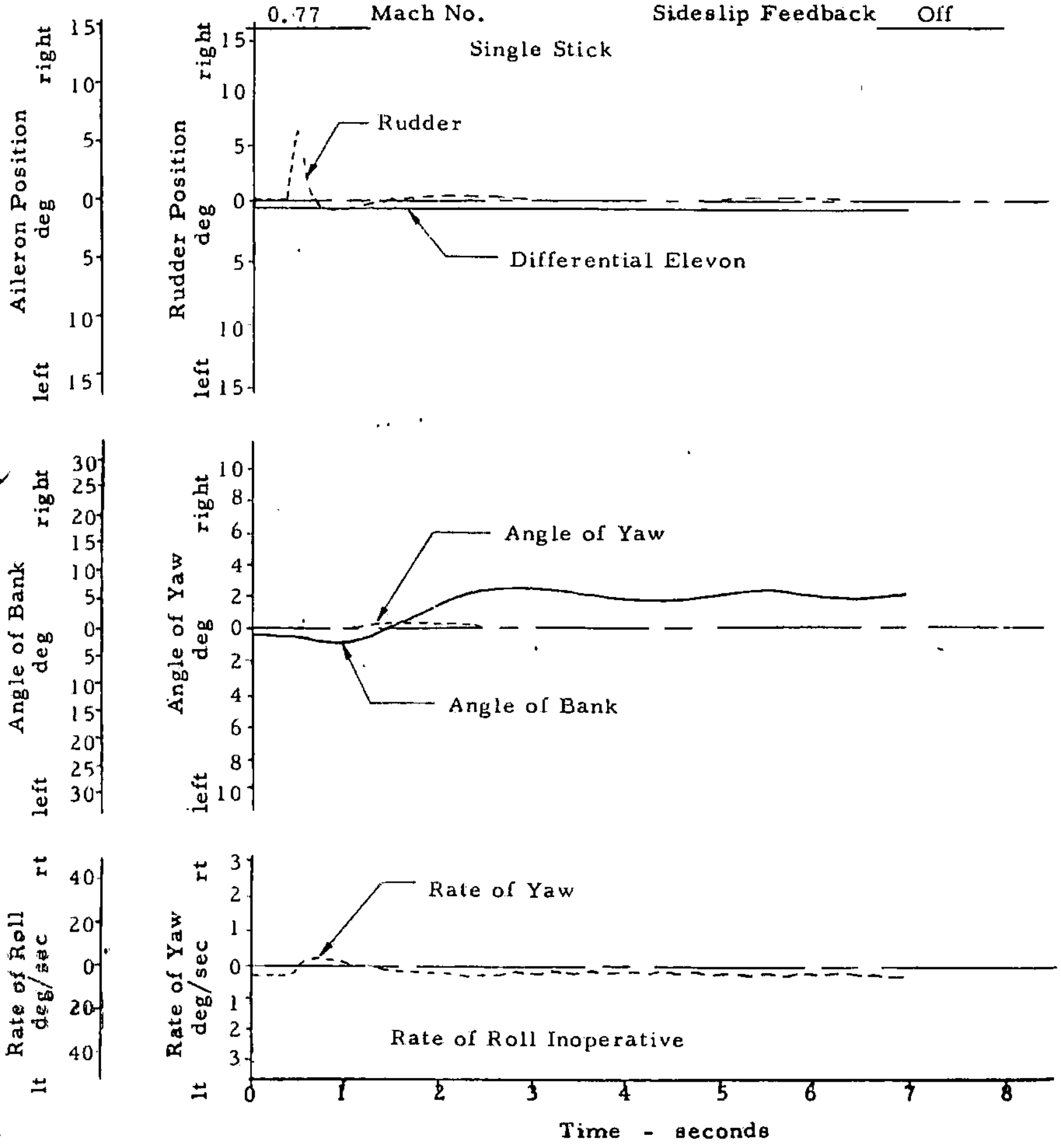
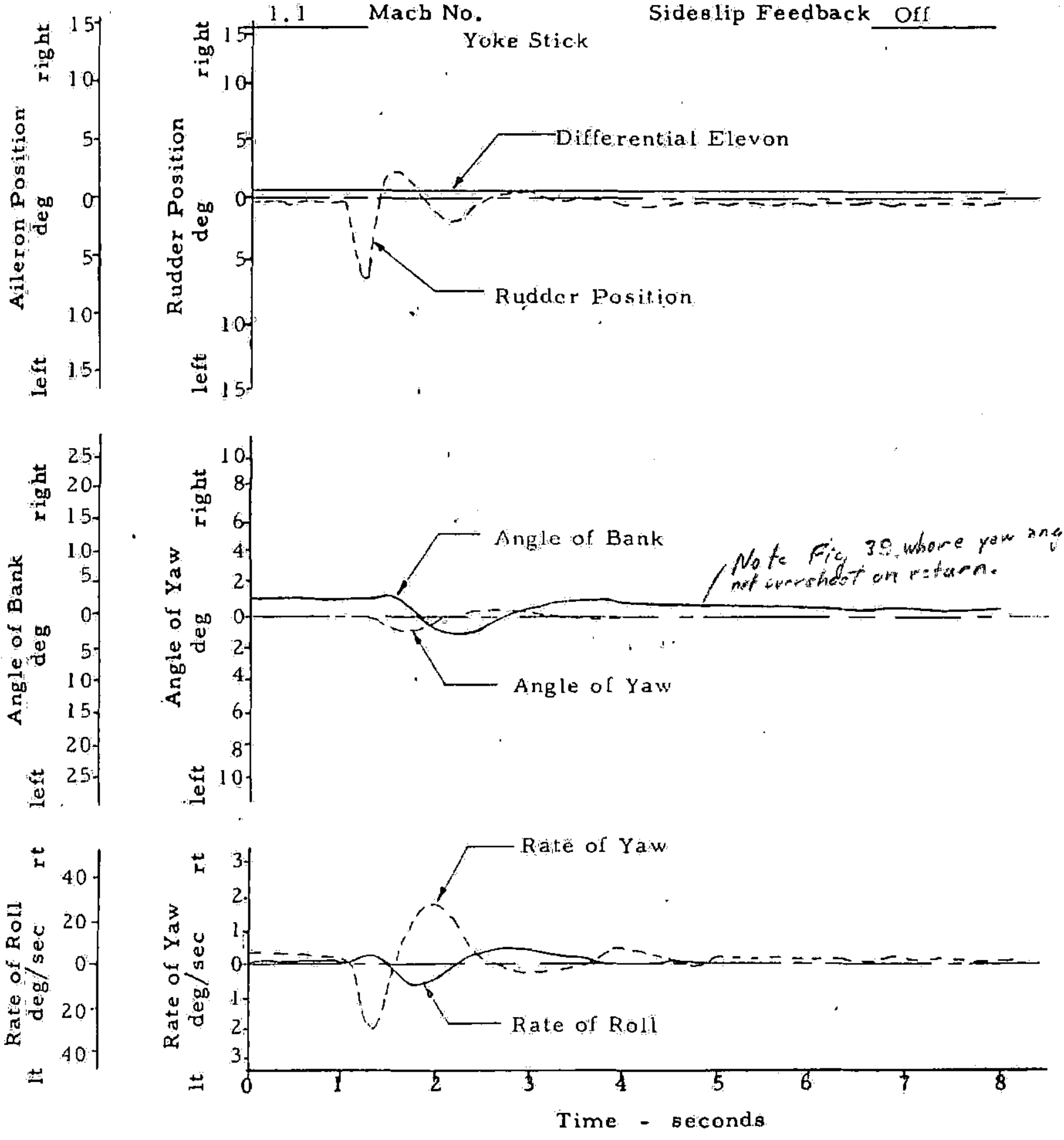


Figure No. 41

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off



Note Fig 39, where yaw angle did not overshoot on return.

Figure No. 42

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000' Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

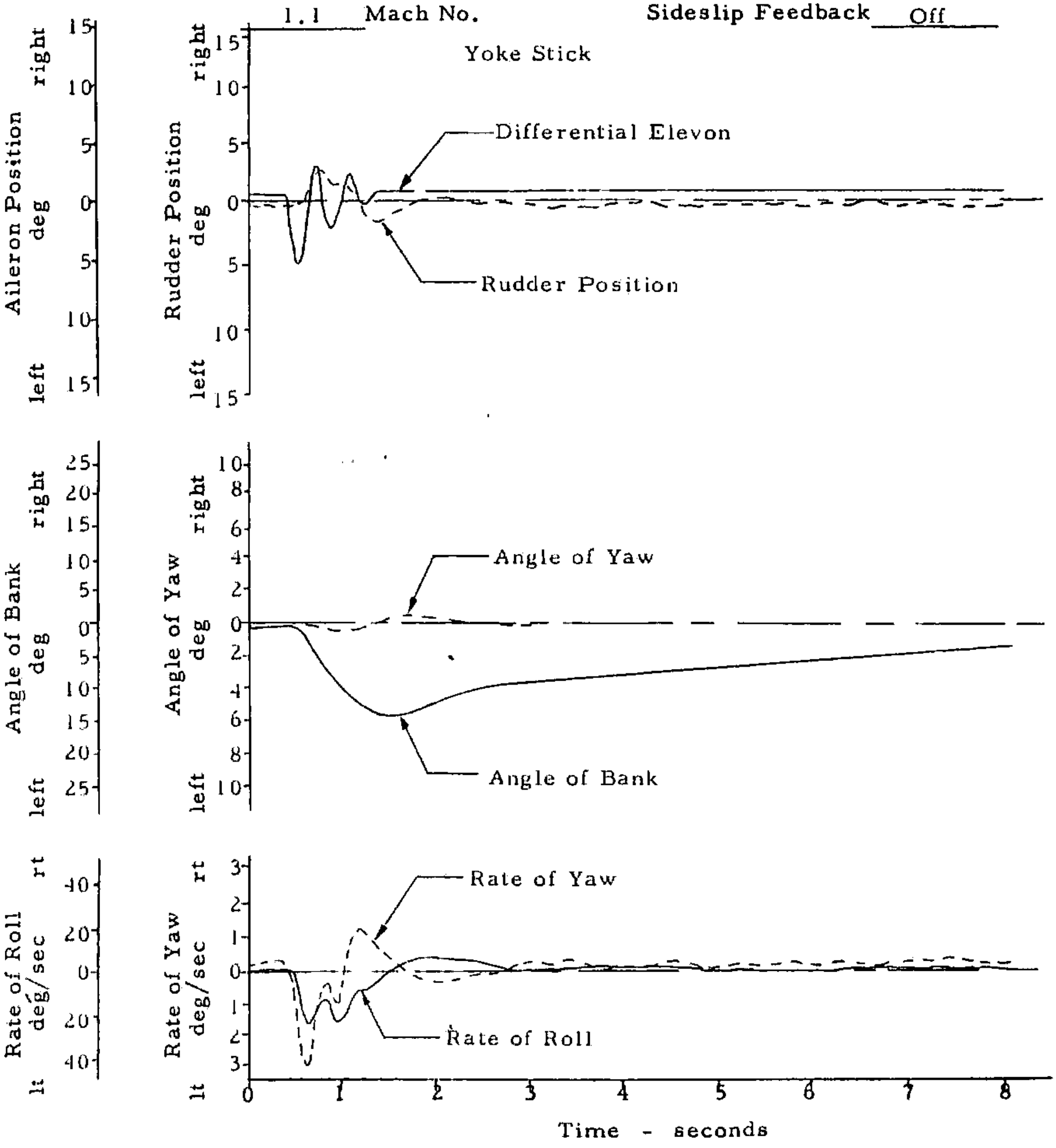


Figure No. 43

DYNAMIC LATERAL DIRECTIONAL STABILITY

F-102A USAF No. 53-1796

Combat Configuration

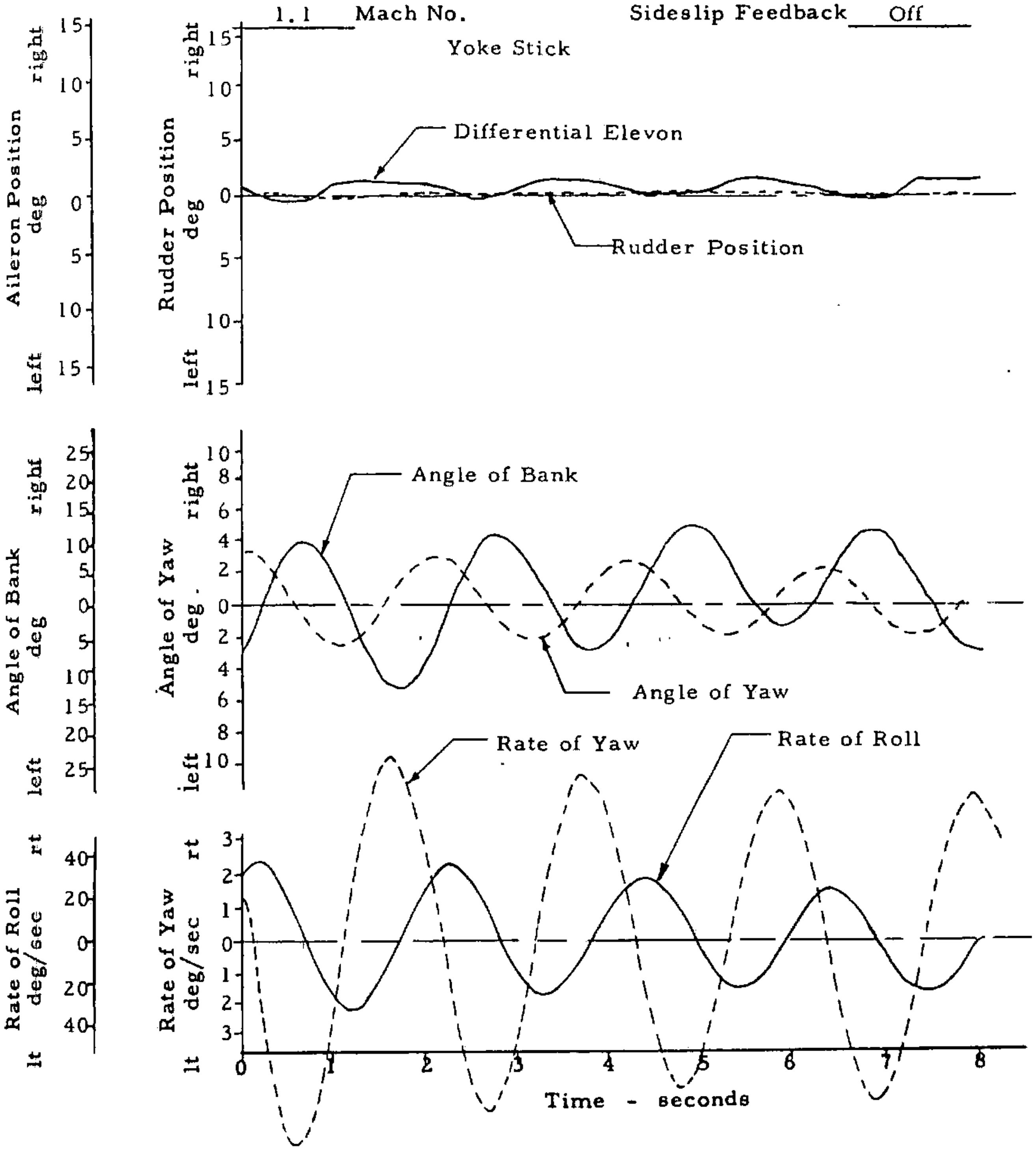
35,000 Altitude

1.1 Mach No.

Yaw Damper Off

Turn Coordinator Off

Sideslip Feedback Off



From Fig	Input	Result
1) 44	5° L.R.	1.5°/sec. L yaw rate
2). 46	11° R.A.	6°/sec. R yaw rate
3) 45	8° R.A + 4° L.R.	3°/sec. R. yaw rate

From (1)

4) 4° L.R. should produce $\left(\frac{4}{5}\right)\left(\frac{1.5^\circ}{sec}\right) = 1.2^\circ/sec$ L yaw

From (2)

5) 8° R.A should produce $\left(\frac{8}{11}\right)(6^\circ/sec) = 4.3^\circ/sec$ R yaw.

Combining (4) + (5)

6) 8° R.A + 4° L.R should produce

$$\frac{4.3^\circ}{sec} \text{ Right} + \frac{1.2^\circ}{sec} \text{ Left} = \frac{3.1^\circ}{sec.} \text{ right.}$$

This compares well with (3)

I Airplane has favorable yaw with aileron deflection (too much really)

II a/c has positive dihedral effect with yaw.

Figure No. 44

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator On
Sideslip Feedback Off

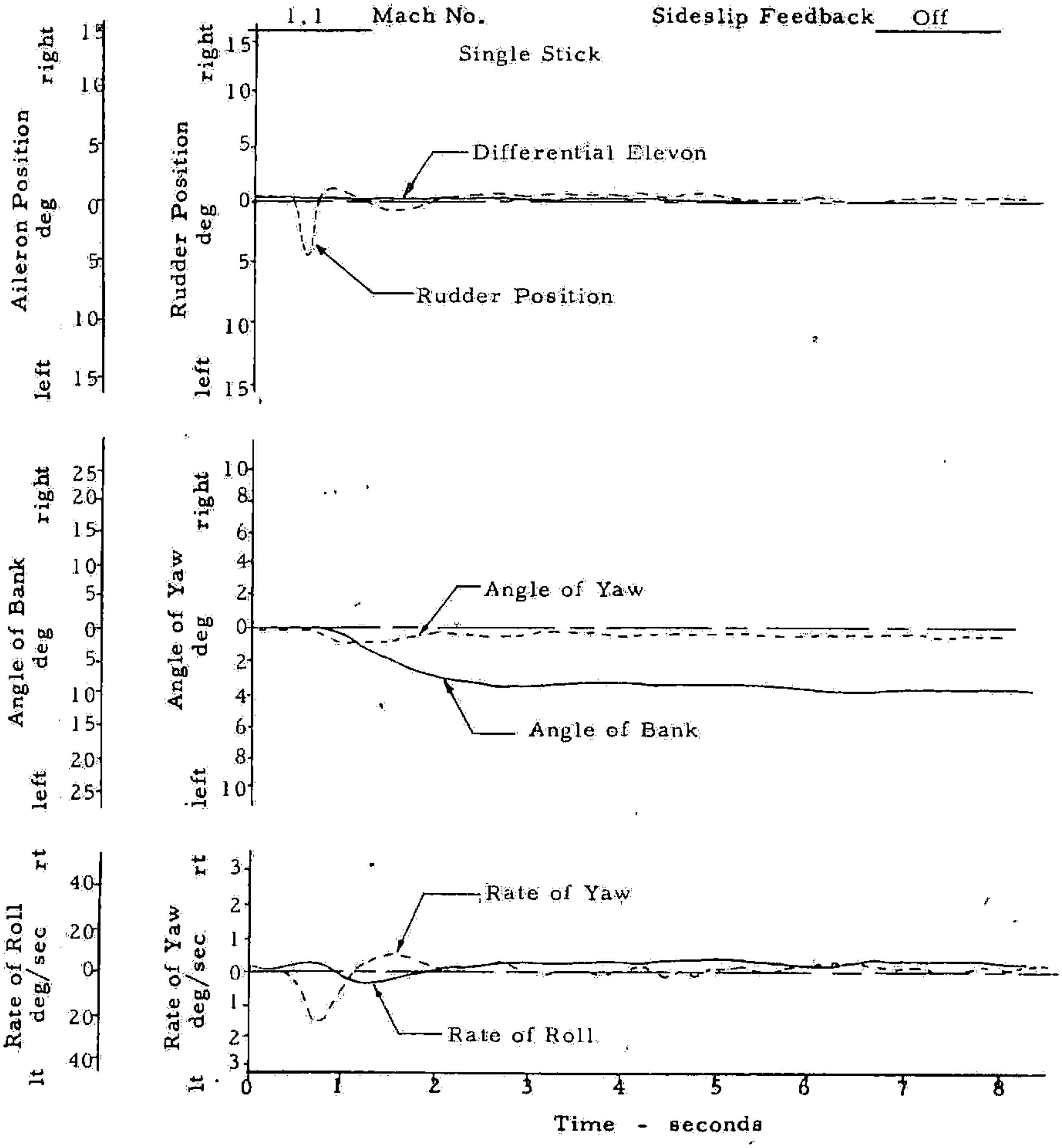


Figure No. 45

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper On
Turn Coordinator On
Sideslip Feedback Off

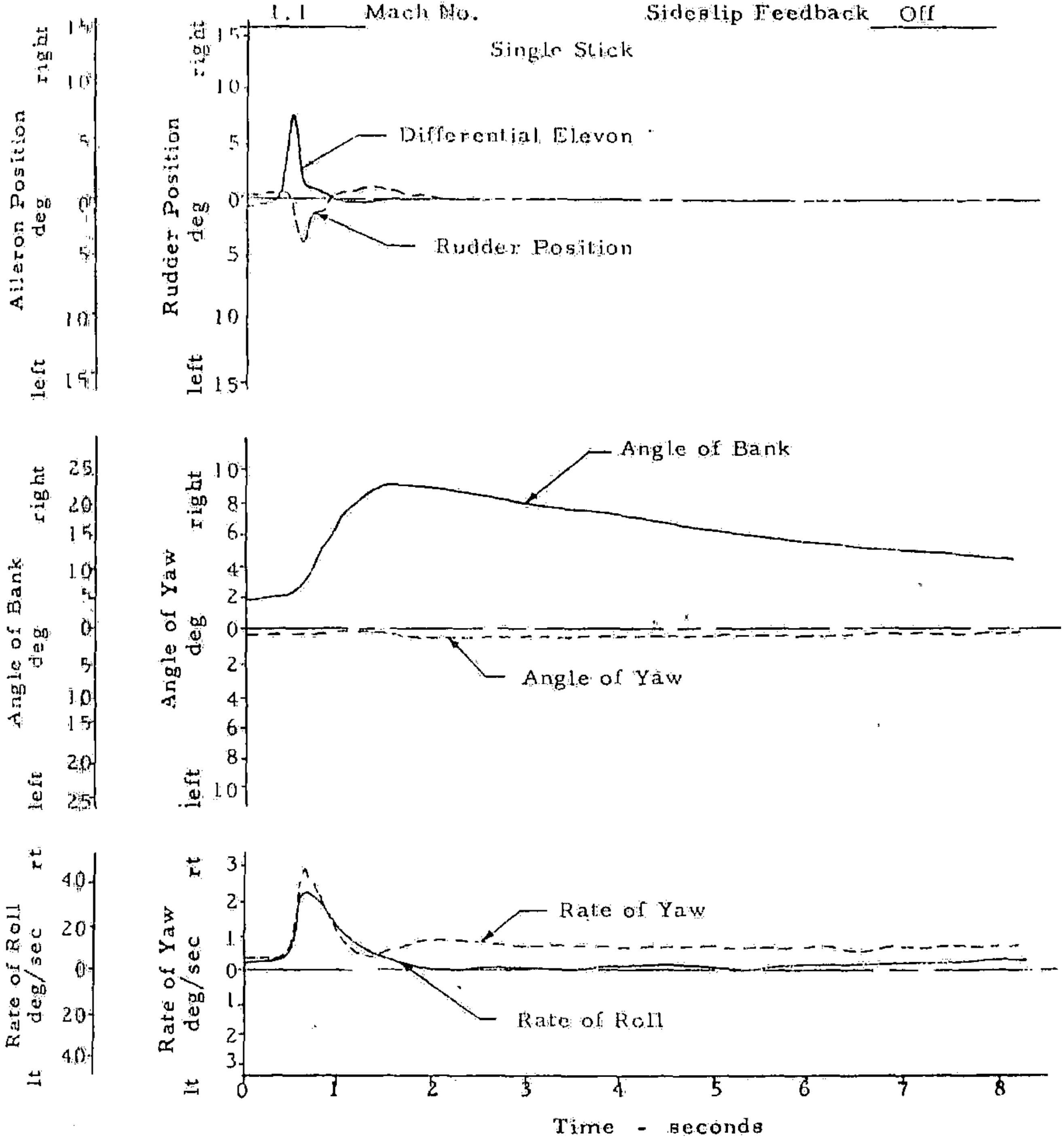


Figure No. 46

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Combat Configuration
35,000 Altitude
1.1 Mach No.

Yaw Damper Off
Turn Coordinator Off
Sideslip Feedback Off

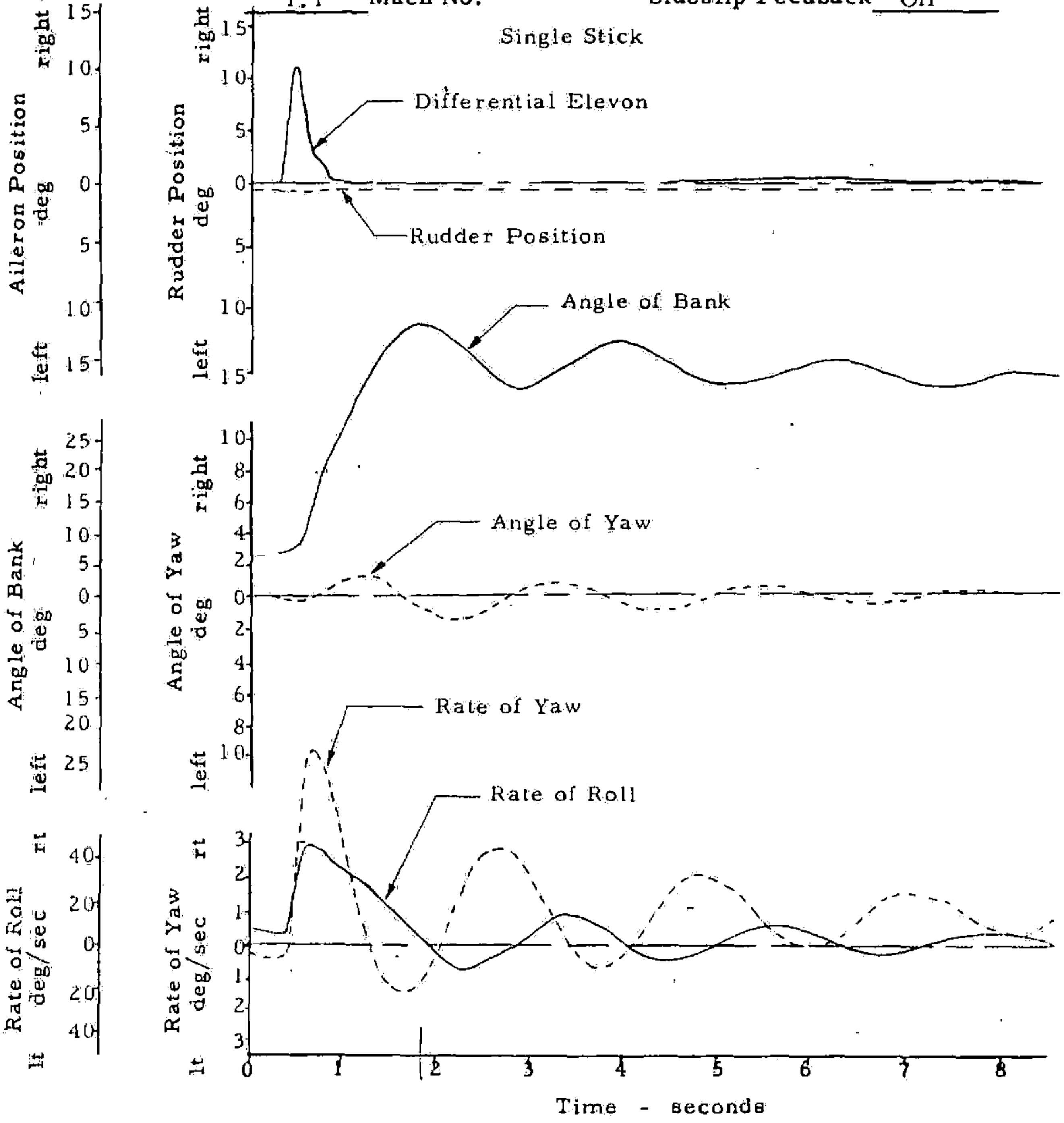


Figure No. 47

DYNAMIC LATERAL DIRECTIONAL STABILITY

F-102A USAF No. 53-1796

Cruise Configuration

35,000 Altitude

0.9 Mach No.

Yaw Damper On

Turn Coordinator Off

Sideslip Feedback Off

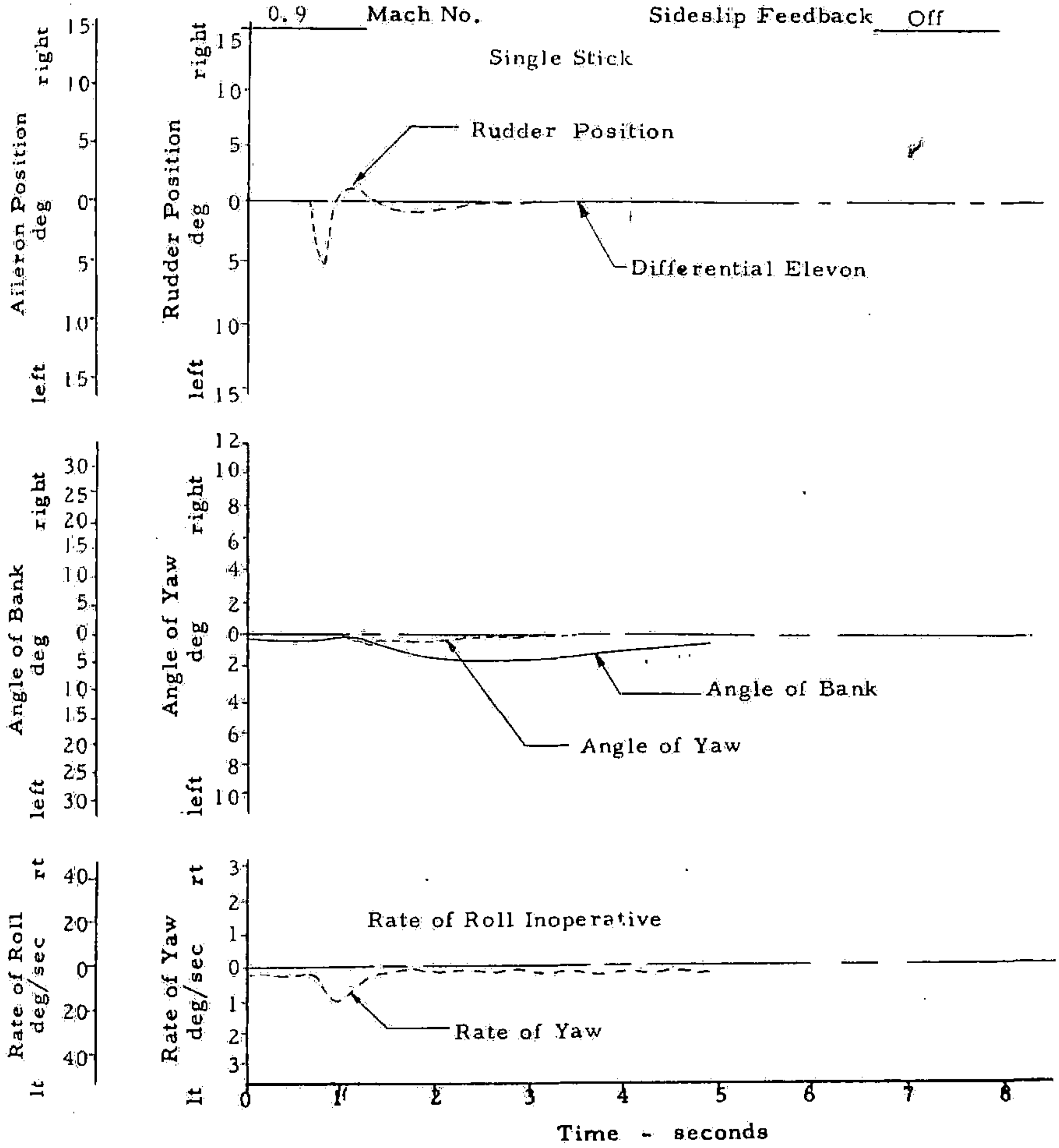


Figure No. 48

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration

35,000 Altitude

0.7 Mach No.

Yaw Damper On

Turn Coordinator Off

Sideslip Feedback Off

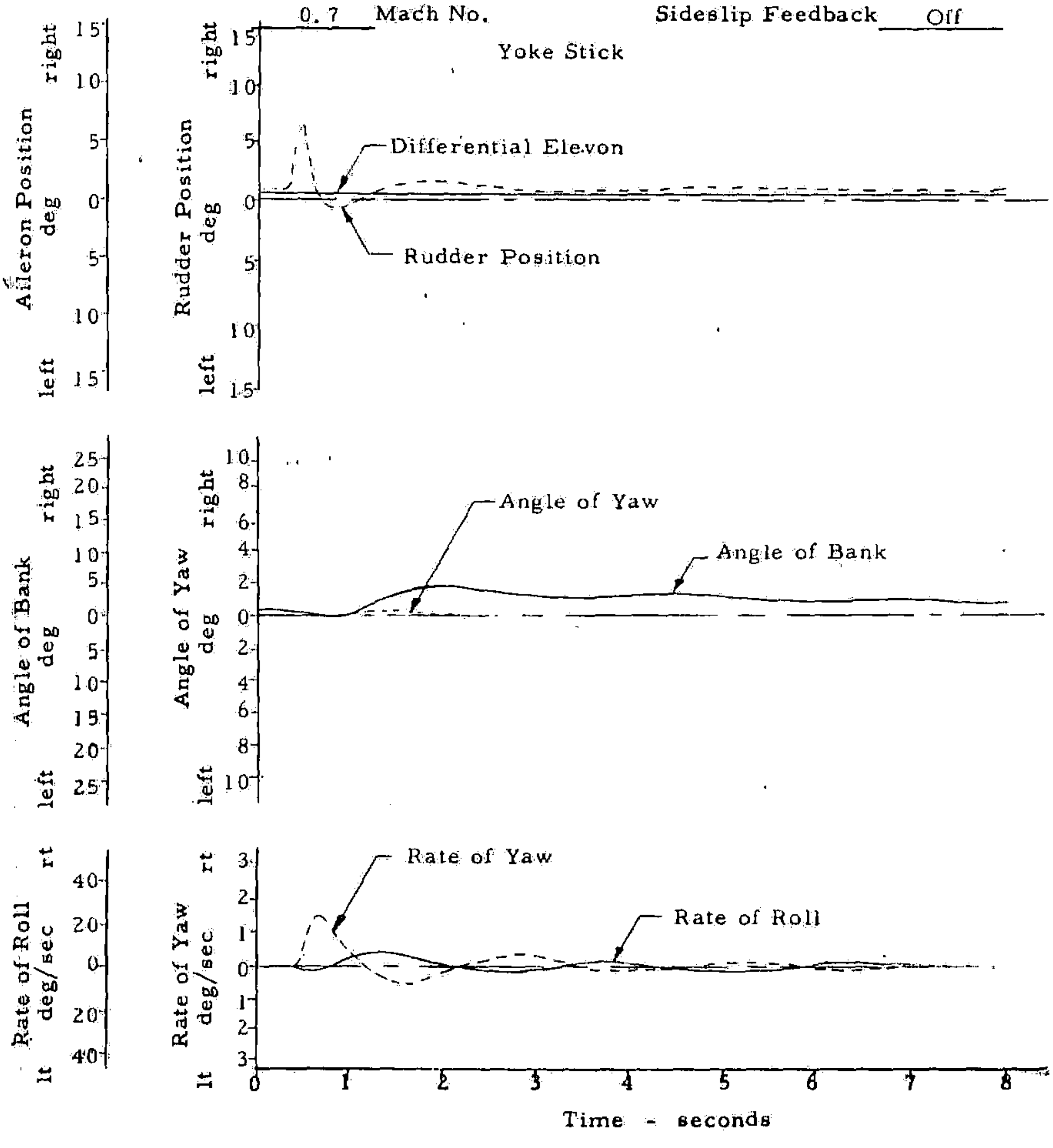
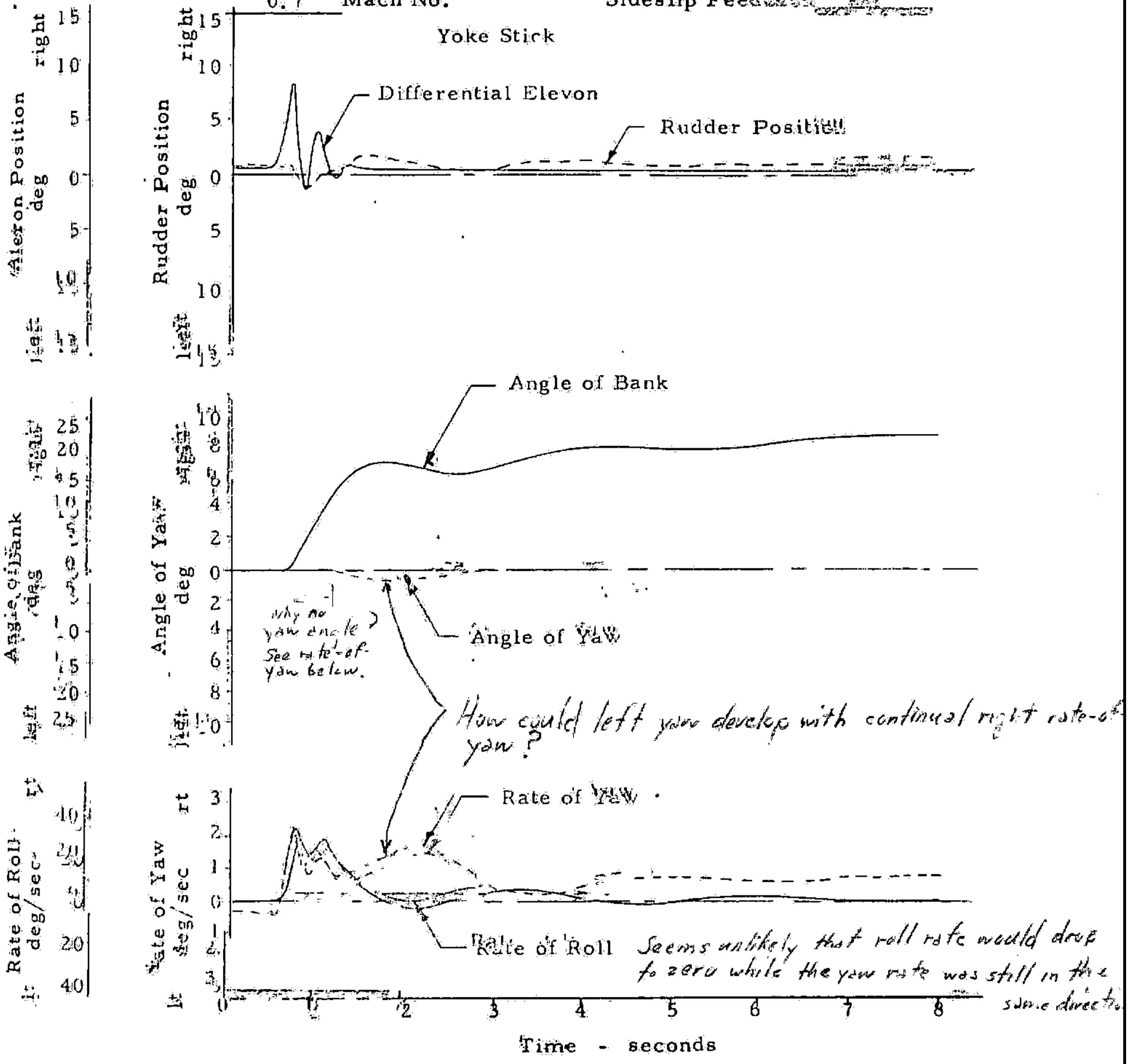


Figure No. 49

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35,000 Altitude
0.7 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off



Why no yaw angle? See rate-of-yaw below.

How could left yaw develop with continual right rate-of
yaw?

Seems unlikely that roll rate would drop to zero while the yaw rate was still in the same directi

Figure No. 50

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
35,000' Altitude
0.7 Mach No.

Yaw Damper Off
Turn Coordinator Off
Sideslip Feedback Off

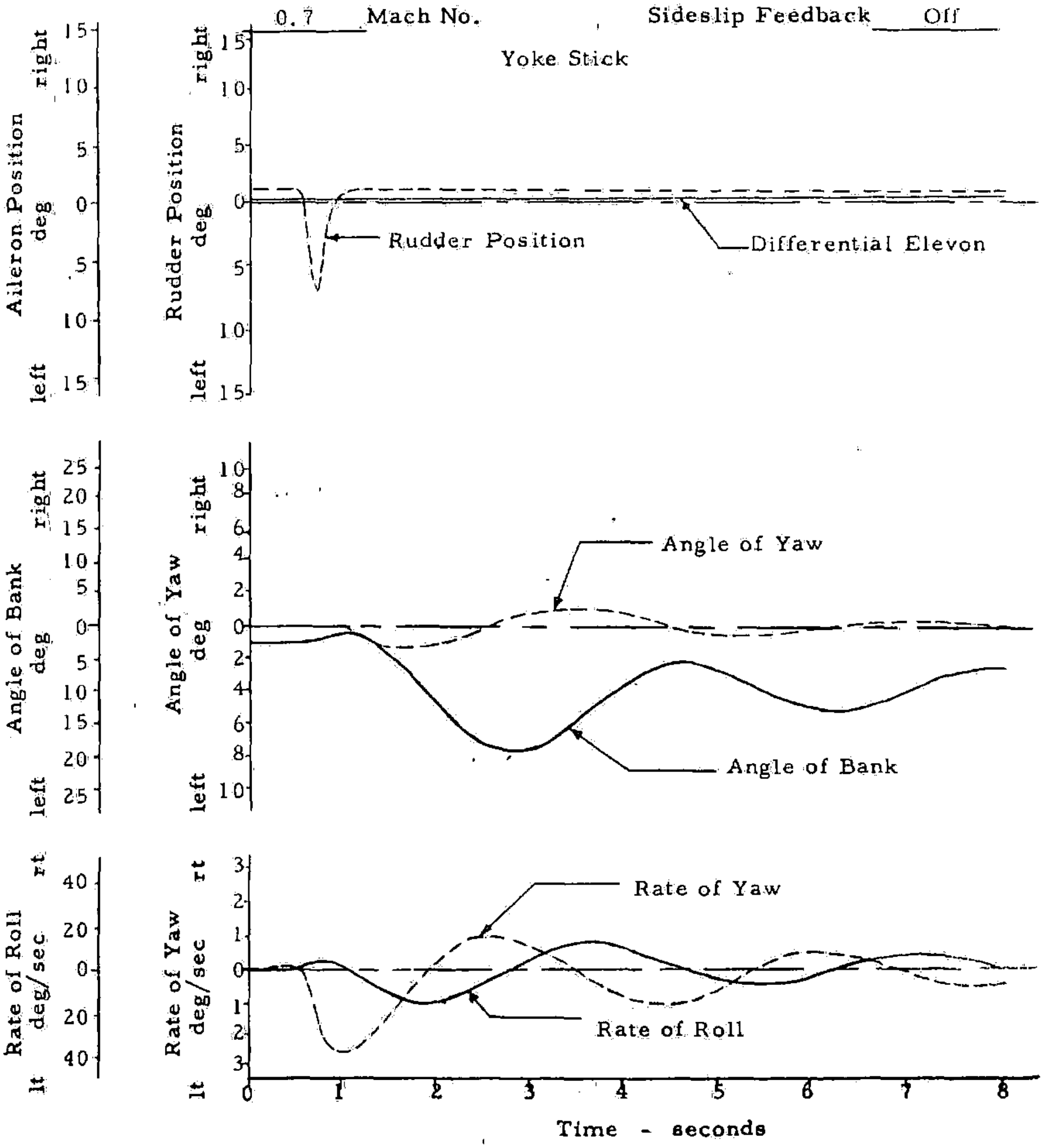


Figure No. 51

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
10,000 Altitude
0.7 Mach No.

Yaw Damper On
Turn Coordinator Off
Sideslip Feedback Off

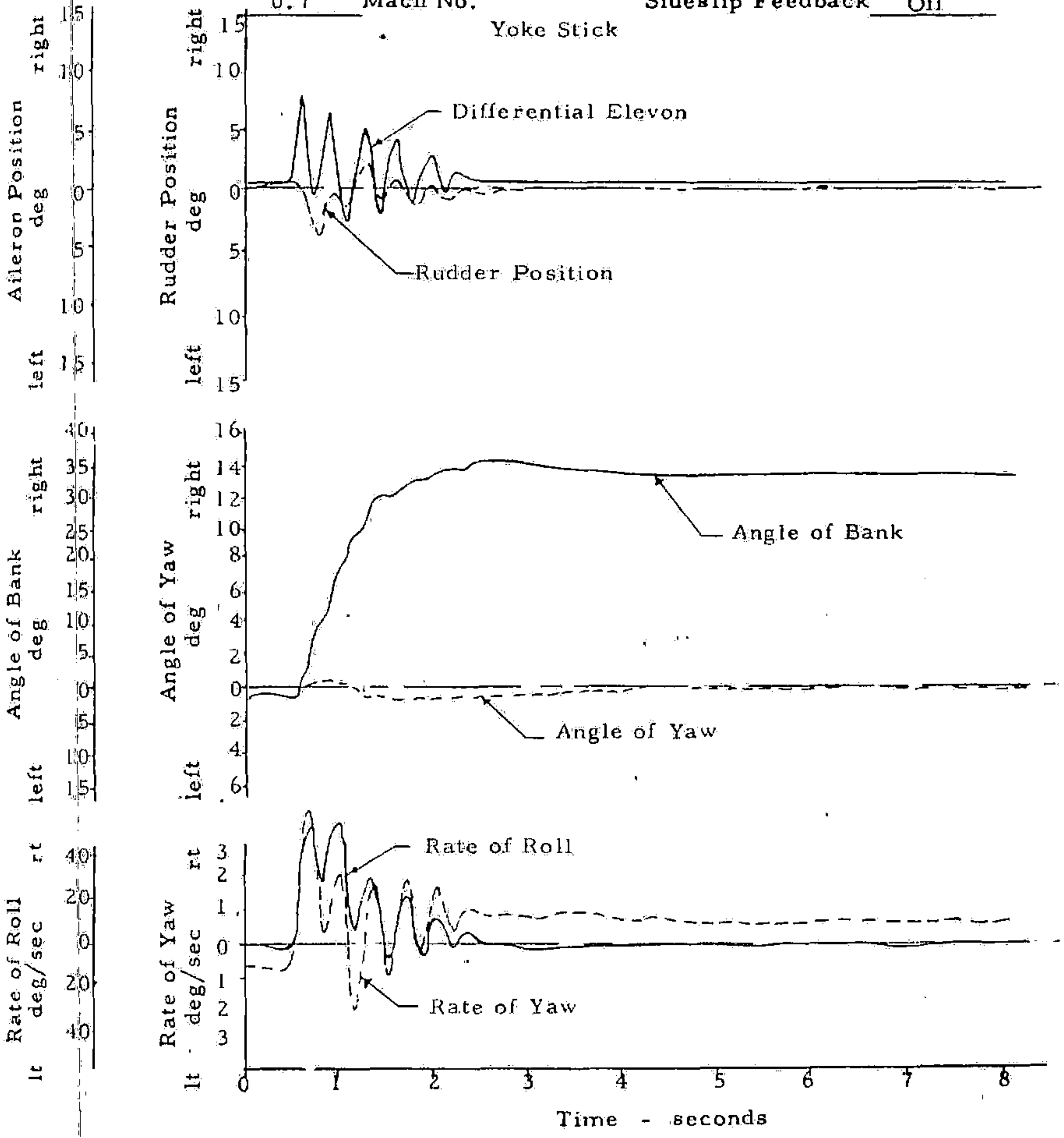


Figure No. 52

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
10,000 Altitude
0.5 Mach No.
Yaw Damper On
Turn Coordinator On
Sideslip Feedback Off

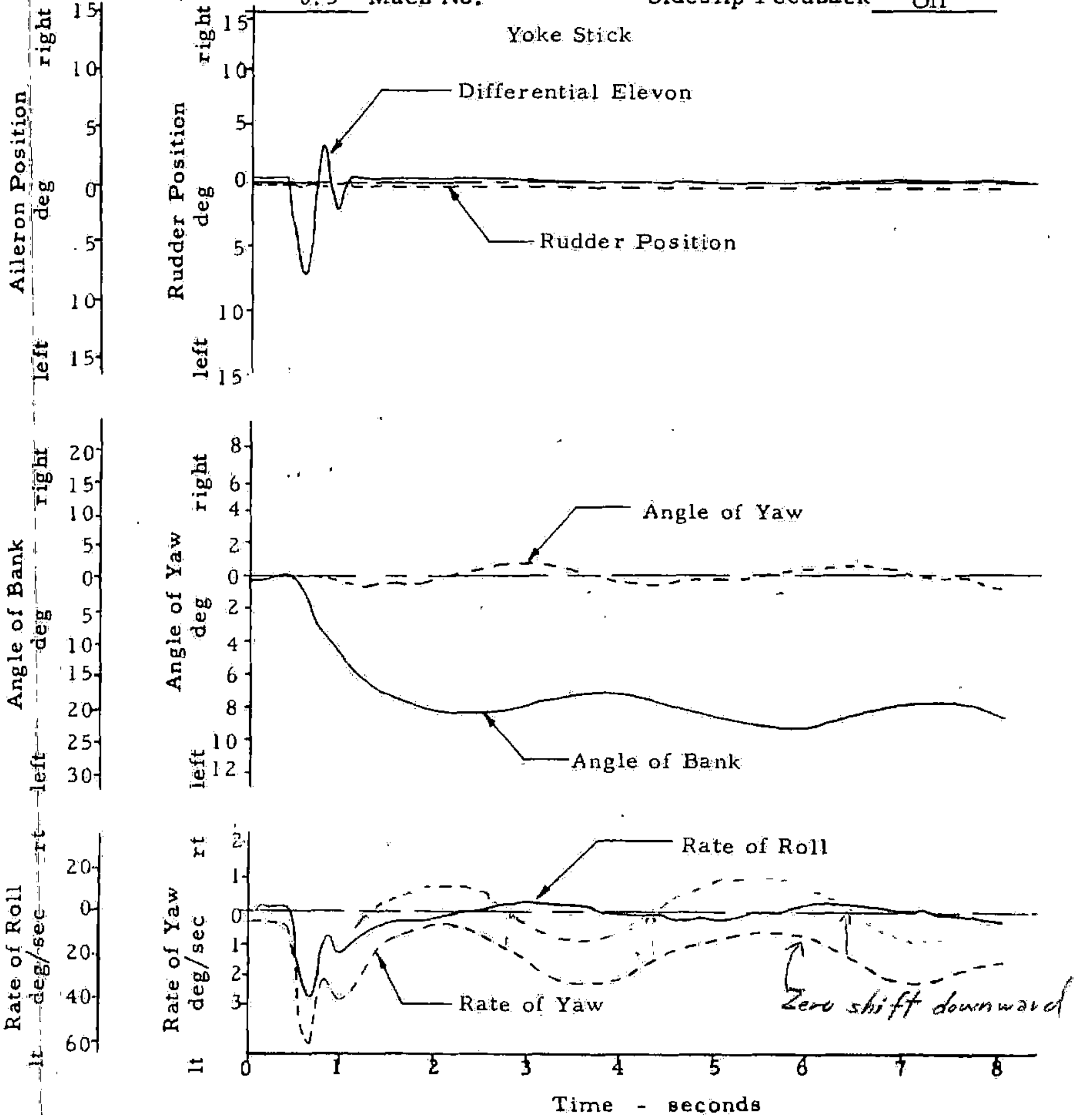


Figure No. 53

DYNAMIC LATERAL DIRECTIONAL STABILITY
F-102A USAF No. 53-1796

Cruise Configuration
10,000 Altitude
0.5 Mach No.
Yaw Damper On
Turn Coordinator On
Sideslip Feedback Off

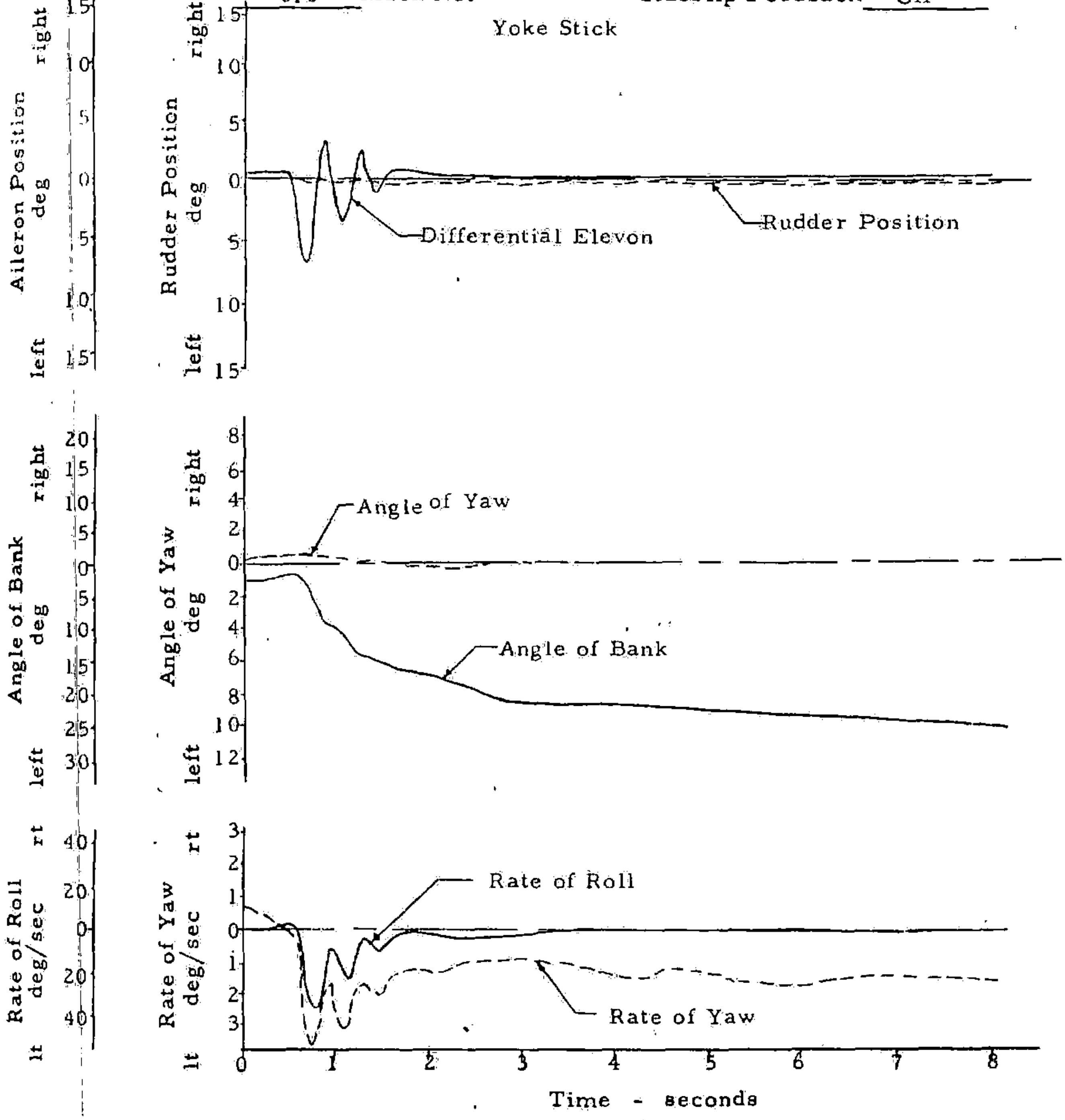


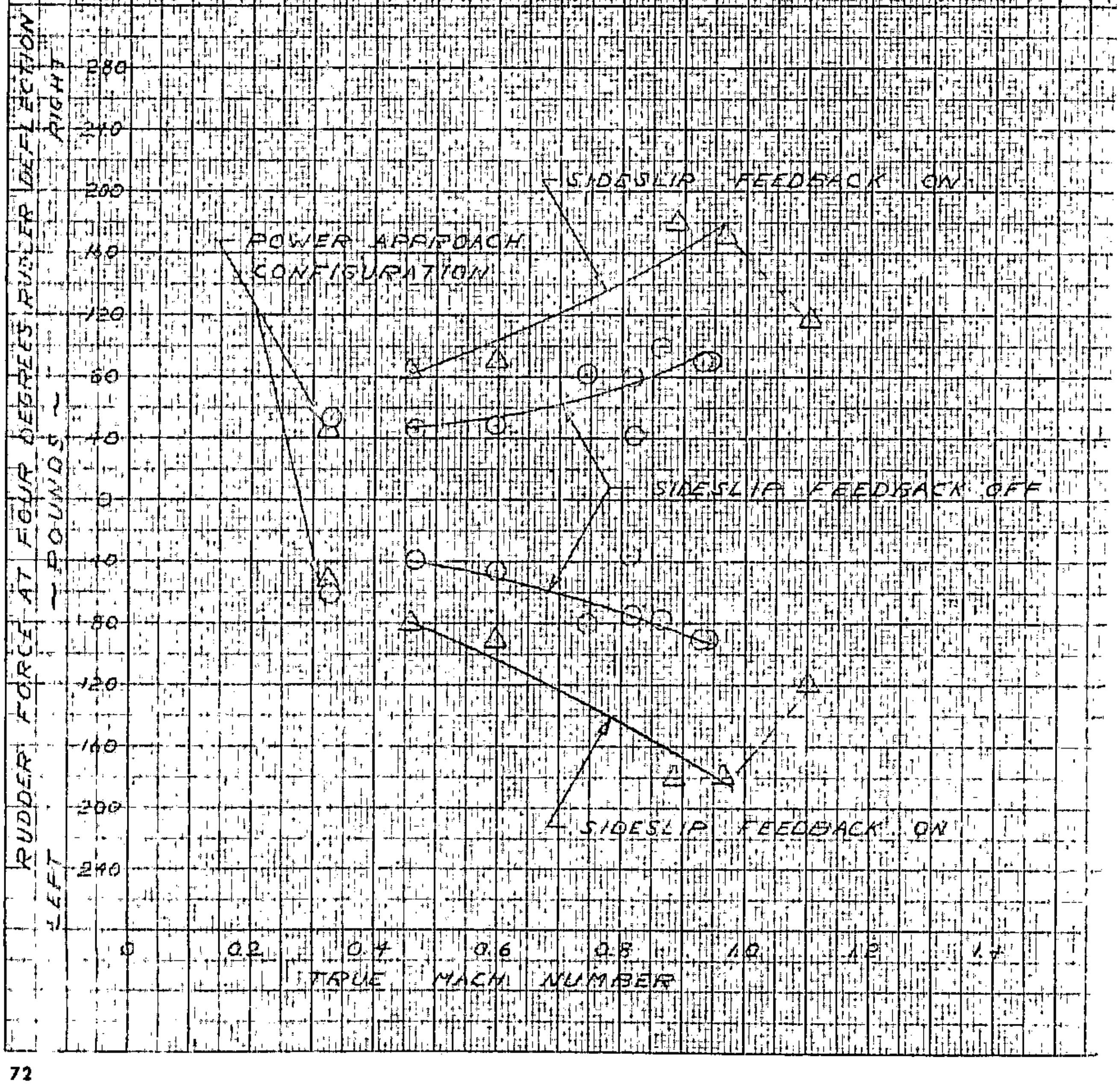
FIGURE NO. 54

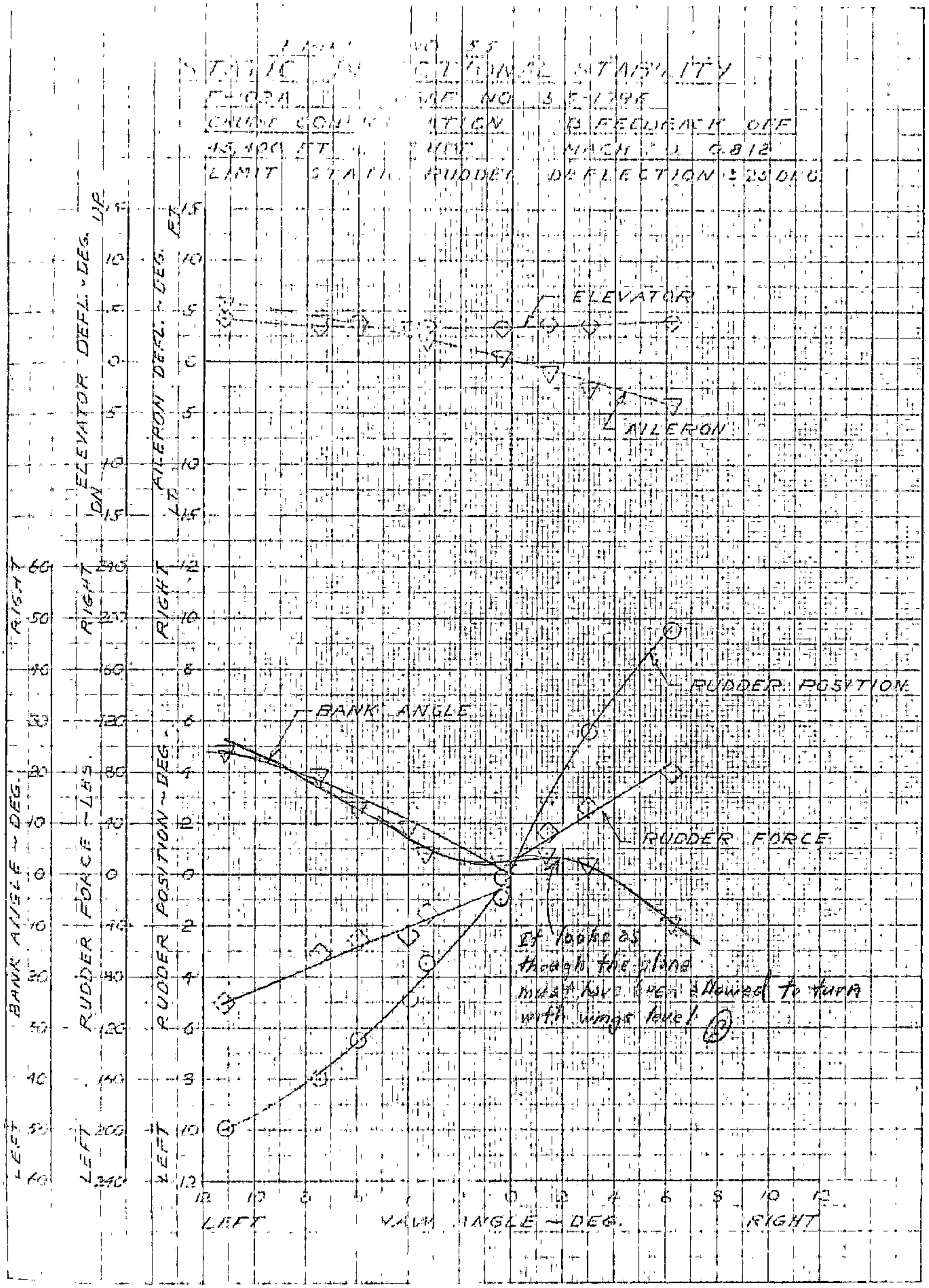
STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION

VARIATION IN RUDDER FORCE WITH MACH NUMBER TO MAINTAIN FOUR DEGREES OF RUDDER DEFLECTION





It looks as though the plane must have been allowed to turn with wings level!

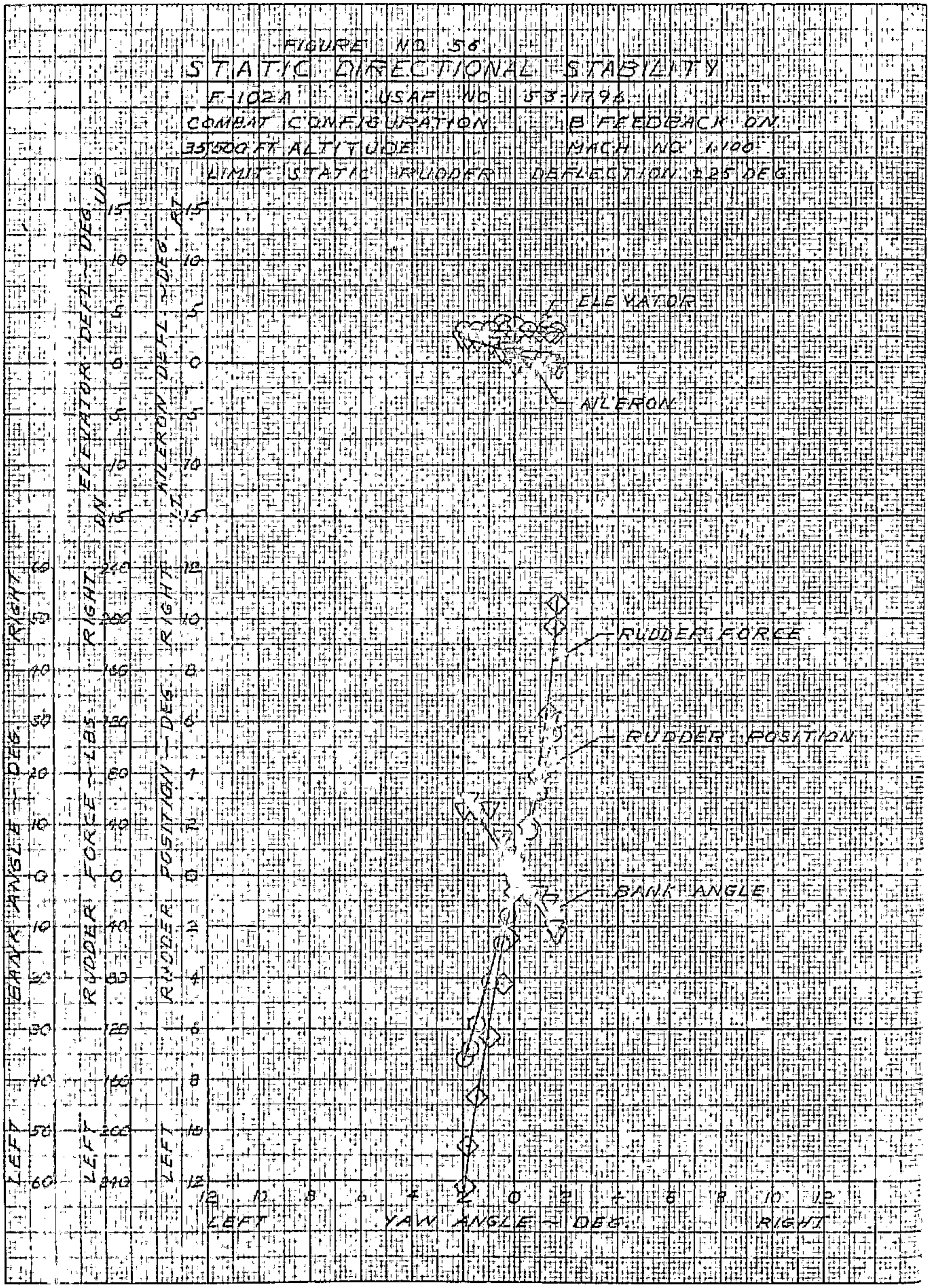


FIGURE NO. 56
STATIC DIRECTIONAL STABILITY
F-102A USAF NO. 53-1796
COMBAT CONFIGURATION B FEEDBACK ON
35,500 FT ALTITUDE MACH NO. 1.100
LIMIT STATIC RUDDER DEFLECTION ±25 DEG

FIGURE NO. 57

STATIC DIRECTIONAL STABILITY

F-102A USAF NO [illegible]-1776

CRUISE CONFIGURATION β FEEDBACK ON

35,400 ALTITUDE MACH NO. 0.965

LIMIT STATIC RUDDER DEFLECTION ± 25 DEG

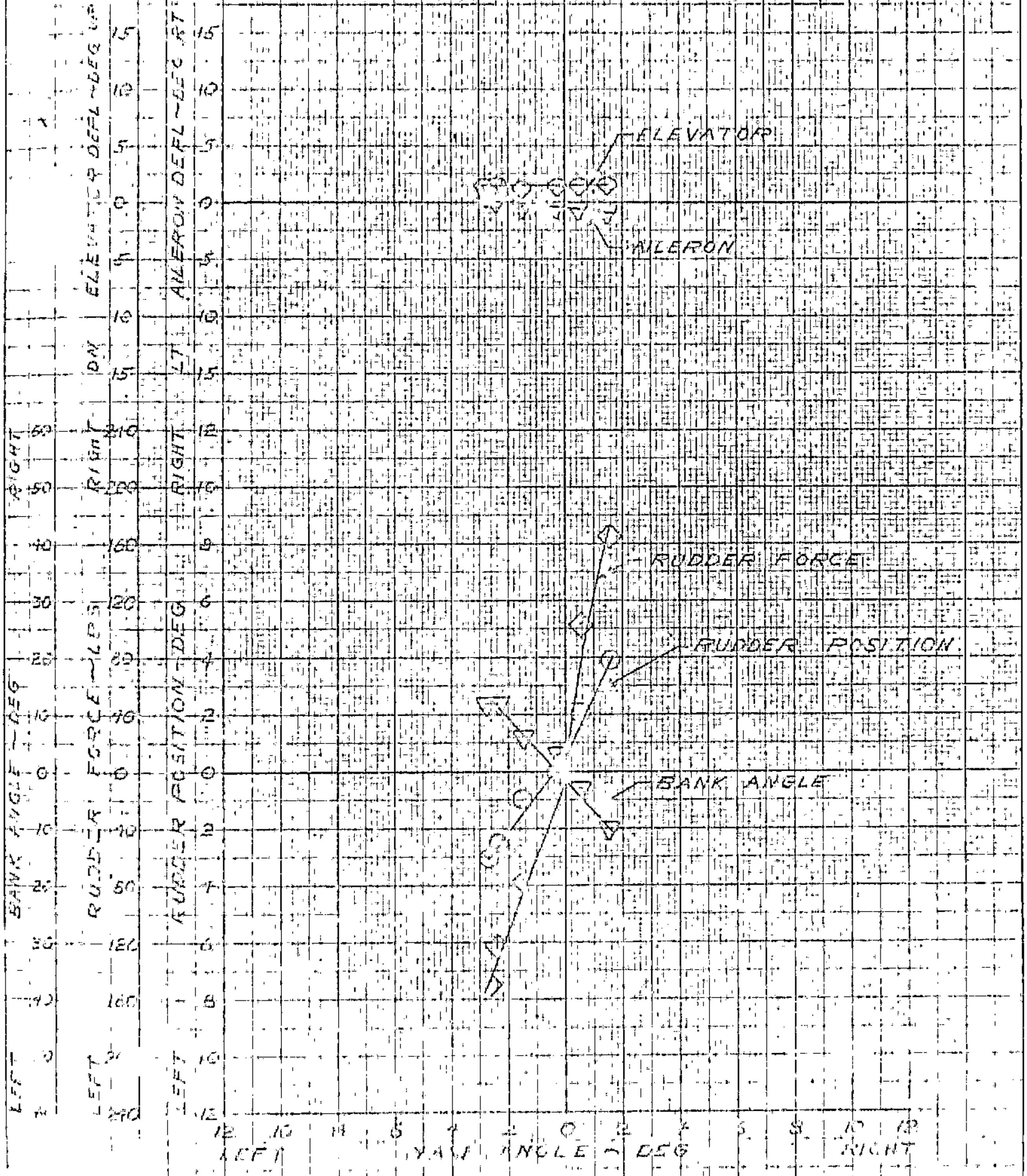


FIGURE NO. 58

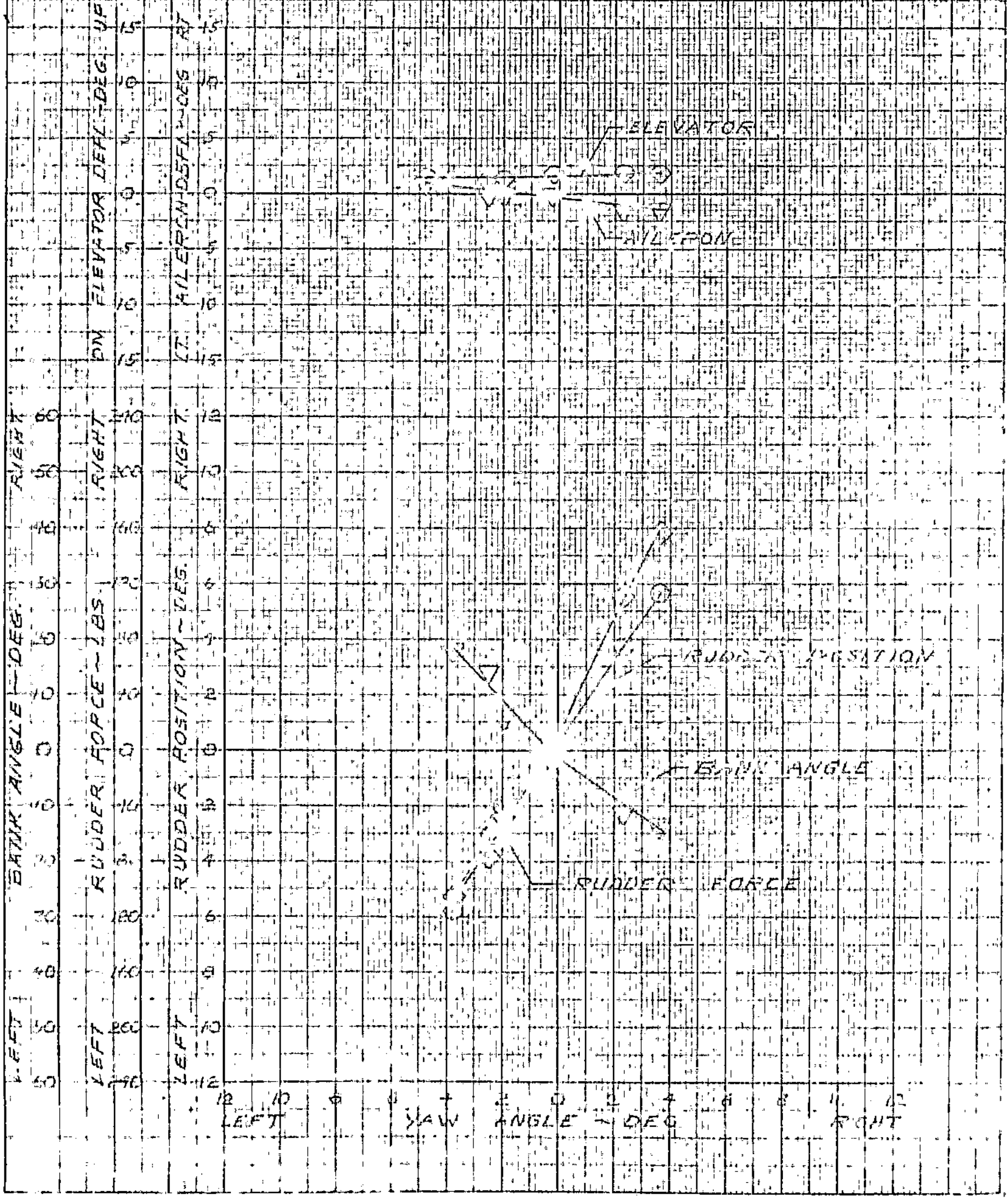
STATIC DIRECTIONAL STABILITY

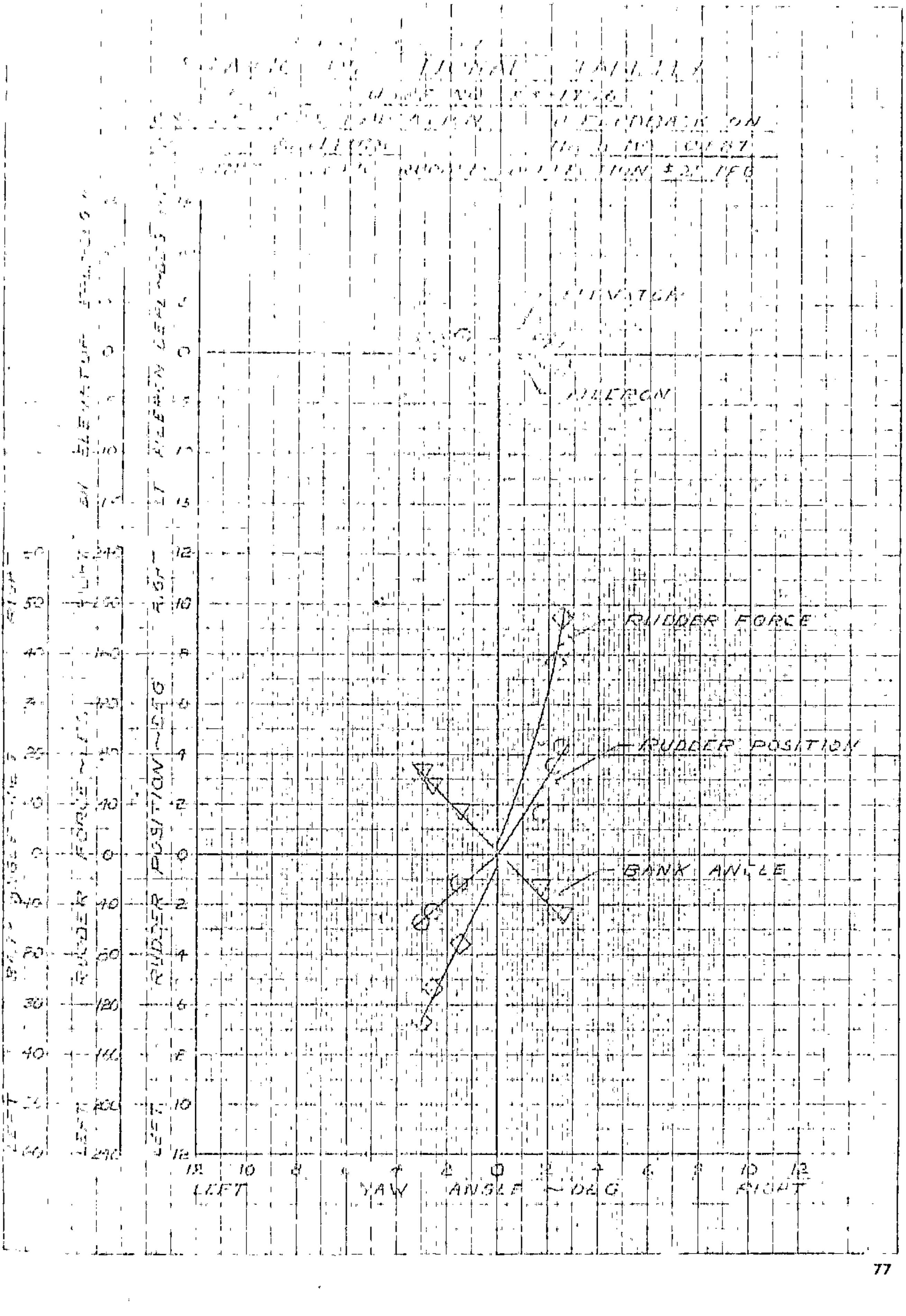
F-102A USAF NO 53-1796

CRUISE CONFIGURATION β FEEDBACK OFF

35,700 ALTITUDE MACH NO 0.965

LIMIT STATIC RUDDER DEFLECTION ±25 DEG





ELEVATOR
AILERON
RUDDER FORCE
RUDDER POSITION
BANK ANGLE
LEFT BANK ANGLE ~ DEG — RIGHT
LEFT RUDDER FORCE ~ LBS — RIGHT
LEFT RUDDER POSITION ~ DEG — RIGHT
ELEVATOR
LT AILERON DEFL
50
40
30
20
10
0
10
20
30
40
50
60
240
200
160
120
80
40
0
40
80
120
160
200
240
12
10
8
6
4
2
0
2
4
6
8
10
12
15
10
5
0
5
10
15
12
10
8
6
4
2
0
2
4
6
8
10
12
LEFT
YAW ANGLE ~ DEG
RIGHT

FIGURE NO. 60

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION β FEEDBACK OFF

35,280 ALTITUDE MACH NO. 0.887

LIMIT STATIC RUDDER DEFLECTION ± 25 DEG

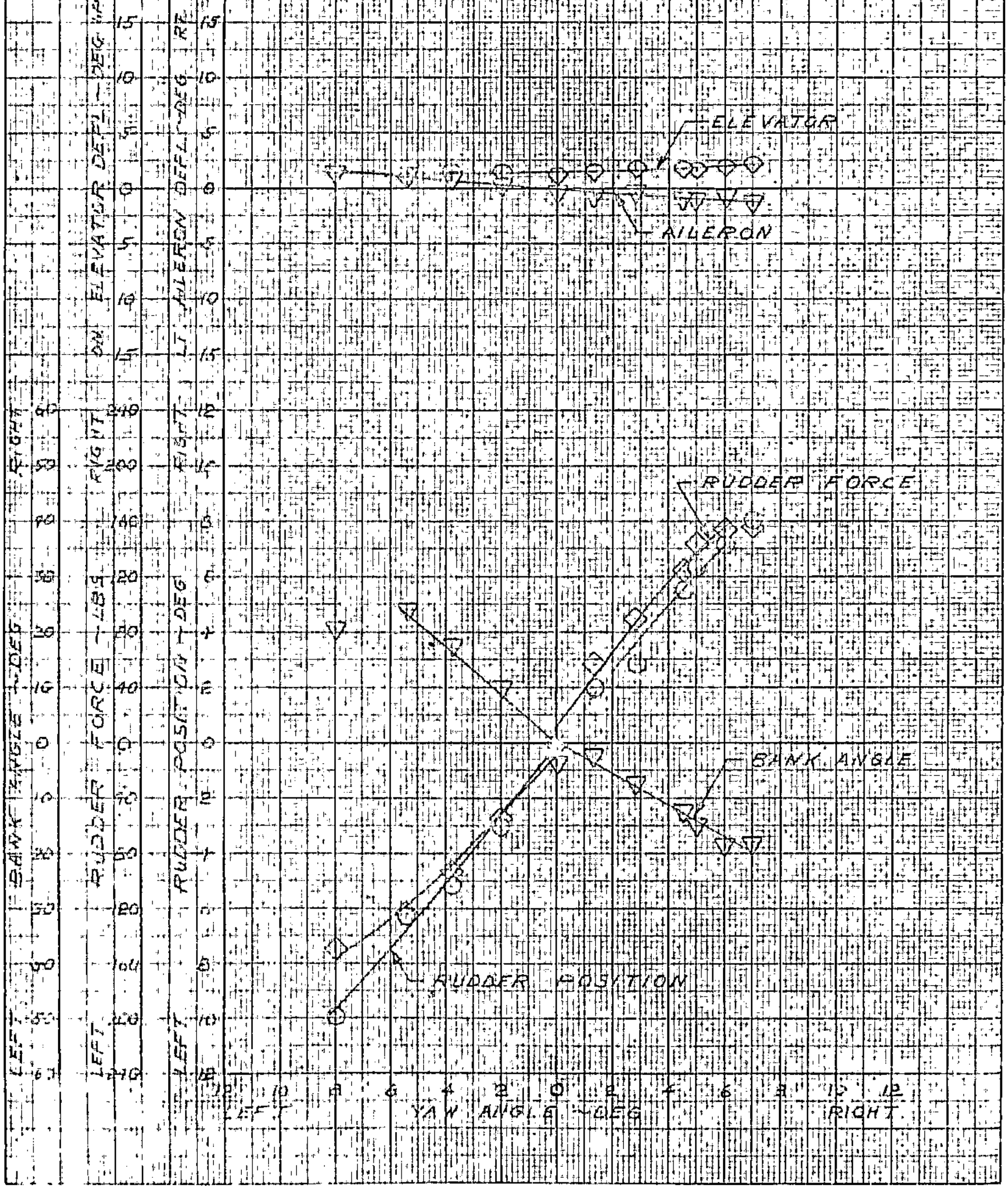


FIGURE NO. 61

STATIC DIRECTIONAL STABILITY

CRUISE CONFIGURATION

MACH NO. 0.9

LIMIT STATIC RUDDER DEFLECTION ± 2 DEG

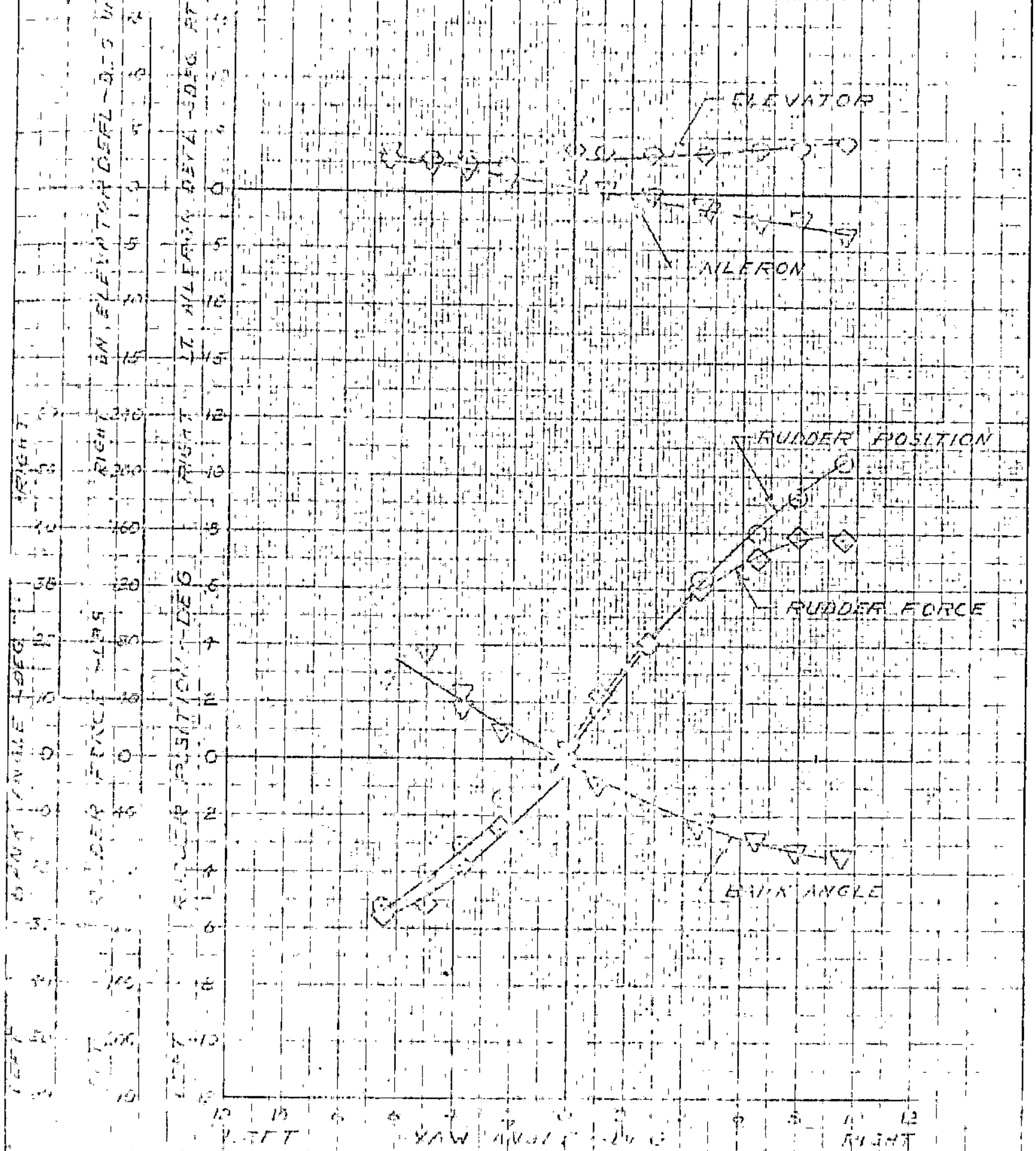


FIGURE NO. 62

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

CRUISE CONFIGURATION β FEEDBACK OFF

10,000 FT ALTITUDE MACH NO. 0.818

LIMIT STATIC RUDDER DEFLECTION ±25 DEG

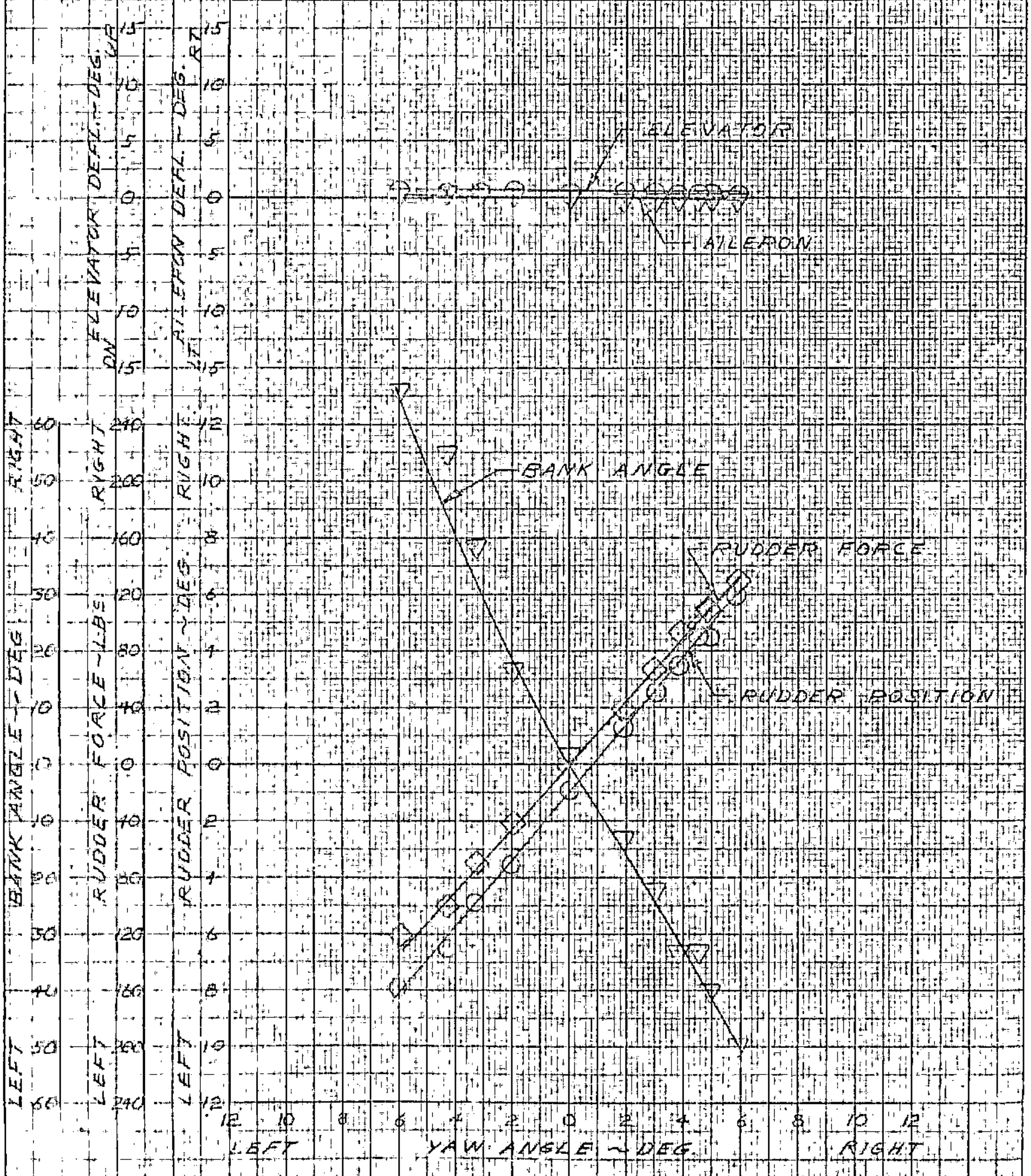


FIGURE NO. 52

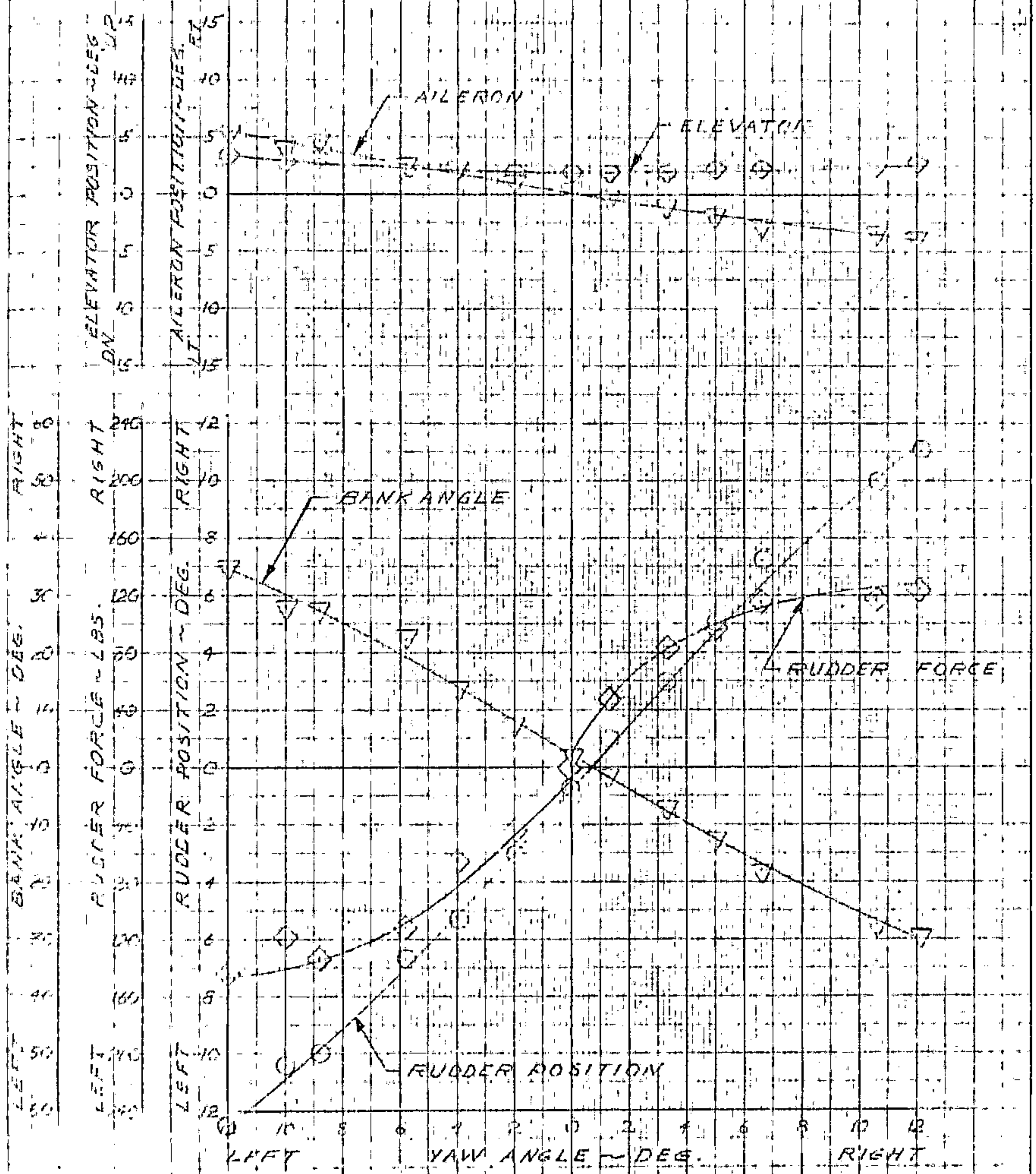
STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1776

CRUISE CONFIGURATION [illegible] ON

10,000 FT ALTITUDE MACH NO. 0.464

LIMIT STATIC RUDDER DEFLECTION ±25 DEG



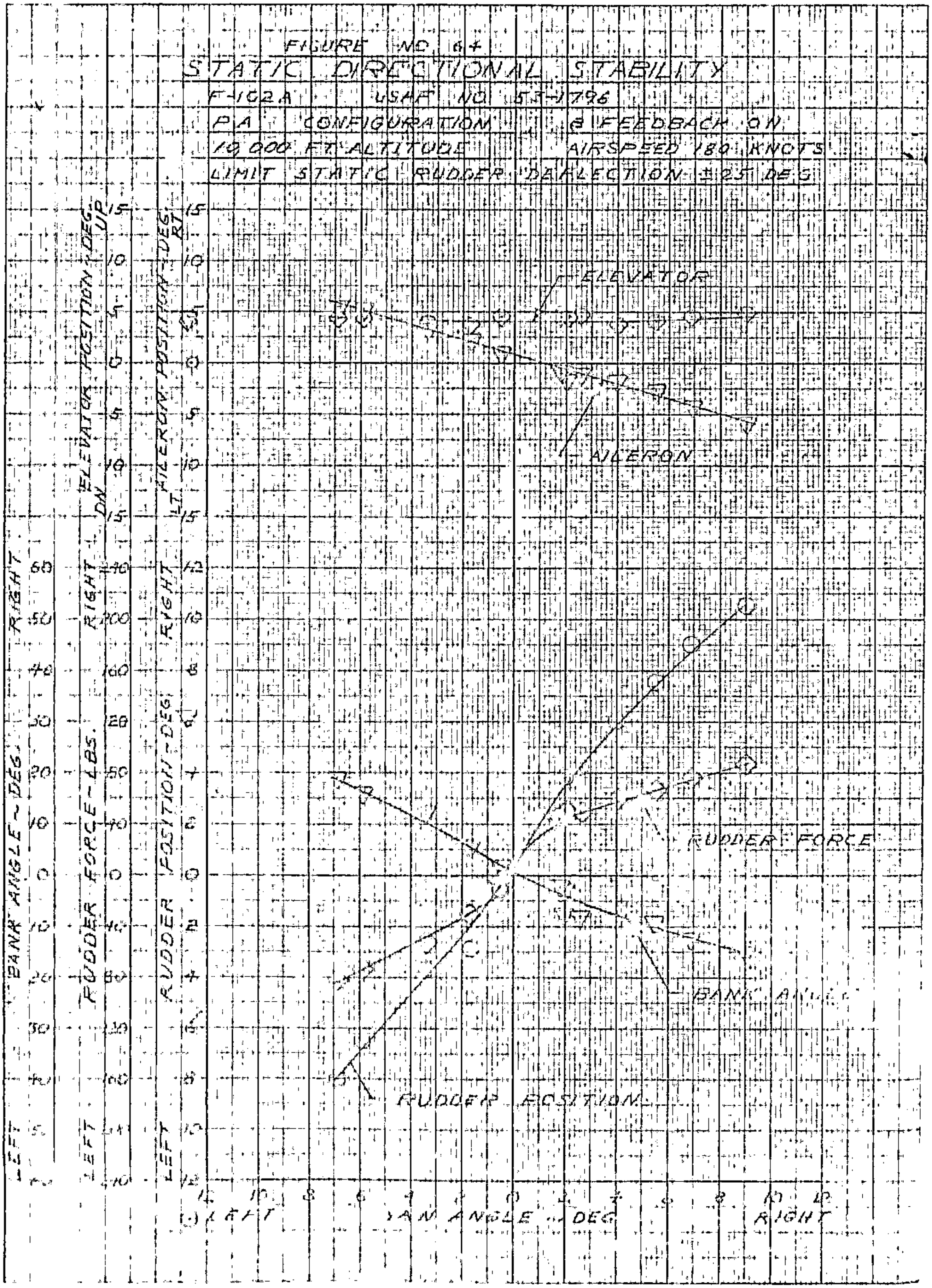


FIGURE NO. 64

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

PA CONFIGURATION G FEEDBACK ON

10,000 FT ALTITUDE AIRSPEED 180 KNOTS

LIMIT STATIC RUDDER DEFLECTION ± 25 DEG.

[illegible] 83

STATIC DIRECTIONAL STABILITY

F-102A USAF NO. 53-1796

PA CONFIGURATION 'G' FEEDBACK OFF

10,000 FT ALTITUDE INDICATED 180 KNOTS

LIMIT STATIC RUDDER DEFLECTION ±25 DEG

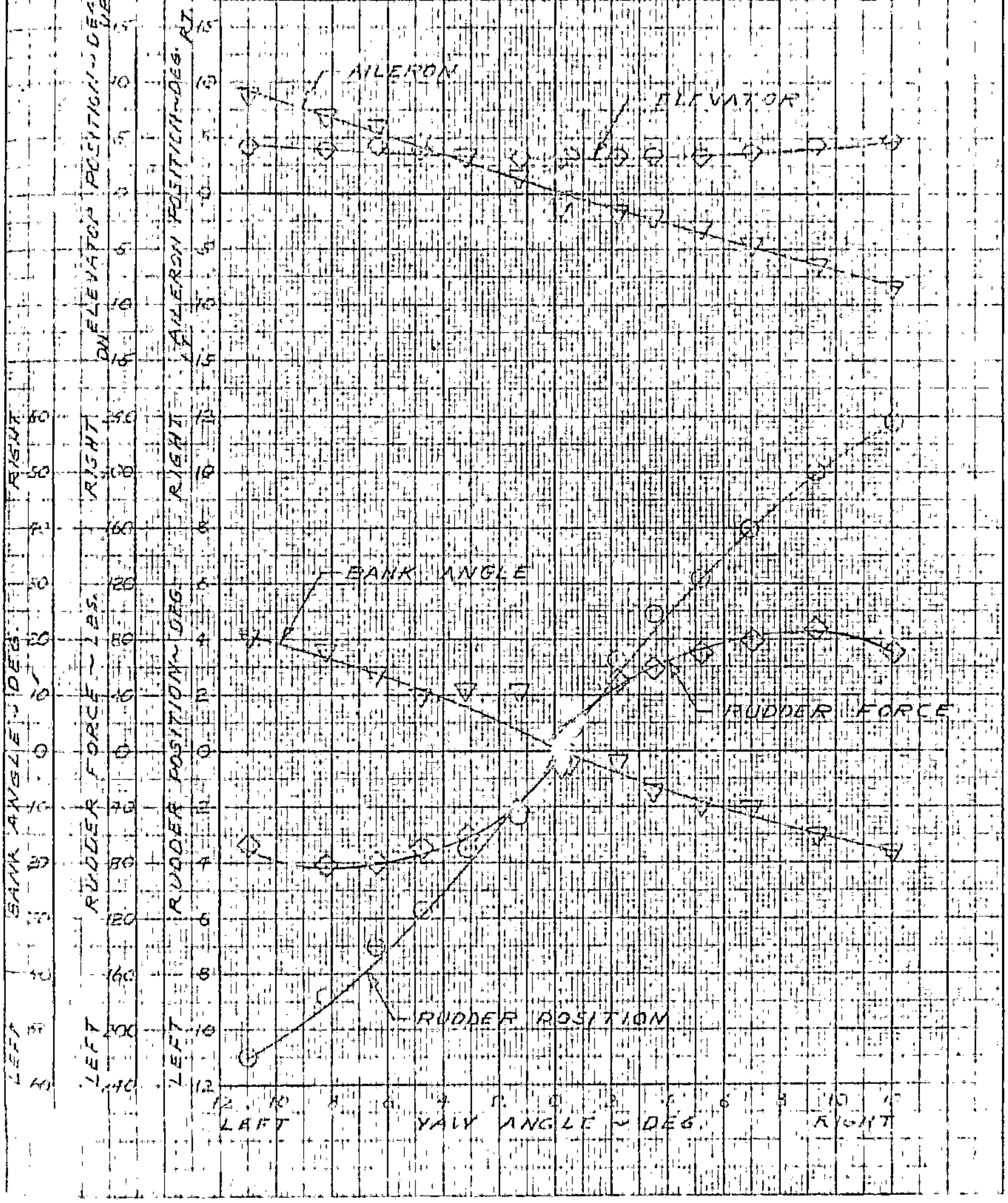


FIGURE NO 66

LATERAL CONTROL

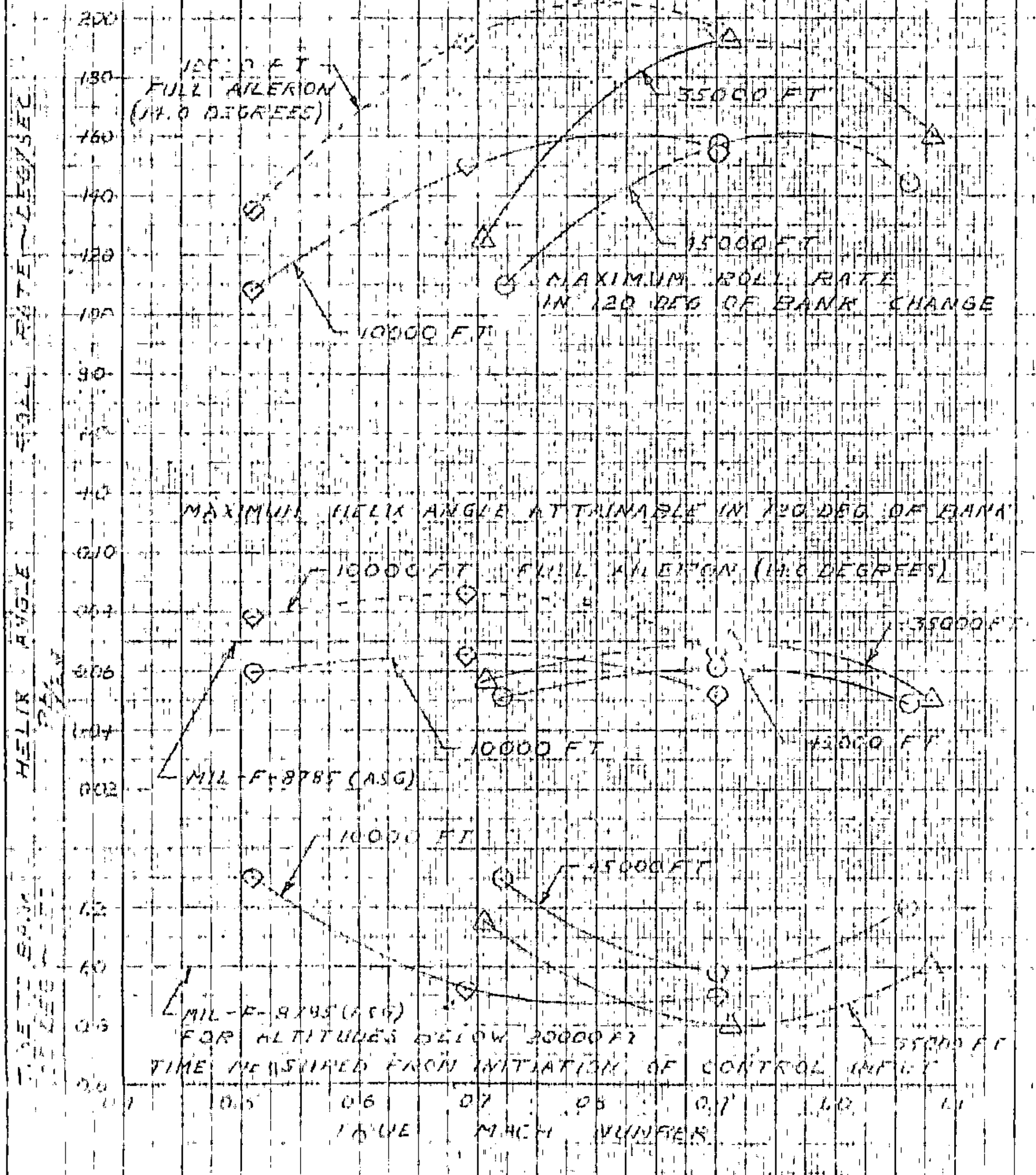
F-102A USAF NO 53-1796

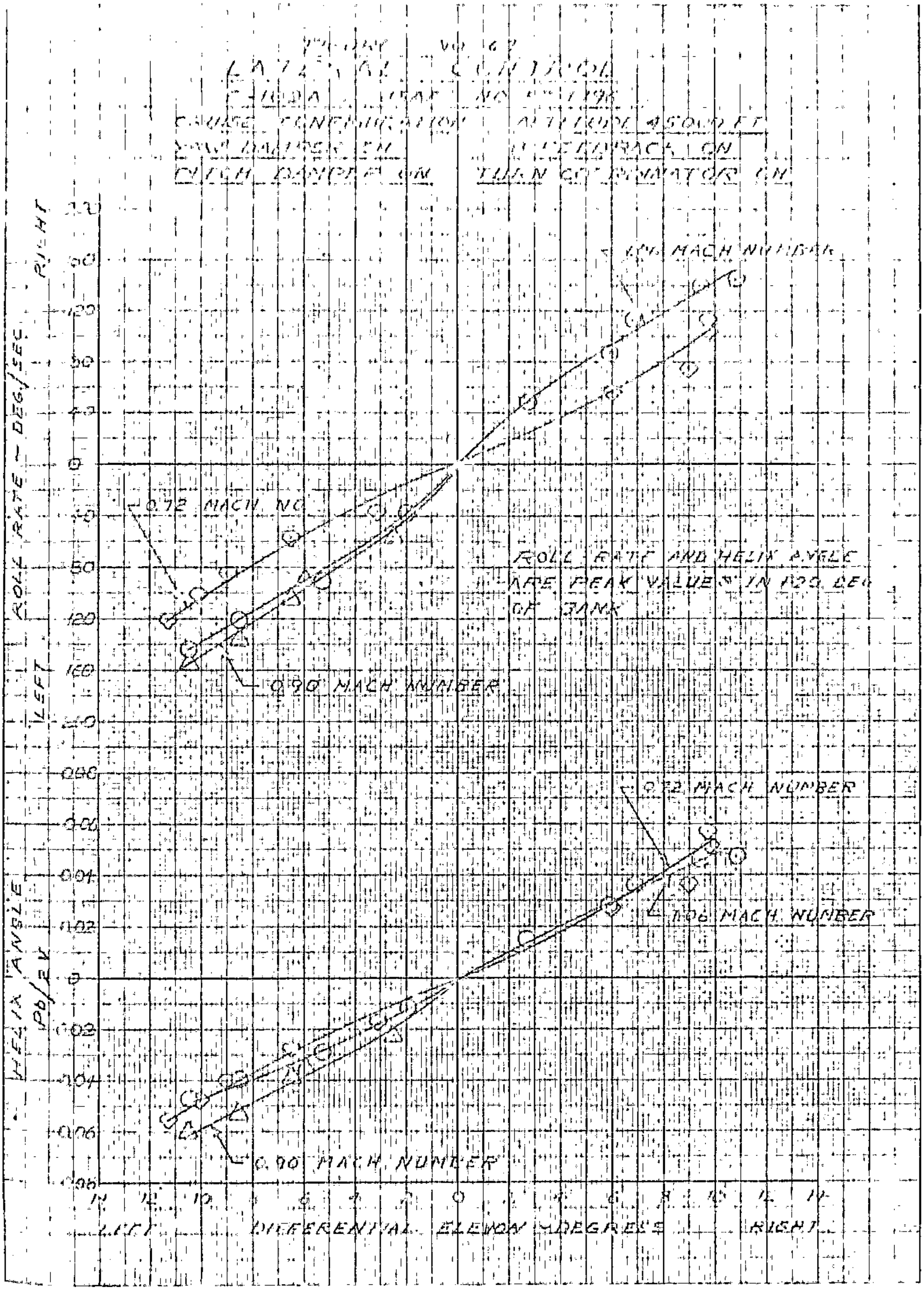
YAW DAMPER ON β FEEDBACK ON

PITCH DAMPER ON TURN COORDINATOR ON

CRUISE CONFIGURATION

10.5 DEG AILERON DEFLECTION




LATERAL CONTROL
F-102A USAF NO
CRUISE CONFIGURATION ALTITUDE 45000 FT
YAW DAMPER ON
FEEDBACK ON
PITCH DAMPER ON TURN COORDINATOR ON
RIGHT
LEFT
ROLL RATE ~ DEG/SEC
HELIX ANGLE
Pb/2V
MACH NUMBER
0.72 MACH NO
0.90 MACH NUMBER
ROLL RATE AND HELIX ANGLE ARE PEAK VALUES IN 120 DEG OF BANK
0.72 MACH NUMBER
1.06 MACH NUMBER
0.90 MACH NUMBER
200
160
120
80
40
0
40
80
120
160
200
.080
.060
.040
.020
0
.020
.040
.060
.080
14
12
10
8
6
4
2
0
2
4
6
8
10
12
14
LEFT
DIFFERENTIAL ELEVON ~ DEGREES
RIGHT

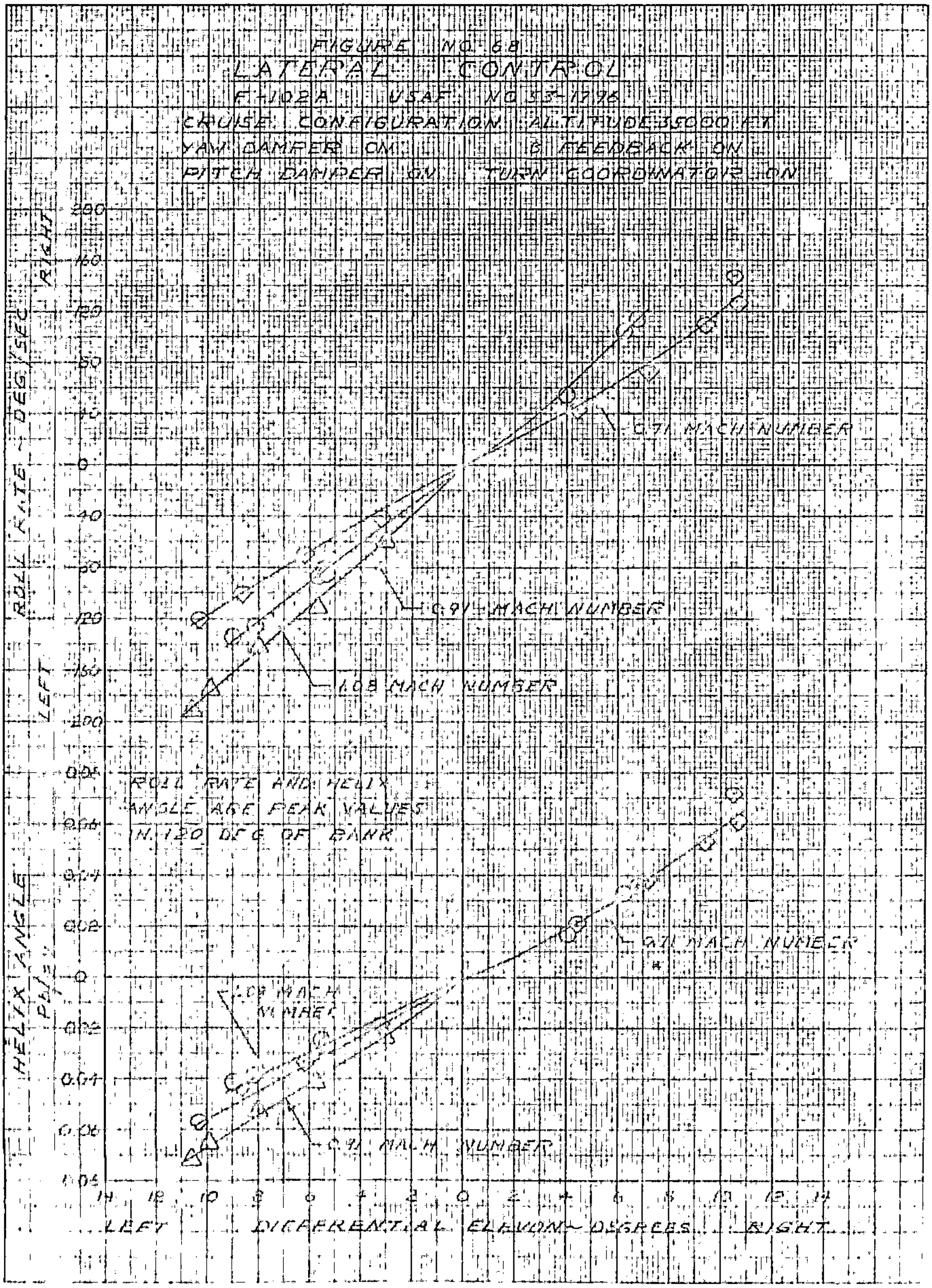


FIGURE NO 68

LATERAL CONTROL

F-102A USAF NO 53-1796

CRUISE CONFIGURATION ALTITUDE 35000 FT

YAW DAMPER ON β FEEDBACK ON

PITCH DAMPER ON TURN COORDINATOR ON

ROLL RATE AND HELIX ANGLE ARE PEAK VALUES IN 120 DEG OF BANK

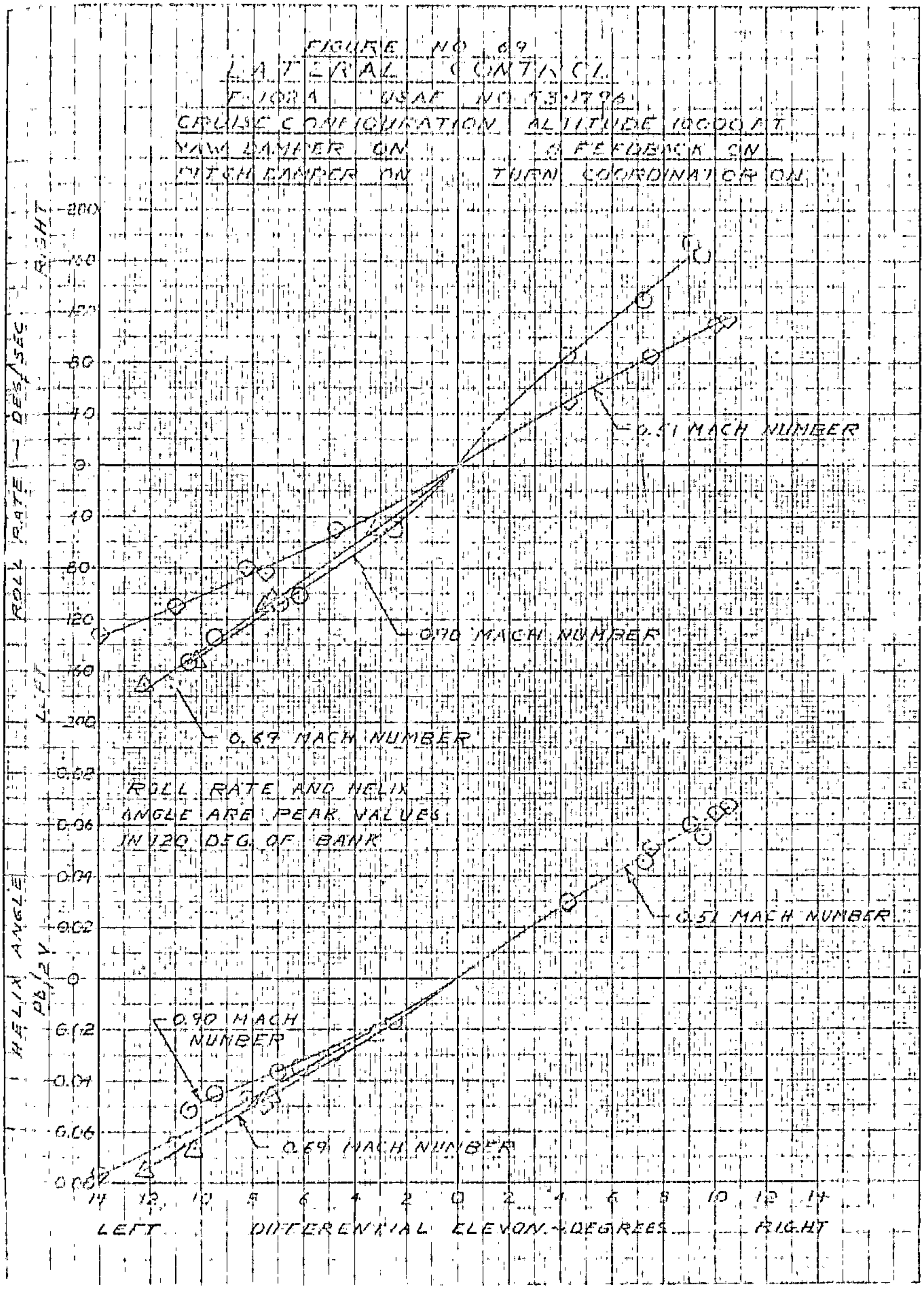


FIGURE NO 69
LATERAL CONTROL
F-102A USAF NO 53-1796
CRUISE CONFIGURATION ALTITUDE 10000 FT
YAW DAMPER ON G FEEDBACK ON
PITCH DAMPER ON TURN COORDINATOR ON

FIGURE NO 70

LATERAL CONTROL

F-102A USAF NO 53-1795

POWER APPROACH CONFIGURATION

10000 FT ALTITUDE AIRSPEED 188 KNOTS

YAW DAMPER ON β FEEDBACK ON

PITCH DAMPER ON TURN COORDINATOR ON

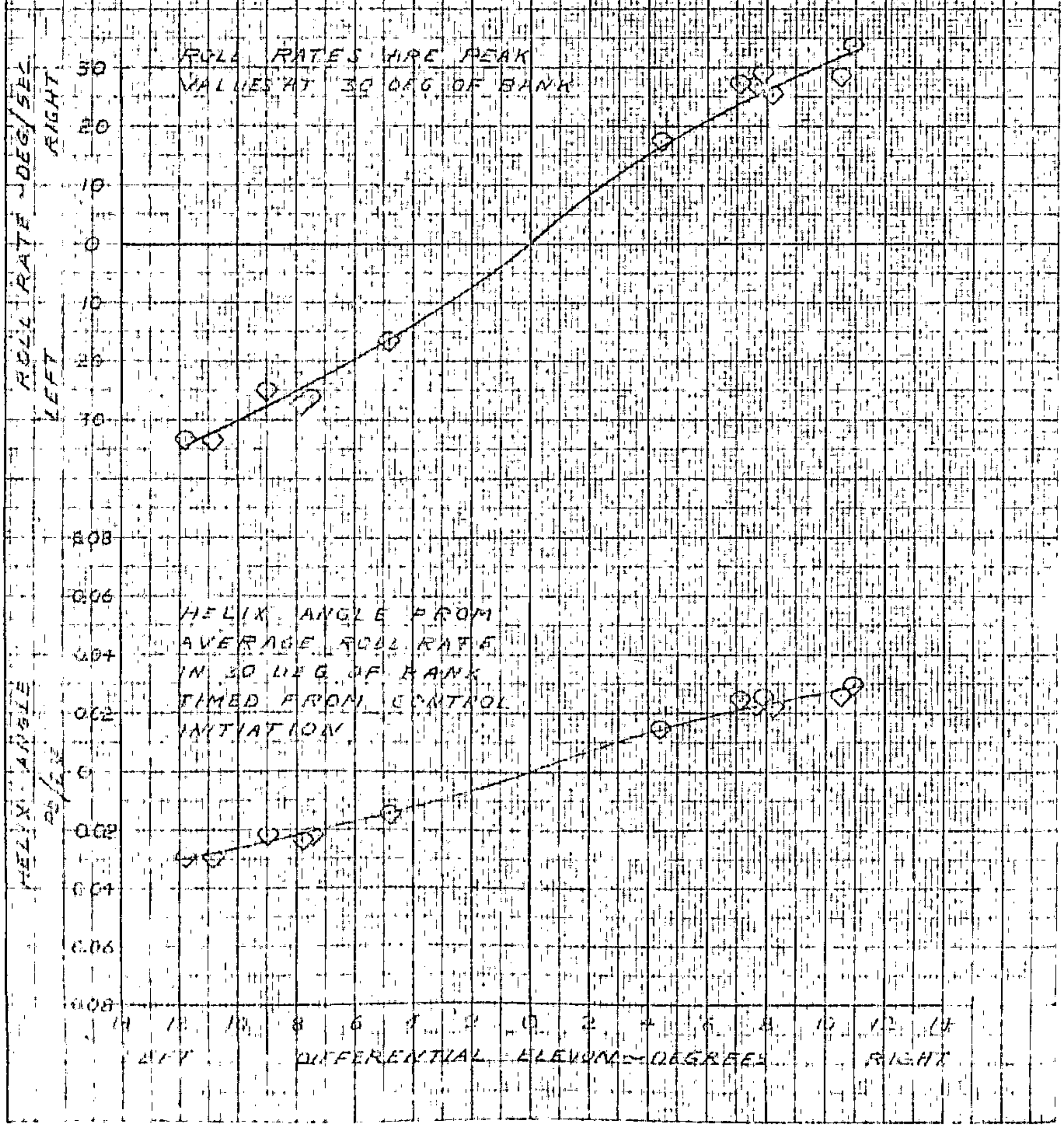


FIGURE NO. 71
INERTIAL COUPLING
F-102A USAF NO. 53-1796
SIDESLIP FEEDBACK ON
ROLL RATE 120 DEG/SEC

stability augmentation?

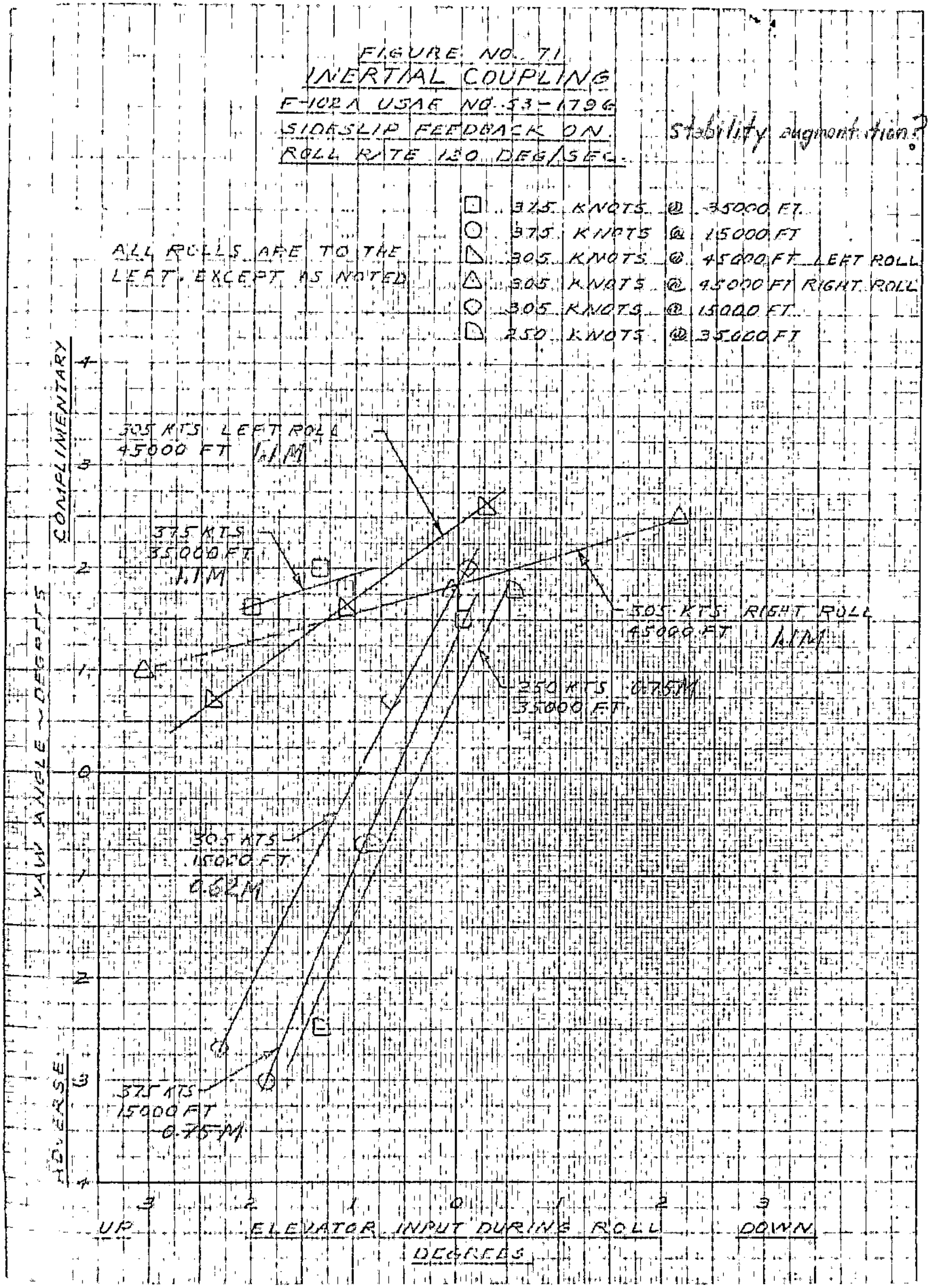


FIGURE NO. 72
INERTIAL COUPLING
F-102A USAF NO. 53-1796
SIDESLIP FEEDBACK OFF
ROLL RATE 80 DEG/SEC

Stability augmentation?

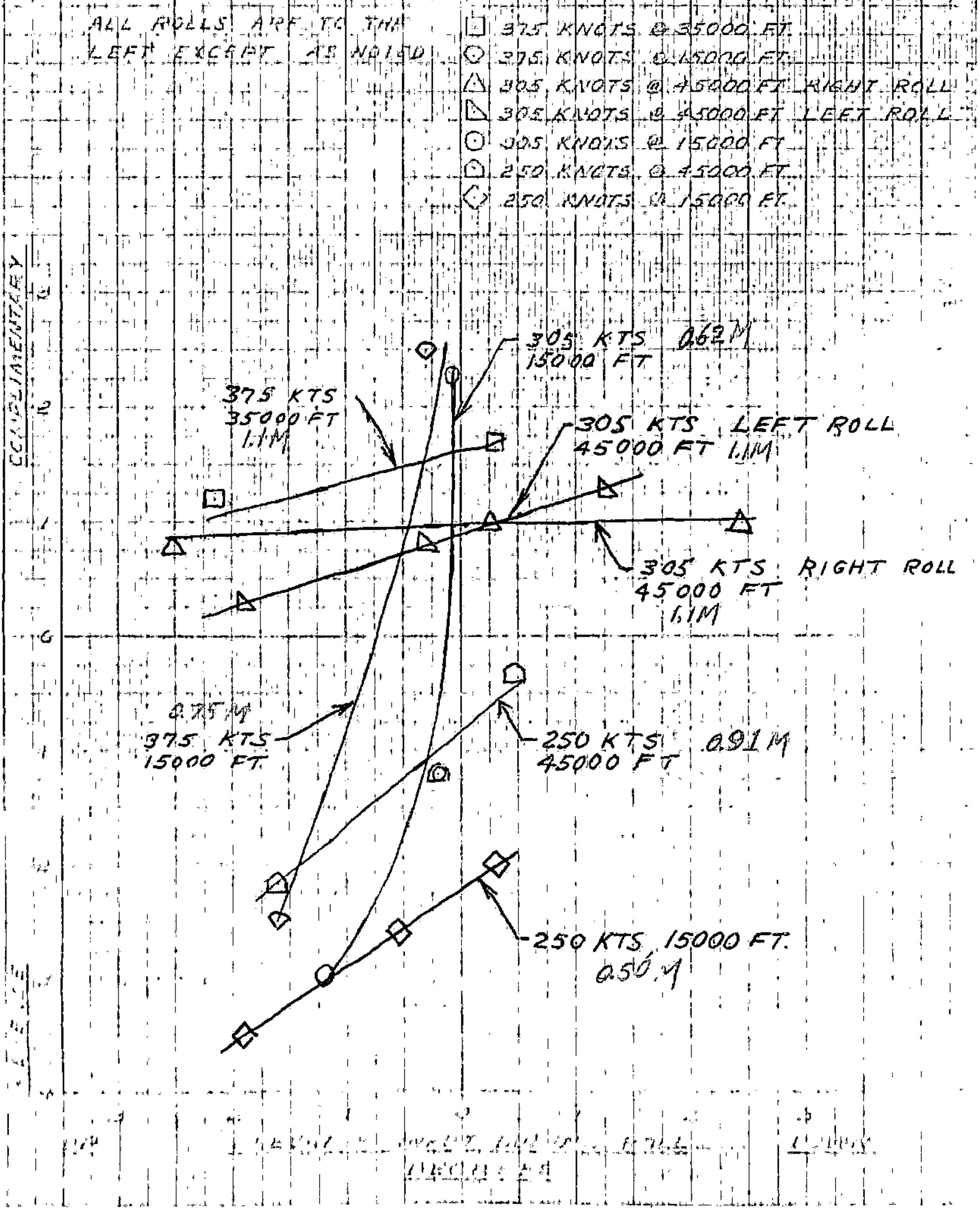


FIGURE NO. 73

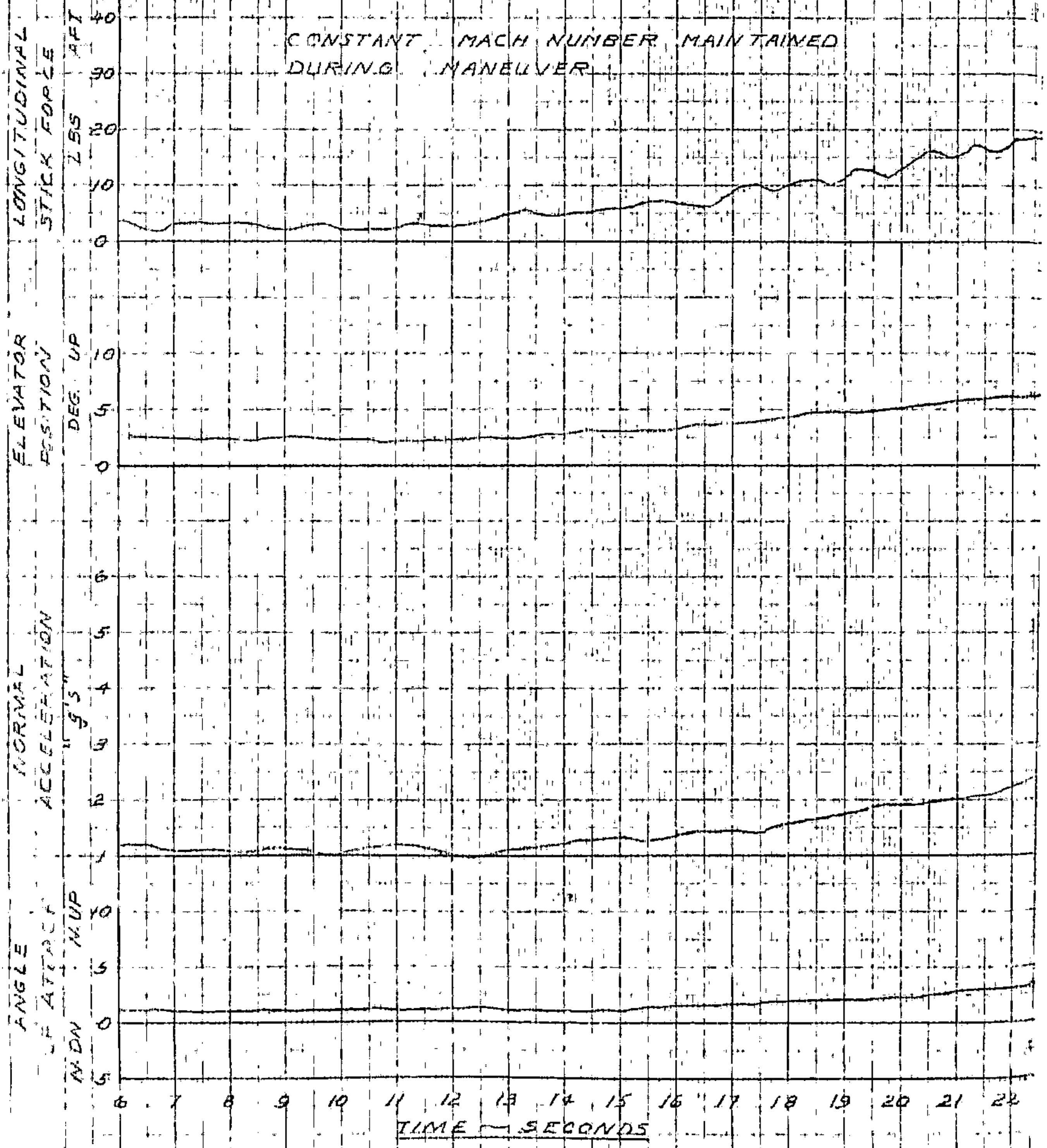
STALL TIME HISTORY (ACCELERATED STALL)

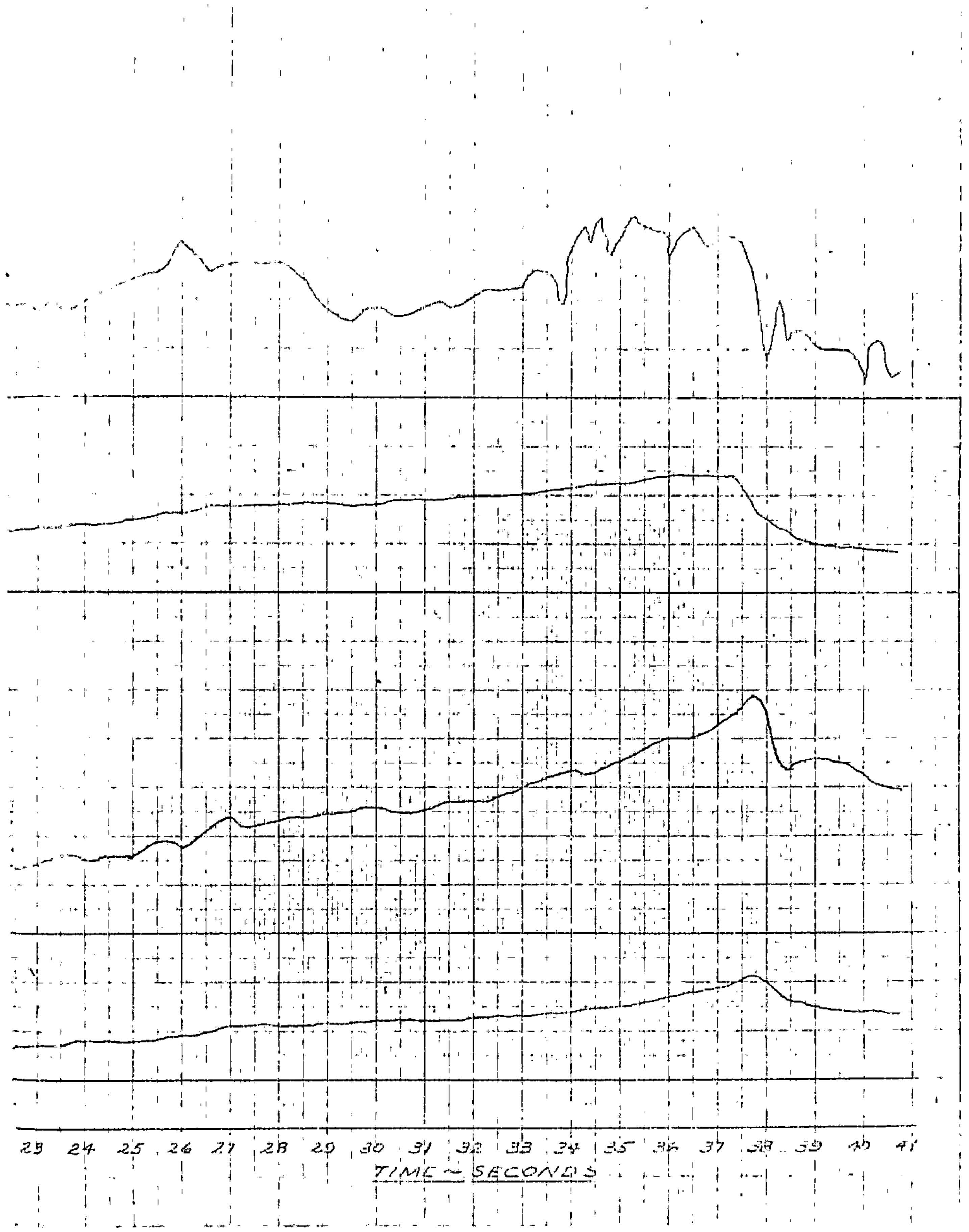
F-102A USAF NO. 53-1796

CRUISE CONFIGURATION ALTITUDE 35000 FT.

WEIGHT 23,800 LBS. C.G. 27.0 PERCENT MAC

TRIM MACH NUMBER 0.975




23
24
25
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31
32
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34
35
36
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39
40
41
TIME ~ SECONDS

FIGURE No. 74
AIRSPEED CALIBRATION
F-102A USAF NO. 53-1746
CRUISE CONFIGURATION

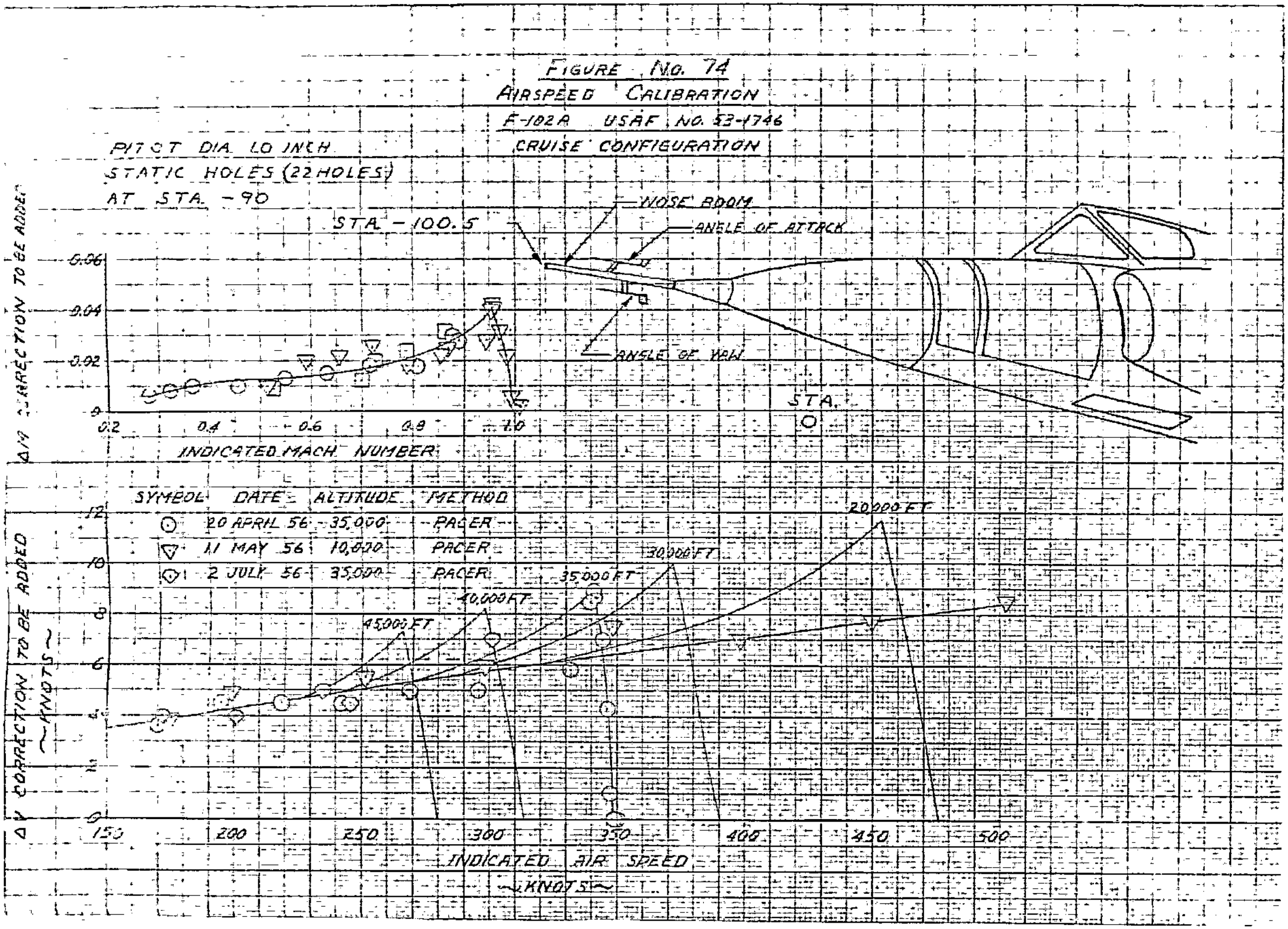


FIGURE NO. 75

LONGITUDINAL BREAKOUT FRICTION

F-106A U.S.AF. No 58-776

MEASURED IN A CLOSED HANGAR

TEMPERATURE 26° C

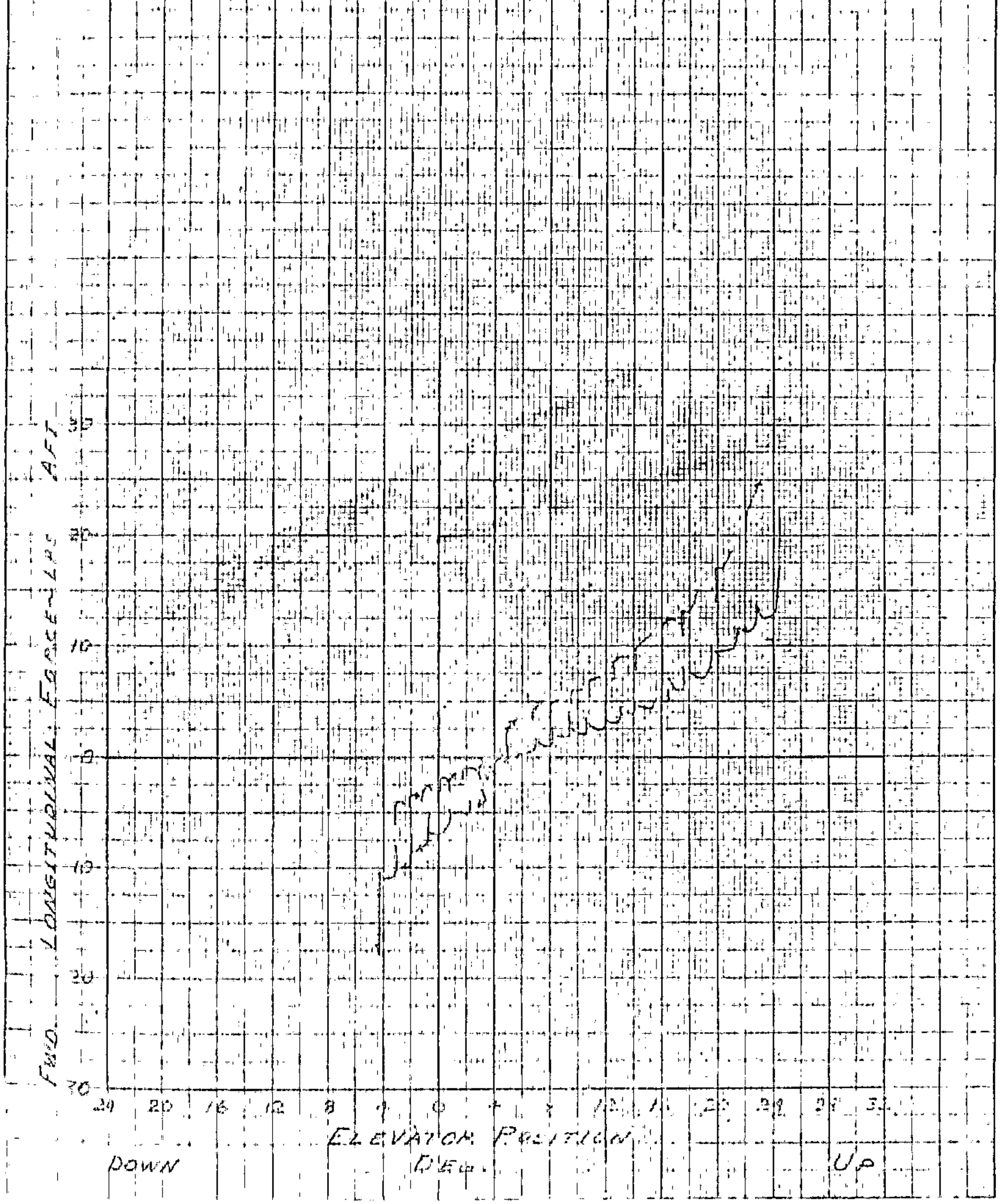


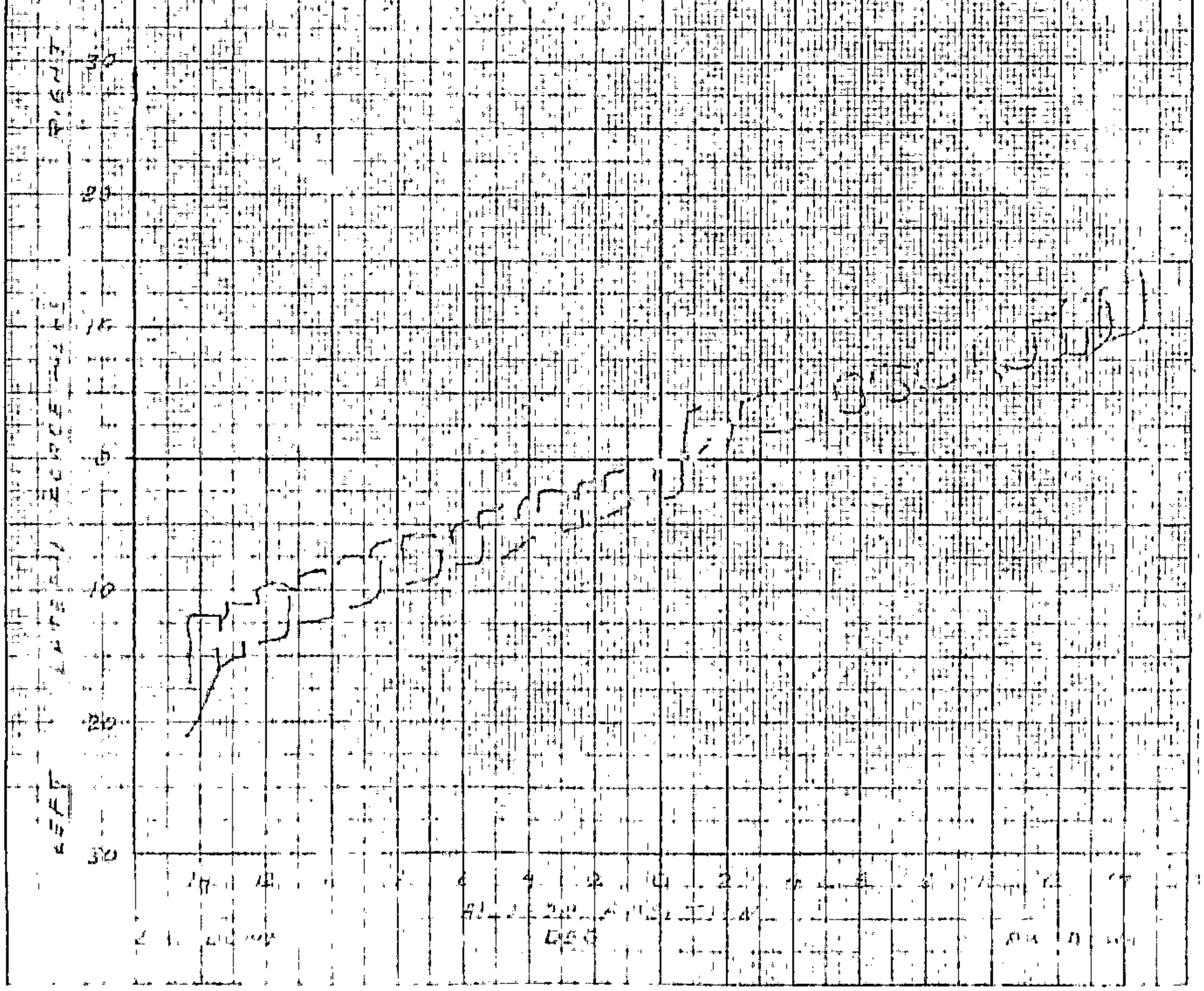
FIGURE NO 76

LATERAL BREAKOUT FRICTION

E-102A USED AS E7-226

MEASURED IN A CLOSED HANGAR

TEMPERATURE 20°C



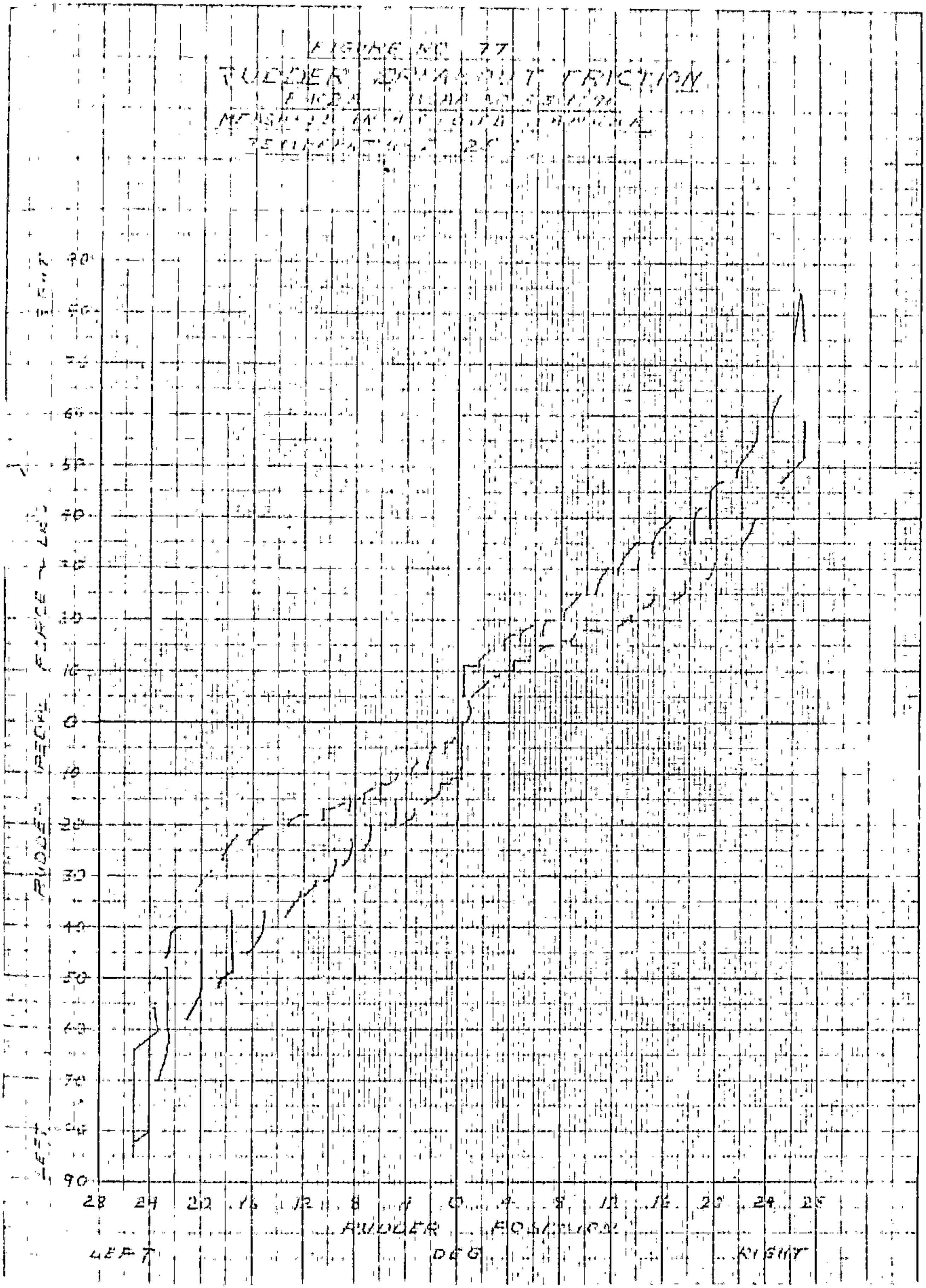


FIGURE NO. 77
RUDDER BREAKOUT FRICTION

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TEST 56-226

ENGINEERING TEST LABORATORIES

CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION

SAN DIEGO

REPORT 56-226

DATE 21 January 1957

MODEL F-102A

TITLE

REPORT NO. 56-226
AILERON SYSTEM AND STICK INSTALLATION (SIDE STICK) PROOF TEST
MODEL F-102A

CONTRACT NO. AF 33(600)-5942

PREPARED BY J. P. McNelly

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FORM 1812 A-4

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Load Deflection Curves

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REPORT NO. 56-226
AILERON SYSTEM AND STICK INSTALLATION (SIDE STICK)
PROOF TEST
MODEL F-102A

REFERENCE:

(a) Convair Structures Memo S-8-37, dated 23 March 1956.

INTRODUCTION:

A production F-102 airplane (Serial Number AF-53-1813) had a side stick installation for evaluation purposes. The installation was made per Convair ACA 8-1111-01. This included a new control stick assembly and the modifications necessary to move the control stick from centerline to the right hand side position. The control stick installation details are shown in Figure 1.

OBJECT:

To prove the structural integrity of the aileron system and stick installation of the Model F-102A airplane with the side stick installation.

CONCLUSION:

Observation of the control system under load and the test deflection data indicated the aileron system and the stick installation were structurally satisfactory for the conditions tested.

GENERAL PROCEDURE:

The testing was accomplished in two parts. Prior to the test of the controls, the castings which made up the new grip assembly were static tested to destruction to confirm the casting design. In the second part of the test, the proof loads were applied to the airplane's control system omitting the castings which made up the pilot's hand grip. The two parts of the test are detailed below.

PART I - GRIP ASSEMBLY CASTINGS

Test Specimen and Procedure:

Three aluminum castings detailed in TVA 47623 (on 8-49520) were used in the grip installation. These were the -7 grip, the -9 grip cover, and the -11 head. The rough grip castings -7 and -9 are shown in Figure 2 and the machining requirements are shown in Figure 3. The three castings were assembled with a -15 bellcrank as shown in Figure 4.

The design ultimate loads and the four Conditions of loading are shown in Figure 5. Condition I through Condition III were proof loads. The castings were to be failed in Condition IV with an ultimate load of 300 pounds applied in the aft direction.

The castings were mounted for test as shown in Figure 6 for Condition I. A TVA 45503-25 sleeve was riveted to a simulated tube and the grip assembly was secured to the sleeve by a TVA 45503-27 nut. The grip assembly was loaded through a loading unit attached to the grip castings by a low melting point alloy metal. A width of three inches was used to simulate the area covered by the pilot's hand. The test load was measured by a dynamometer for the forward, aft, and side loads. The torsion load for Condition II was applied as shown in Figure 7 by a torque wrench. The test loads for Conditions III and IV were applied as shown in Figure 8.

Dial gages were mounted as shown in Figures 5 through 8. For Conditions III and IV, an additional dial gage was placed to measure the vertical movement. Deflections were read in twenty percent increments of the design ultimate load up to 200 percent and in 50 percent increments thereafter.

Discussion of Results:

The castings were proof loaded to 120 percent of the design ultimate load for Conditions I, II, and III without yielding as shown by the deflection data.

The -7 grip casting failed at a test load of 1,200 pounds, 400 percent of the design ultimate load, in tension due to bending.

Load-deflection curves for the proof Conditions are shown in Figure 10 and the failure Condition in Figure 11.

The grip assembly had a test margin of safety of 3.0 and was structurally satisfactory for the static design ultimate load.

PART II - AIRPLANE PROOF LOADS

Test Specimen and Procedure:

The Model 8-10 No. 23 (AF 53-1813) was used as the test specimen. The side stick installation was checked by Convair Inspection prior to the proof load. This included the positioning of the stops and the setting of the centering devices in the system.

The test loads were applied to the airplane by the Service Engineering rigging gage as shown in Figure 12. To use this tool, the electrical wires to the grip assembly were broken and the -7 grip casting was removed.

The aft loads on the control stick were applied in tension by a cable winch as shown in Figure 13. The forward stick load was applied by a hydraulic ram as shown in Figure 14. The test loads were measured by an aluminum strain gage dynamometer which attached to the rigging tool. The dynamometer was designed to apply the load one-half inch above the centerline of the rigging tool and the proper distance aft. The torsion load for the aileron system was applied by a torque wrench as shown in Figure 15.

The deflection of the stick for the fore and aft loads was measured by means of a protractor and a plumb bob as shown in Figure 14. The stick deflections were read to a tenth of a degree. The deflection of the grip for the aileron loads was measured by the protractor on the rigging gage as shown in Figure 15 and was read to the nearest quarter of a degree. The elevon surface movement was measured by a protractor attached to the fuselage. Deflections were recorded in ten percent increments of the proof load starting with a zero load. Set readings were taken after each load increment returning to a tare load of ten percent.

The detail requirements and loads for the test conditions are as follows:

A. Aileron System

(1) With the hydraulic power system inoperative and the aileron system in the neutral position, apply 300 inch-pounds torque to the control stick grip. The load will be necessary to hold the surface in the neutral position. Measure the rotation (in degrees) of the grip.

(2) Same as (A-1), except apply torque in the opposite direction.

(3) With the hydraulic power system inoperative and the aileron system in an extreme travel position, (one elevon 7° down), apply 300 inch-pounds torque to the grip in the direction that tends to return the surfaces to neutral. The load will be reacted at the control valve stops. It will be necessary to hold the surfaces in their 7° up and 7° down position. Measure the rotation of the grip.

(3A) Same as (A-3), except replace the 40 inch-pound elevon centering cam with the 50 inch-pound cam.

(4) Same as (A-3), except the aileron system is in the opposite extreme travel position.

(5) With the hydraulic power system in operation, apply 300 inch-pounds torque to the grip. The load will be reacted at the forward stop. Measure the elevon surface travel, the grip rotation, and the load required to contact the stop.

(5A) Same as (A-5), except replace the 40 inch-pound elevon centering cam with the 50 inch-pound cam.

(6) Same as (A-5), except apply torque in the opposite direction.

B. Elevator System (Control Stick Installation)

(1) With the hydraulic power system inoperative, zero pressure in the feel cylinder, and the elevator system in the elevator up position, apply a 200 pound load to the grip in the elevator down direction. The load will be reacted at the hydraulic control valve stops. It will be necessary to hold the surfaces in the elevator up position. Measure the control stick travel.

(2) Same as (B-1), except the elevator system is in the elevator down position and the load is applied to the grip in the elevator up direction.

(3) With the hydraulic power system in operation and zero pressure in the feel cylinder, apply a 200 pound load to the control stick in the elevator up direction. The load will be reacted at the forward elevator stops. Measure the control stick travel, the elevon surface travel, and the load required to contact the stop.

(4) Same as (B-3), except apply the load in the elevator down direction.

RESULTS:

No evidence of any structural damage resulted from the proof loading of the control system. Load-deflection data are given in Table I. Set versus deflection data are given in Table II. Load-deflection curves are shown in Figures 16 through 24.

DISCUSSION:

The set values at proof load exceeded the 5 percent (stated as maximum in reference (a)) of the total deflection in three of the loading conditions. These values, 6.5 percent for Condition A-3A, 8.6 percent for Condition B-1, and 10.0 percent for Condition B-3 were not considered excessive by the Structures Group.

NOTE:

The test data from which this report was prepared are recorded in Engineering Test Laboratories Data Book No. 376, page 90 for Part I and in Data Book No. 379, pages 90 through 95 for Part II.

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FORM 73

TABLE I
LOAD - DEFLECTION DATA
(DATA PRESENTED IN DEGREES)

SYSTEM	CONDITION NUMBER	LOCATION OF READING	PERCE							
			10	20	10	30	10	40	10	5
AILERON	A-1	GRIP	24	39	0	45	0	49.5	0	53
	A-2	GRIP	23.5	38	2.3	43	2.3	46.5	2	5
	A-3	GRIP	10.8	17.8	-.5	30.3	-.5	41.5	-.3	53
	A-3A	GRIP	6	15	1.5	27	2.5	41.5	4	5
	A-4	GRIP	9.5	18	-.5	30	.5	40	.5	4
	A-5	GRIP	24.5	58.5	2.5	63	2.5	63.5	1.5	6
		R.H. ELEVON	2.2	6.3	.2	6.8	.1	6.8	0	7
	A-5	L.H ELEVON	2.3	6.5	.1	6.9	.1	7	0	6
	A-5A	GRIP	23	46.5	0	51	0	58	1	6
		R.H. ELEVON	2	4.5	0	4.9	-.1	5.8	0	6.
	A-5A	L.H ELEVON	2.1	4.5	0	4.9	0	6.3	.1	6.
	A-6	GRIP	23	55	0	58.5	.1	60	2	6
		R.H ELEVON	2.1	6.3	0	6.6	.1	6.8	.2	6.
AILERON	A-6	L.H ELEVON	2.1	6.3	0	6.6	.1	6.8	.4	6
ELEVATOR	B-1	STICK	.5	1.2	.2	2.4	.2	4.5	.4	6
	B-2	STICK	.8	1.8	.1	2.9	.2	4.3	.1	5
	B-3	STICK	15.7	21.2	2.6	22.7	2.4	23.4	2.6	2
		R.H. ELEVON	14.5	22.5	3.3	24.5	3	24.5	2.6	2
	B-3	L.H. ELEVON	14.5	22.7	3.3	25	3	25	2.6	2
	B-4	STICK	9.3	10.2	0	10.5	.1	10.6	.1	10
ELEVATOR	B-4	R.H. L.H. ELEVON	AGAINST STOP AT 5%							

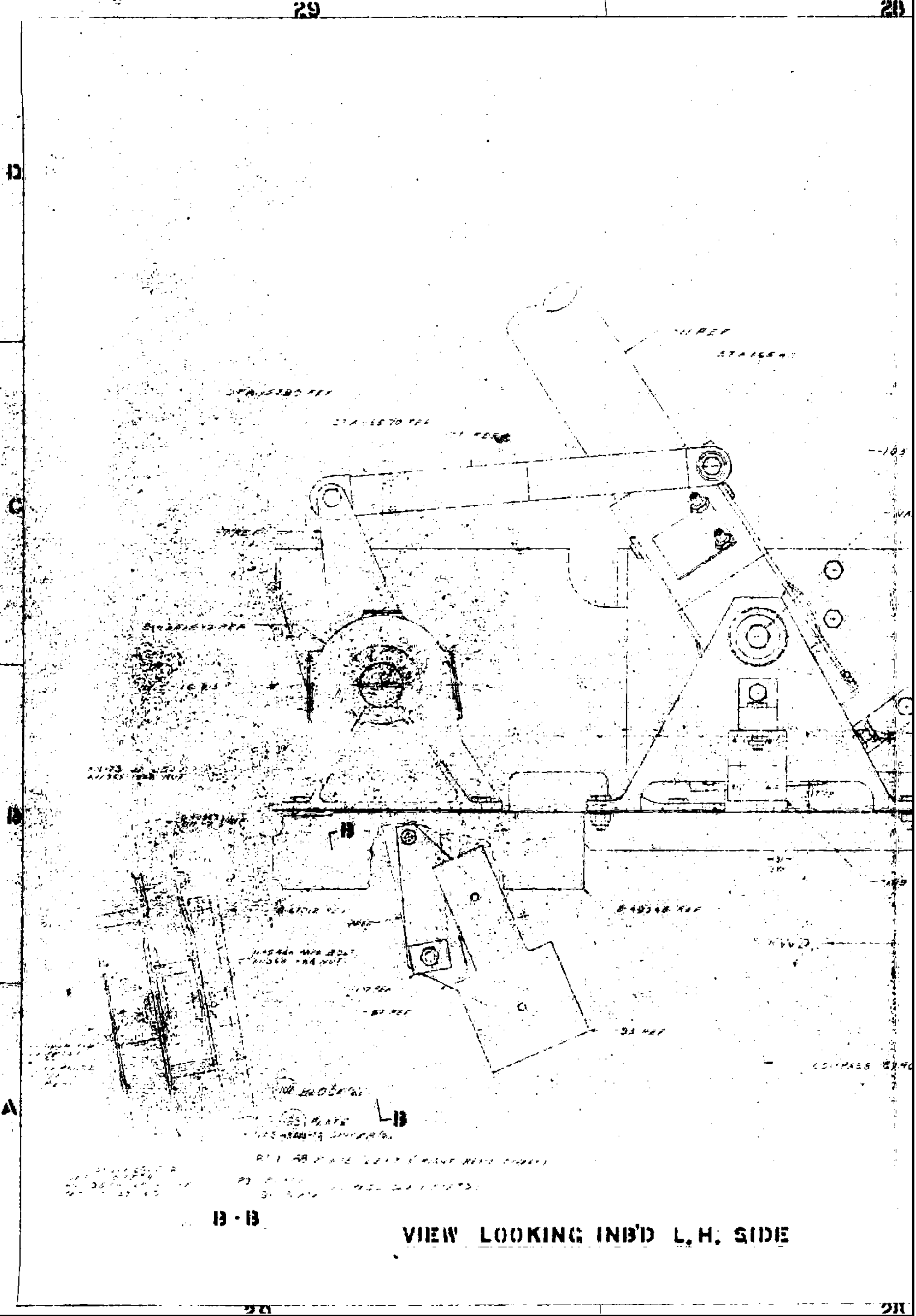
T - PROOF LOAD

)	10	60	10	70	10	80	10	90	10	100	10
5	0	56	1.5	59	1	61	1	63.5	1.5	64.3	3
5	2.3	52	1	55.3	1.5	58	.8	60	.5	62.5	.3
5	.3	57.5	.8	63.5	.8	68.5	1.3	74.5	1.8	77.5	1.8
	4.5	57	4.5	63	4.5	68	5	73	5	77	5
5	.5	55.5	.5	62.5	1	65	1	69	2	73	2.5
	1	64.5	1.5	65.5	3	66	2	66.5	2.8	66.5	3
	0	6.9	0	6.9	.1	6.9	0	7	0	6.9	.1
)	0	7	0	7	.1	7	0	7	0	7	0
5	1.5	63.5	3	63.5	1	63.5	2.5	63.5	2.5	63.5	2.8
	2	6	.1	6	-.8	6	0	6.8	0	6.5	0
	.1	7.5	.5	7.5	.3	7.5	.3	6.8	.1	7.0	.3
3	2	63	1	63	2	64.5	1.5	64.5	2	65	2.5
3	.1	6.8	.1	6.8	0	6.8	.1	6.8	.1	6.8	.1
3	.1	6.8	.1	6.8	0	6.8	.1	6.8	.1	6.8	.1
	.5	8.5	.7	10.9	.7	12.8	.9	16.1	1.6	18.6	1.6
3	.2	7.3	.3	9.6	.3	12.2	.4	14.8	.4	18.5	.4
4	2.6	23.9	2.5	23.6	2.3	24	2.4	24	2.5	24	2.4
.5	2.8	24.5	2.8	24.5	2.8	24.5	2.8	24.5	2.7	24.5	2.5
	2.8	25	2.8	25	2.8	25	2.8	25	2.7	25	2.5
8	.1	10.8	.2	11	.2	11.1	.3	11.2	.2	11.4	.4

TABLE II

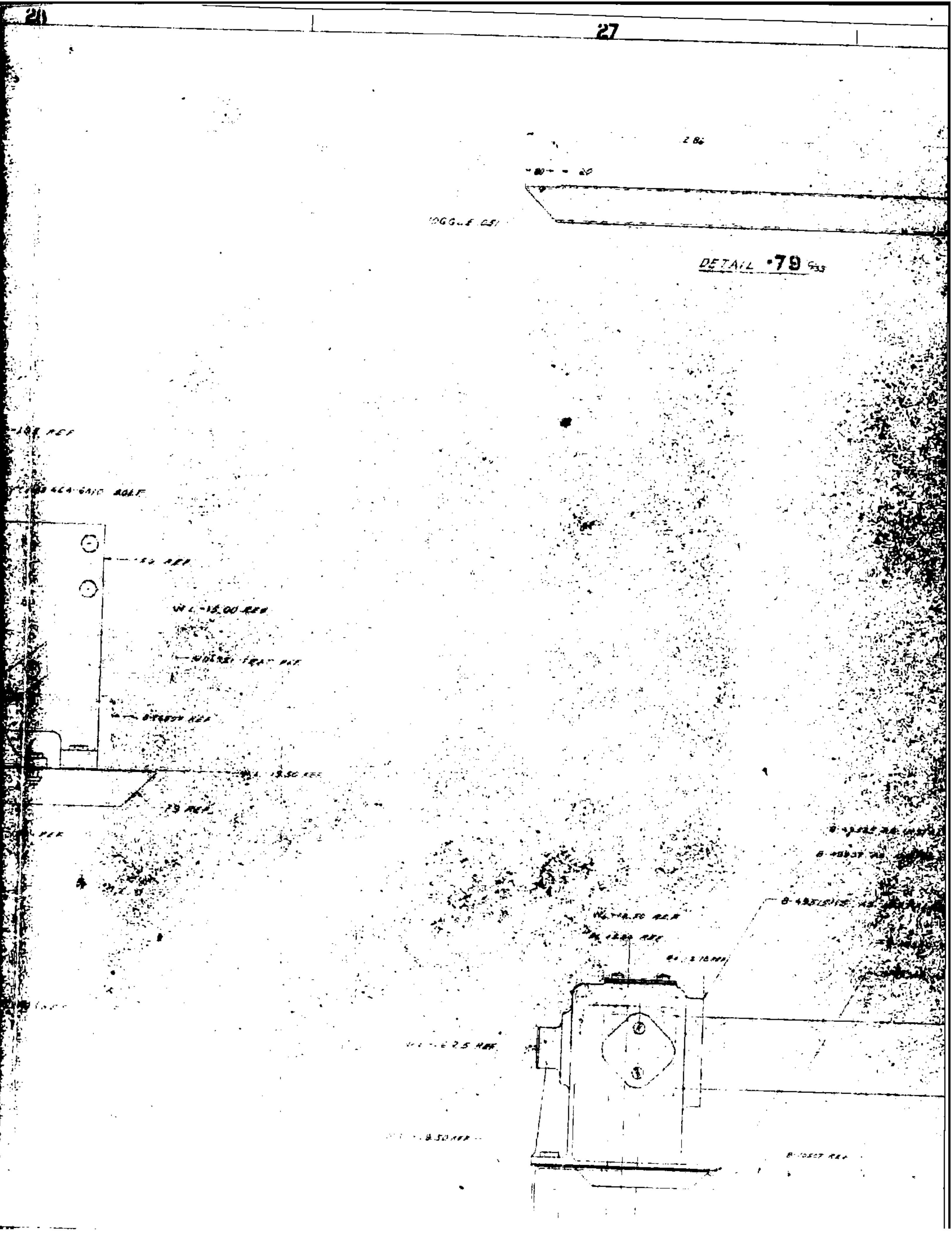
SET VERSUS DEFLECTION VALUES

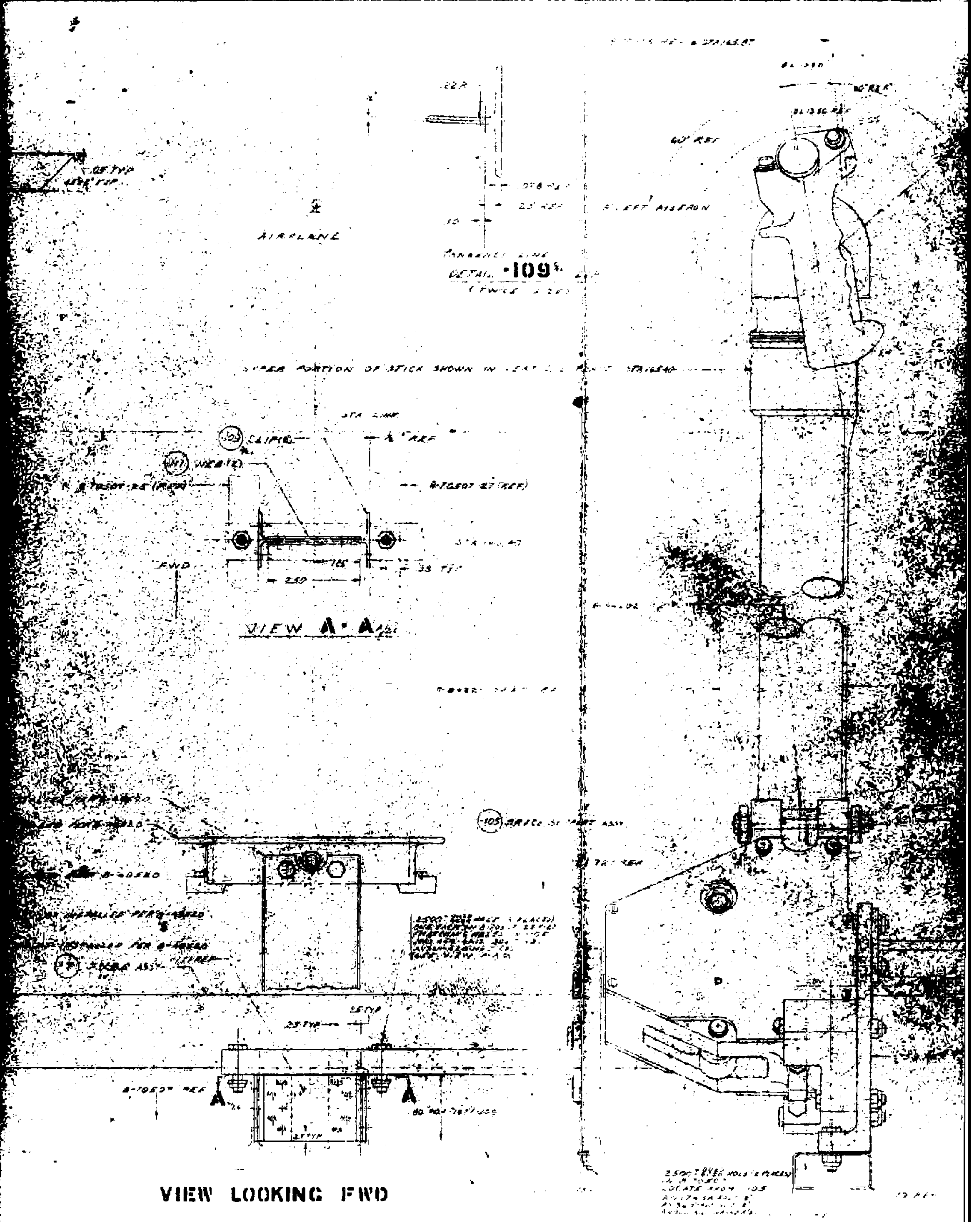
SYSTEM	CONDITION NUMBER	HYDRAULIC POWER	DEFLECTION AT 100%		SET AT 10%		PERCENT SET/DEF.
			DEGREES	INCHES	DEGREES	INCHES	
AILERON	A-1	OFF	64	-	3.0	-	4.70
	A-2	OFF	63	-	0.25	-	0.40
	A-3	OFF	78	-	1.75	-	2.24
	A-3A	OFF	77	-	5.0	-	6.49
	A-4	OFF	73	-	2.5	-	3.42
	A-5	ON	67	-	3.0	-	4.48
	A-5A	ON	64	-	2.75	-	4.30
AILERON	A-6	ON	65	-	2.50	-	3.85
ELEVATOR	B-1	OFF	18.6	6.86	1.6	0.59	8.60
	B-2	OFF	18.5	6.46	0.4	0.14	2.16
	B-3	ON	24.0	8.38	2.4	0.84	10.00
ELEVATOR	B-4	ON	11.4	3.98	0.4	0.14	3.50

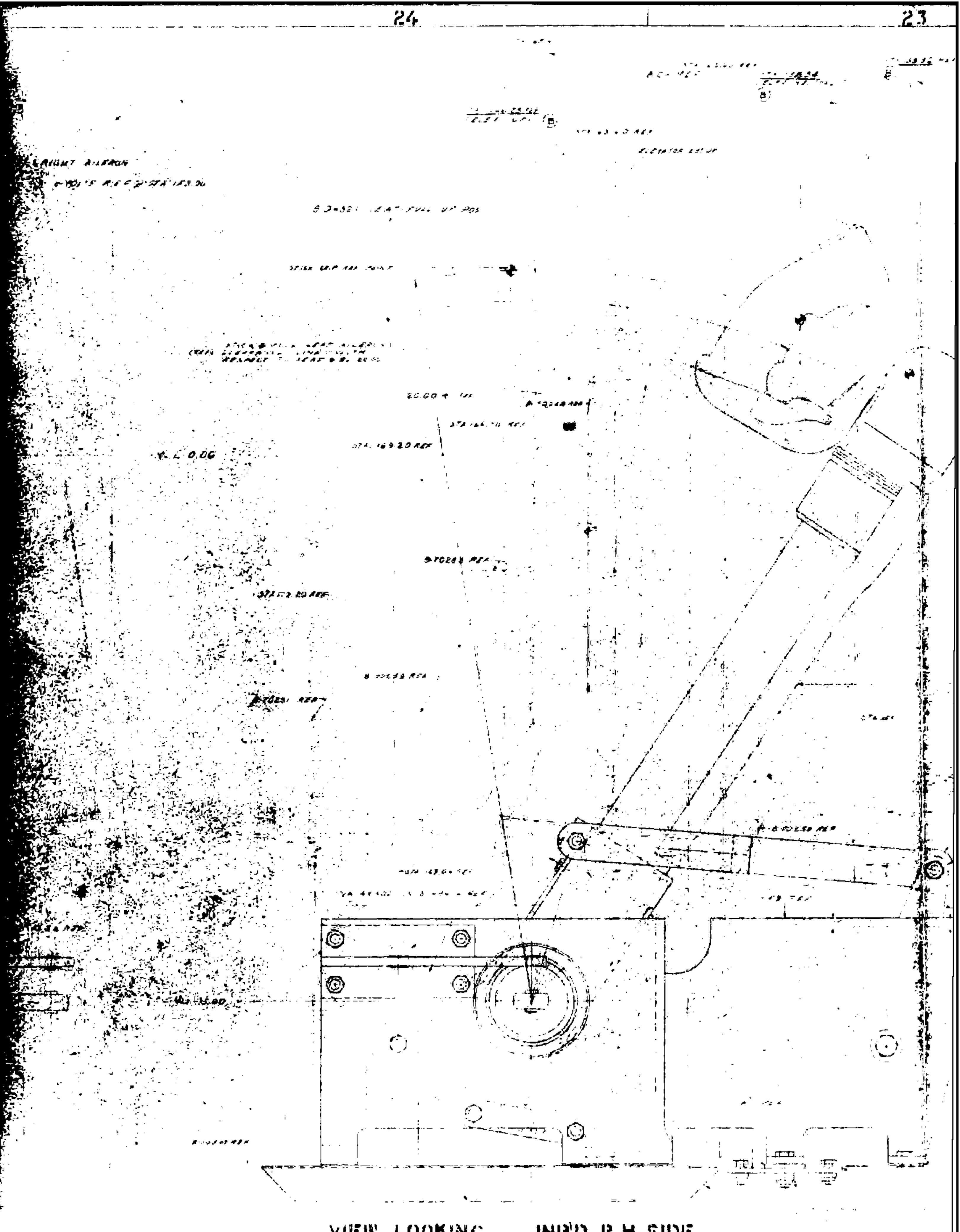


B - B

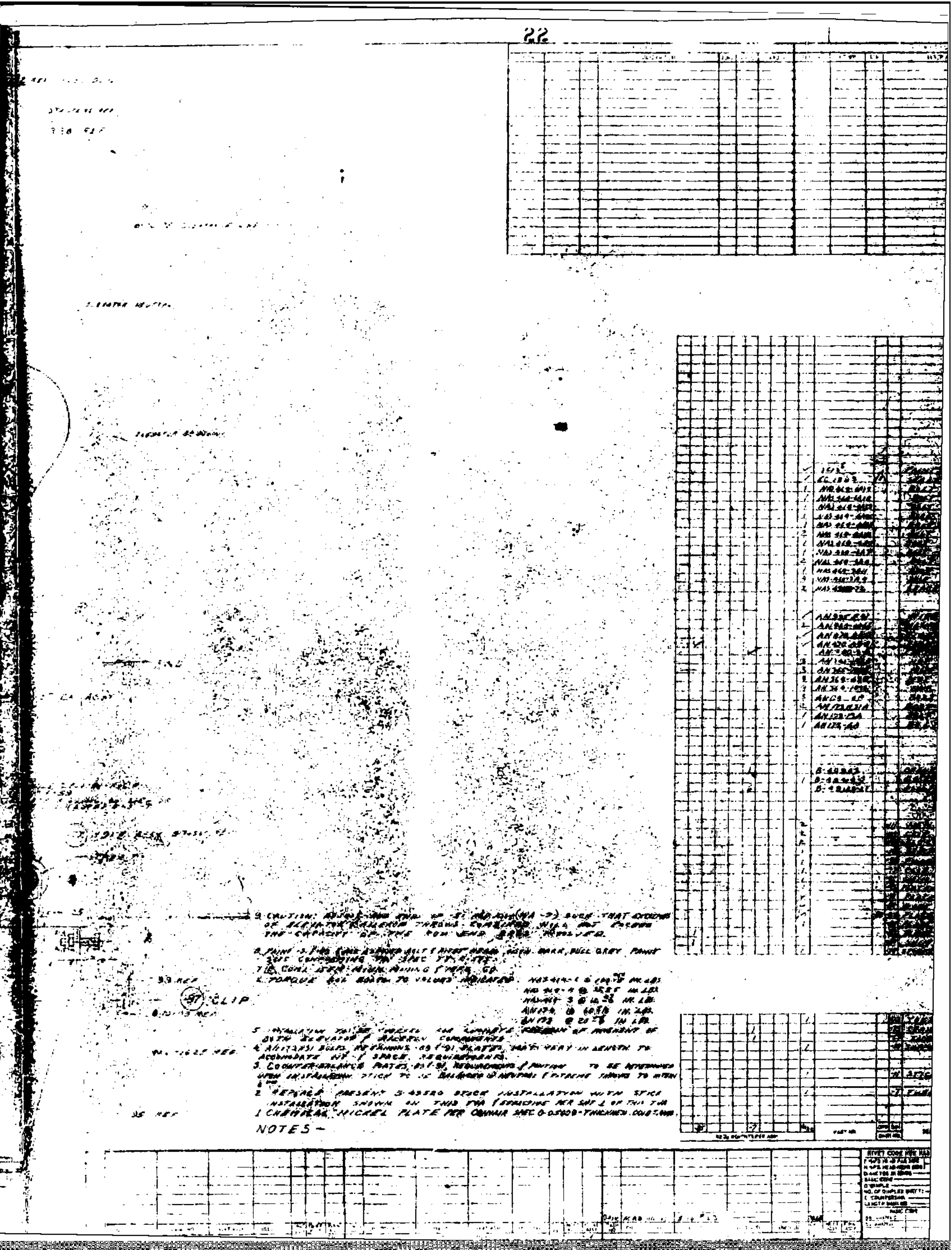
VIEW LOOKING INB'D L,H, SIDE

21
27
2.86
.80
.20
JOGGLE .05
DETAIL 79
REF
W.L. -15.00 REF
W.L. 19.50 REF
.79 REF
REF
W.L. -8.50 REF
REF
B-40537
W.L. 2.5 REF
B-10537 REF
8.50 REF


AIRPLANE
DETAIL -109
(TWICE SIZE)
TANGENCY LINE
LEFT AILERON
60° REF
UPPER PORTION OF STICK SHOWN IN LEFT STA
STA LINE
-105 CLIPIC
-117 WEB (2)
A REF
FWD
2.50
.15
.35 TYP
VIEW A-A
-105 BRACE SUPPORT ASSY
2.5 TYP
.25 TYP
A
A
VIEW LOOKING FWD



VIEW LOOKING INB'D R.H SIDE



9. CAUTION: ADJUST [illegible] (WA -9) SUCH THAT LOADS OF ELEVATOR & AILERON THROWS COMBINED WILL NOT EXCEED THE CAPACITY OF THE ROD END BRG. INVOLVED.

8. PAINT -3 [illegible] BOLT & RIVET HEADS [illegible] DARK, DULL GREY PAINT [illegible] CONFORMING TO SPEC [illegible]

7. [illegible] COAT ASSEM. [illegible] GO.

6. TORQUE BOLT & NUTS TO VALUES INDICATED. [illegible] IN. LBS.
[illegible] IN. LBS.
[illegible] IN. LBS.
AN174 @ [illegible] IN. LBS.
AN173 @ [illegible] IN. LBS.

5. INSTALLATION THIS [illegible] FREEDOM OF MOVEMENT OF BOTH ELEVATOR & AILERON COMPONENTS

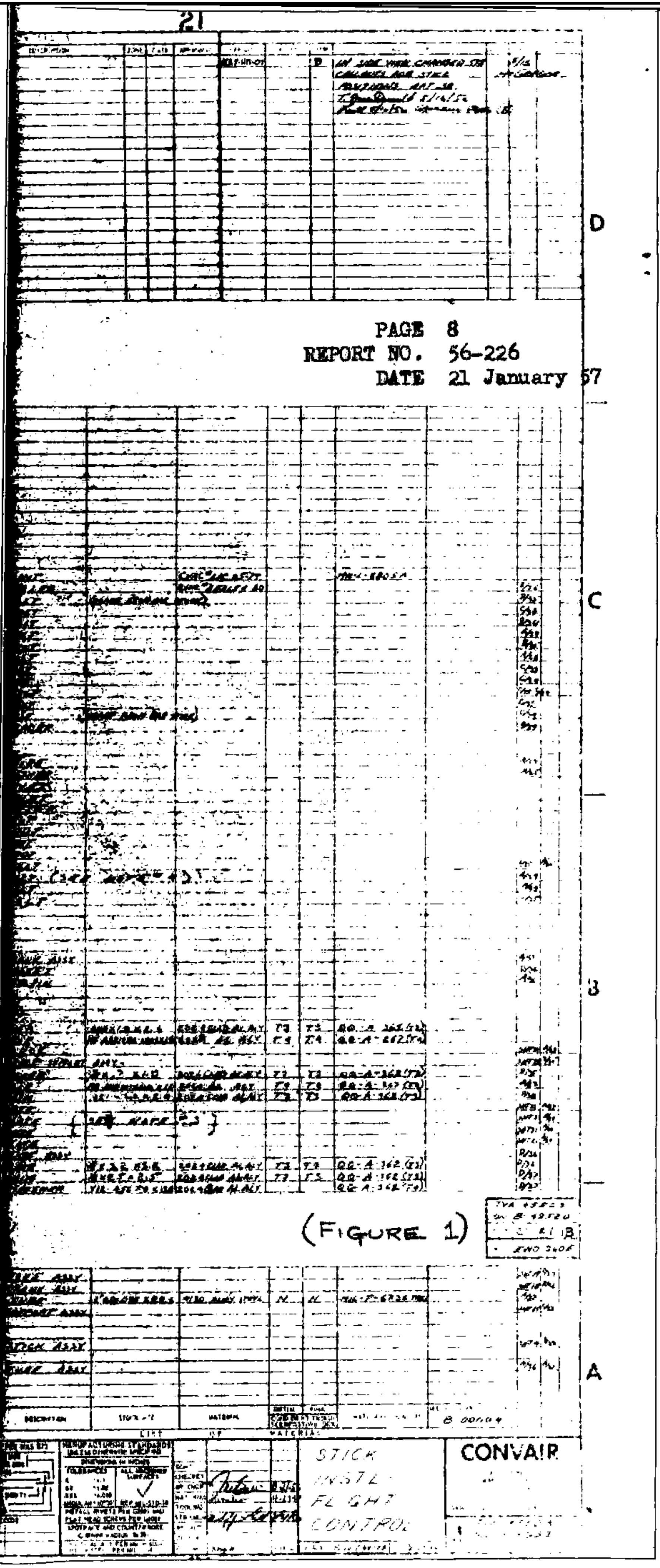
4. AN [illegible] BOLTS [illegible] -89 -91 PLATES, MAY VARY IN LENGTH TO ACCOMMODATE WT & SPACE REQUIREMENTS.

3. COUNTER-BALANCE PLATES, -89 -91, REQUIREMENTS & POSITION TO BE DETERMINED UPON INSTALLATION. STICK TO BE BALANCED @ NEUTRAL & EXTREME THROWS TO WITHIN [illegible]

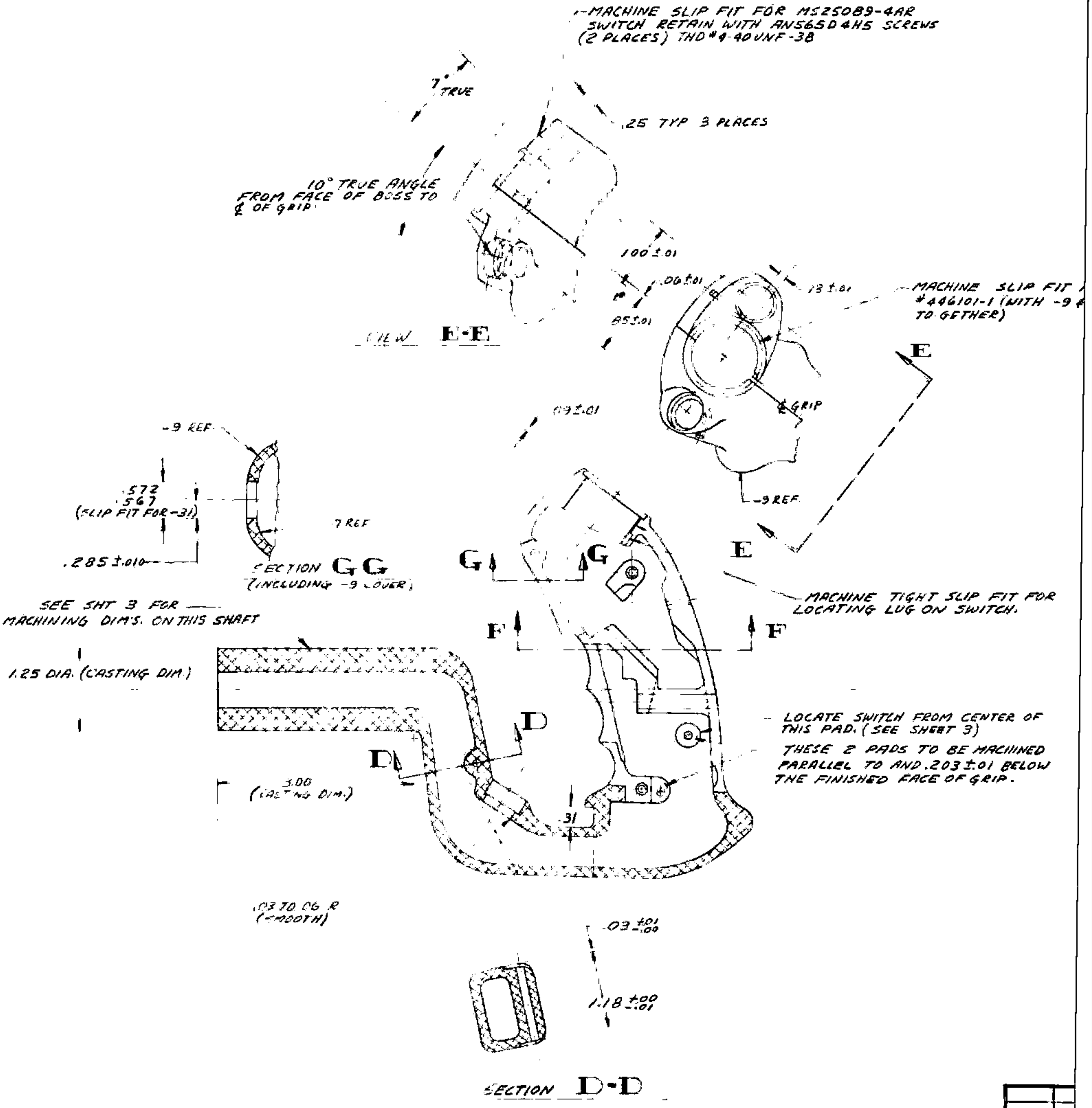
2. REPLACE PRESENT 5-45580 STICK INSTALLATION WITH STICK INSTALLATION SHOWN ON THIS DWG (SPECIFIC PER SHT 2 OF THIS DWG

1. CHEMICAL NICKEL PLATE PER CONVAIR SPEC 0-05008-THICKNESS .0015 MIN.

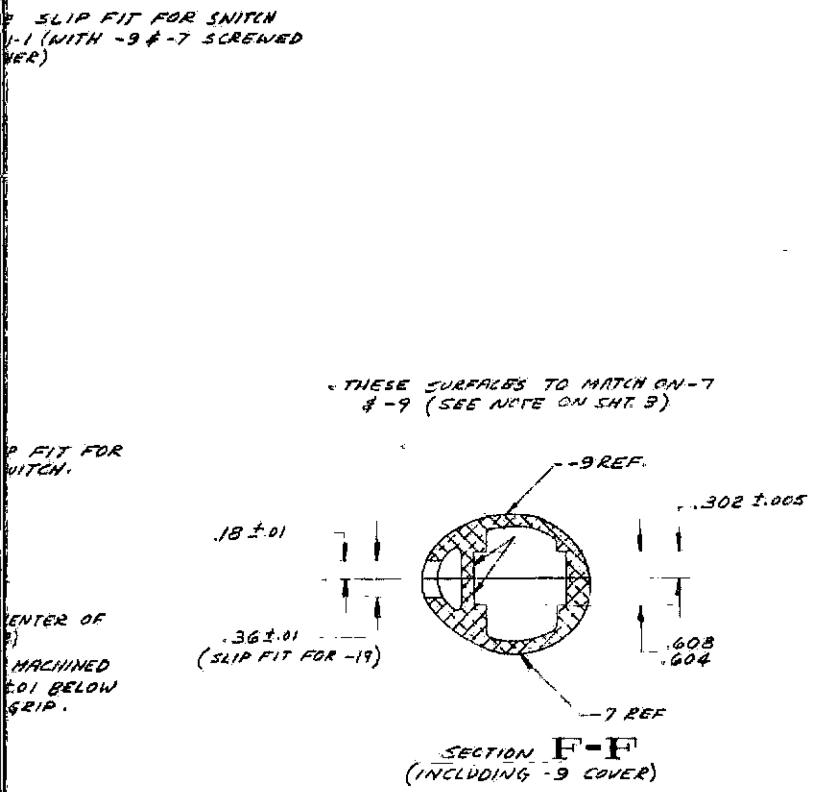
NOTES —



(FIGURE 1)


MACHINE SLIP FIT FOR MS25089-4AR SWITCH RETAIN WITH AN565D4H5 SCREWS (2 PLACES) THD #4-40UNF-3B
7° TRUE
.25 TYP 3 PLACES
10° TRUE ANGLE FROM FACE OF BOSS TO ℄ OF GRIP
.100 ±.01
.06 ±.01
.13 ±.01
.85 ±.01
MACHINE SLIP FIT #446101-1 (WITH -9 TO GETHER)
VIEW E-E
E
℄ GRIP
.09 ±.01
-9 REF
E
-9 REF
.572
.567
(SLIP FIT FOR -31)
.285 ±.010
7 REF
SECTION G G
(INCLUDING -9 COVER)
G
G
MACHINE TIGHT SLIP FIT FOR LOCATING LUG ON SWITCH.
SEE SHT 3 FOR MACHINING DIM'S. ON THIS SHAFT
F
F
1.25 DIA. (CASTING DIM.)
D
D
LOCATE SWITCH FROM CENTER OF THIS PAD. (SEE SHEET 3)
THESE 2 PADS TO BE MACHINED PARALLEL TO AND .203 ±.01 BELOW THE FINISHED FACE OF GRIP.
3.00 (CASTING DIM.)
.31
.03 TO .06 R (SMOOTH)
.03 +.01 -.00
.18 +.00 -.01
SECTION D-D

GOVT	
[illegible]	
[illegible]	[illegible]
MA[illegible]	
[illegible]	
STR[illegible]	3[illegible]
WE[illegible]	
DE[illegible]	
DFTSMN	[illegible]



EWO 3605 ACA 8-1111-01

GOVT
CHECKED
MATERIAL
WEIGHTS
DESIGN
DFTSMN V. Olson 3-22-56

GRIP ASSY -
SIDE CONTROL
STICK.

SCALE FULL
REL

CONVAIR
A DIVISION OF
GENERAL DYNAMICS
CORPORATION
SAN DIEGO CALIFORNIA

DWG SIZE D
TVA 47623 ON
B-49520
SHEET 4 OF 4

4-10-56

FIGURE 2

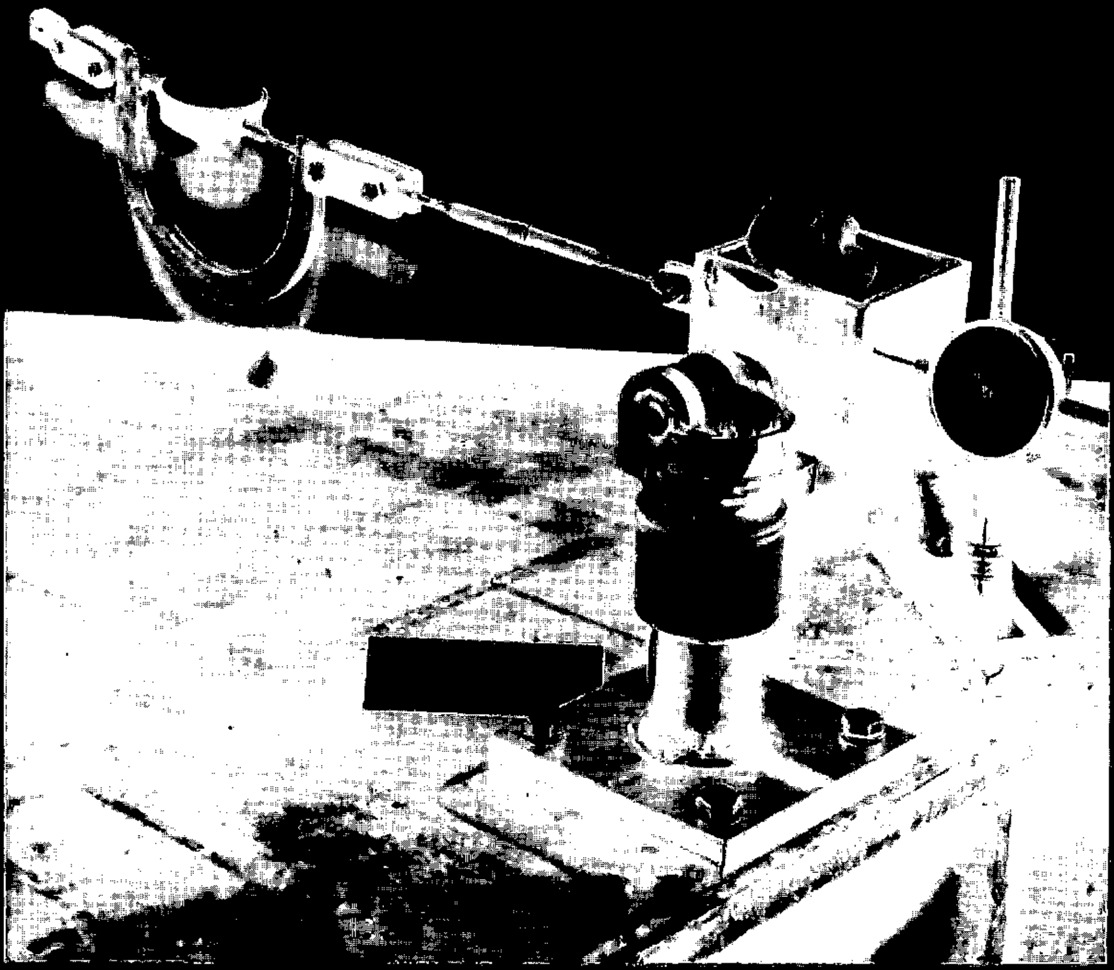


Figure 6 GRIP TEST - CONDITION 1

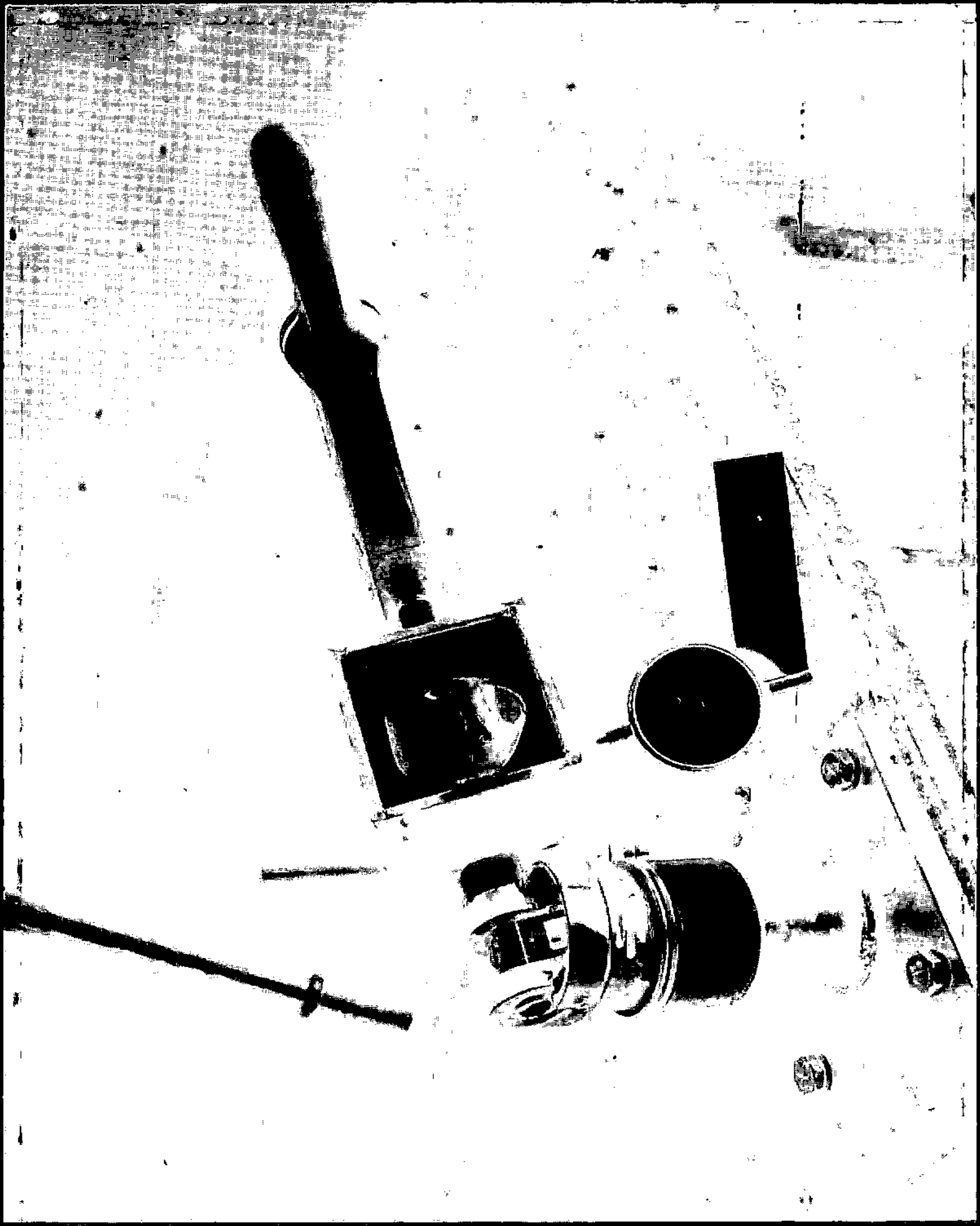


Figure 7 GRIP TEST - CONDITION 2

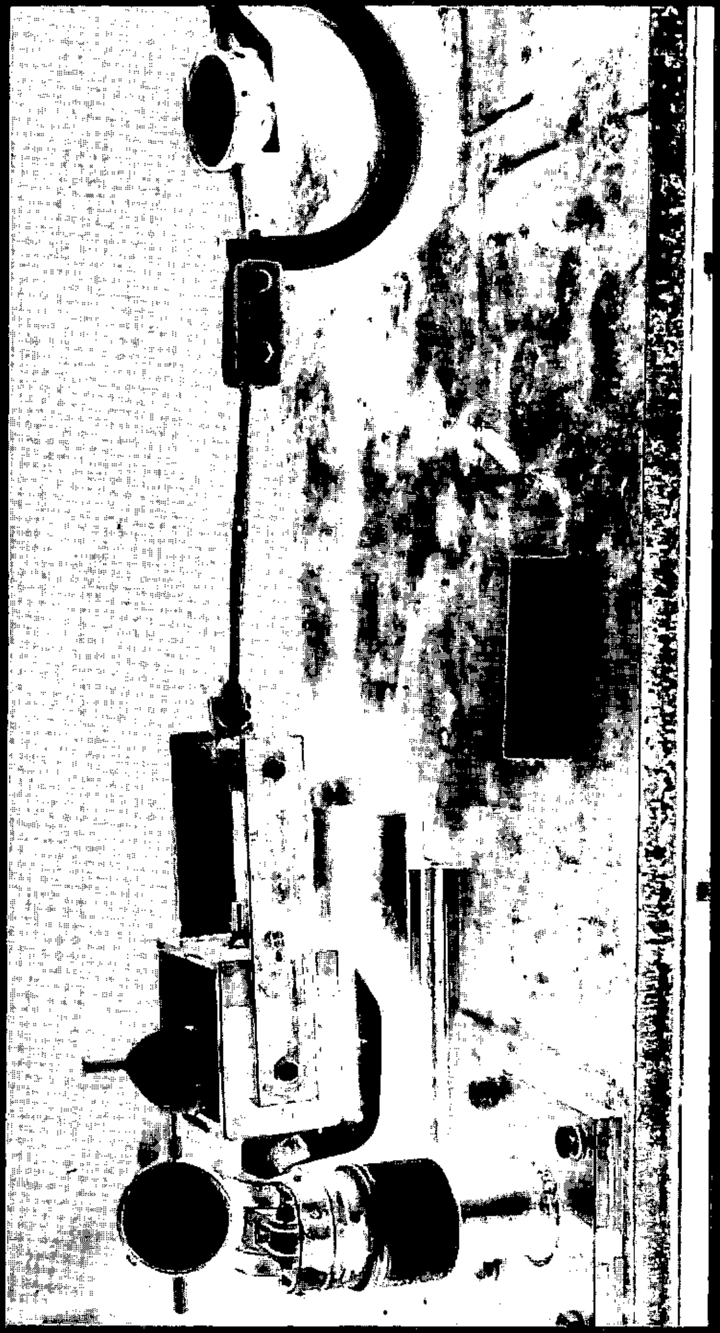


Figure 8 GRIP TEST - CONDITION 4

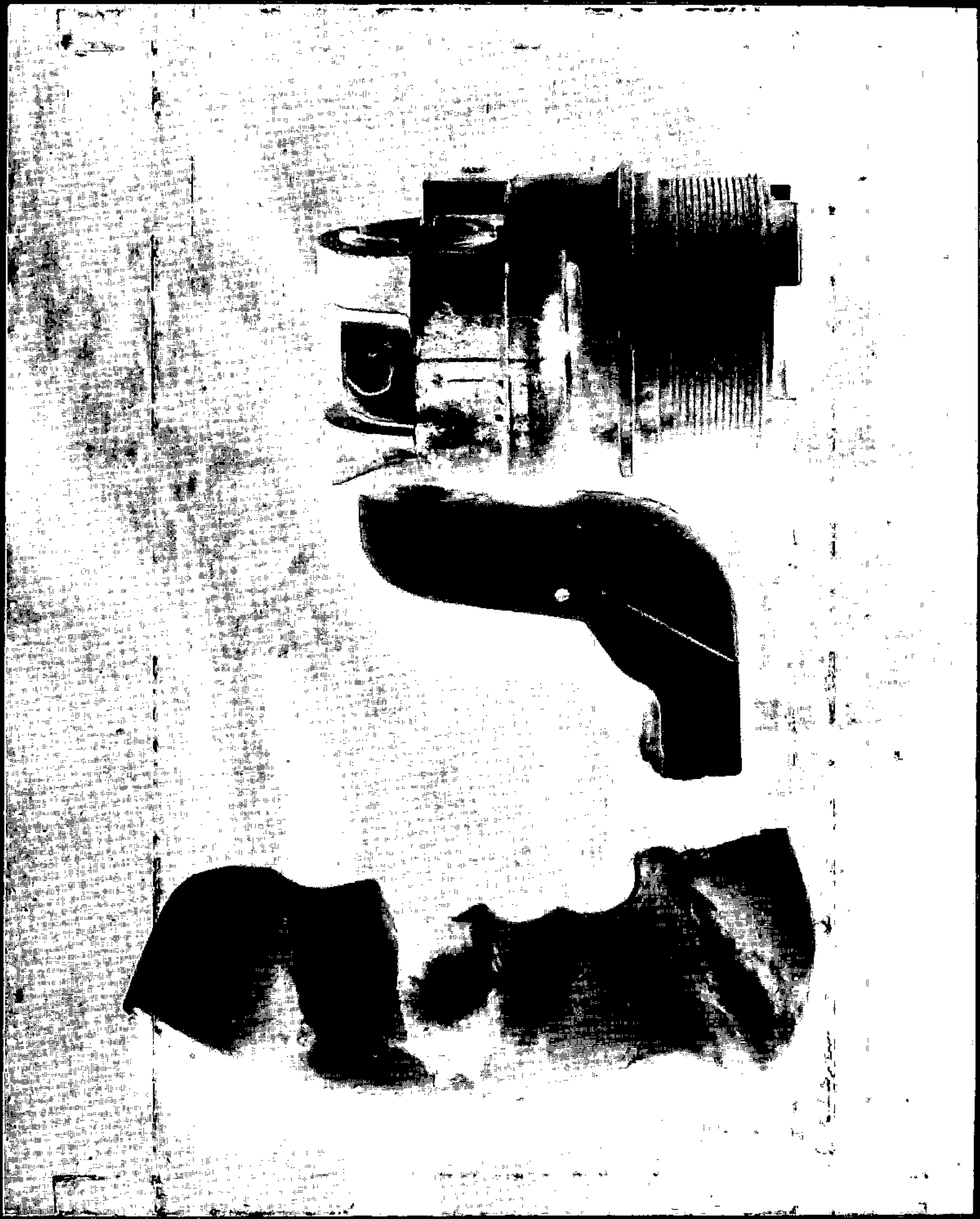
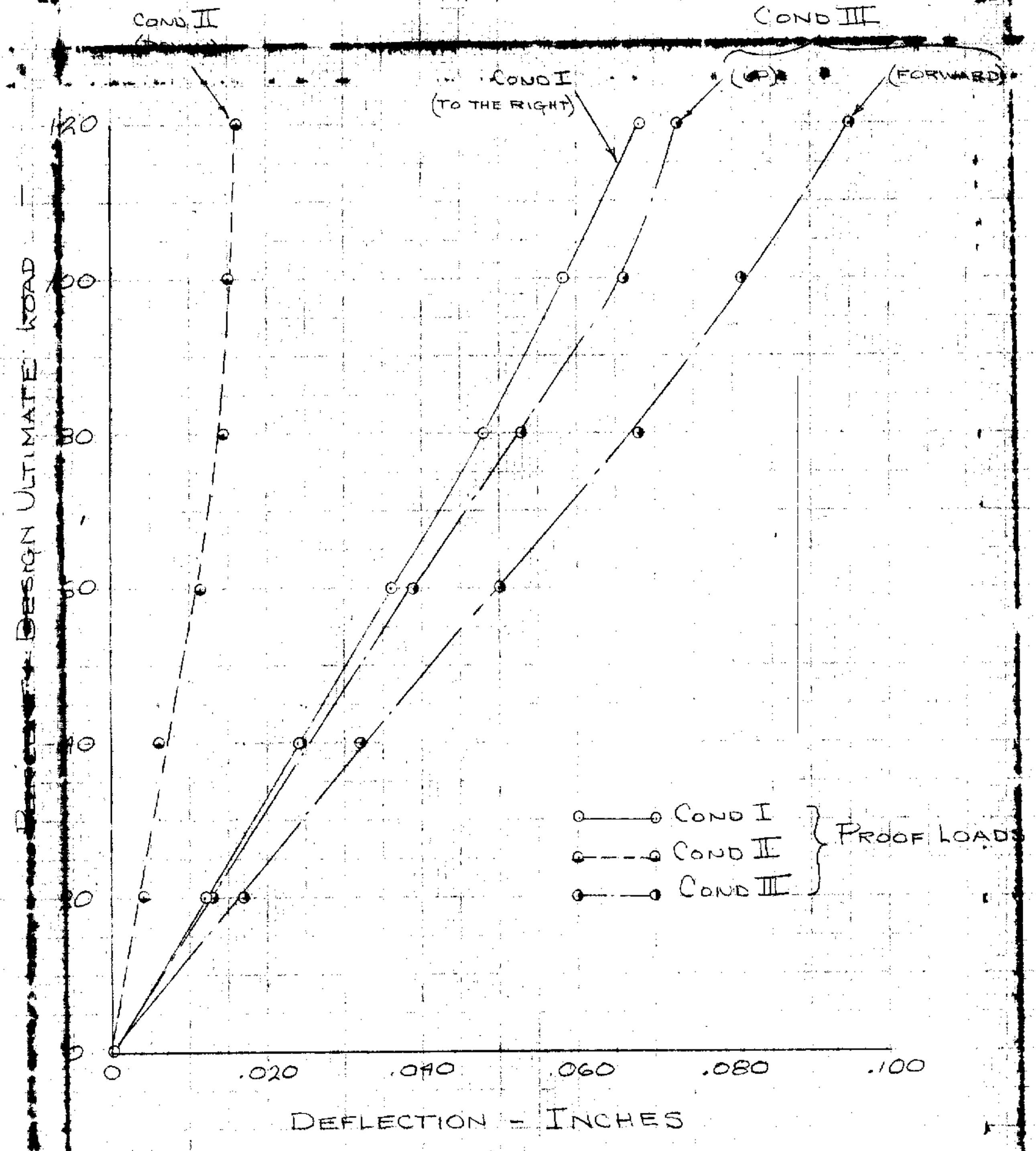


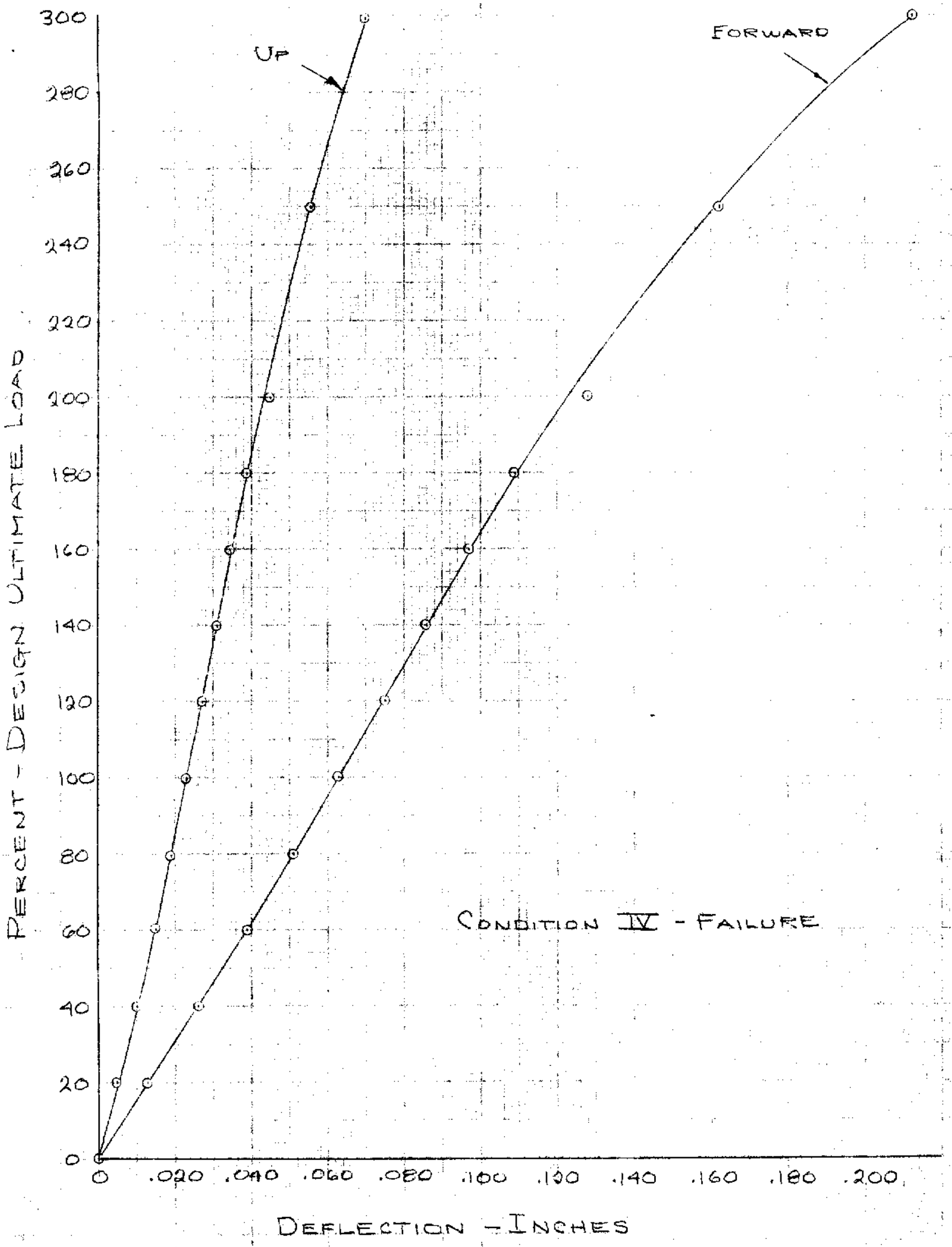
Figure 9 GRIP TEST FAILURE



CALCULATED BY	J.M.	11-8-56	LOAD - DEFLECTION CURVES GRIP ASSY. - SIDE CONTROL STICK [illegible]	PAGE
TRACED BY	J.M.	11-8-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	
APPROVED BY				MODEL F-102A
APPROVED BY				

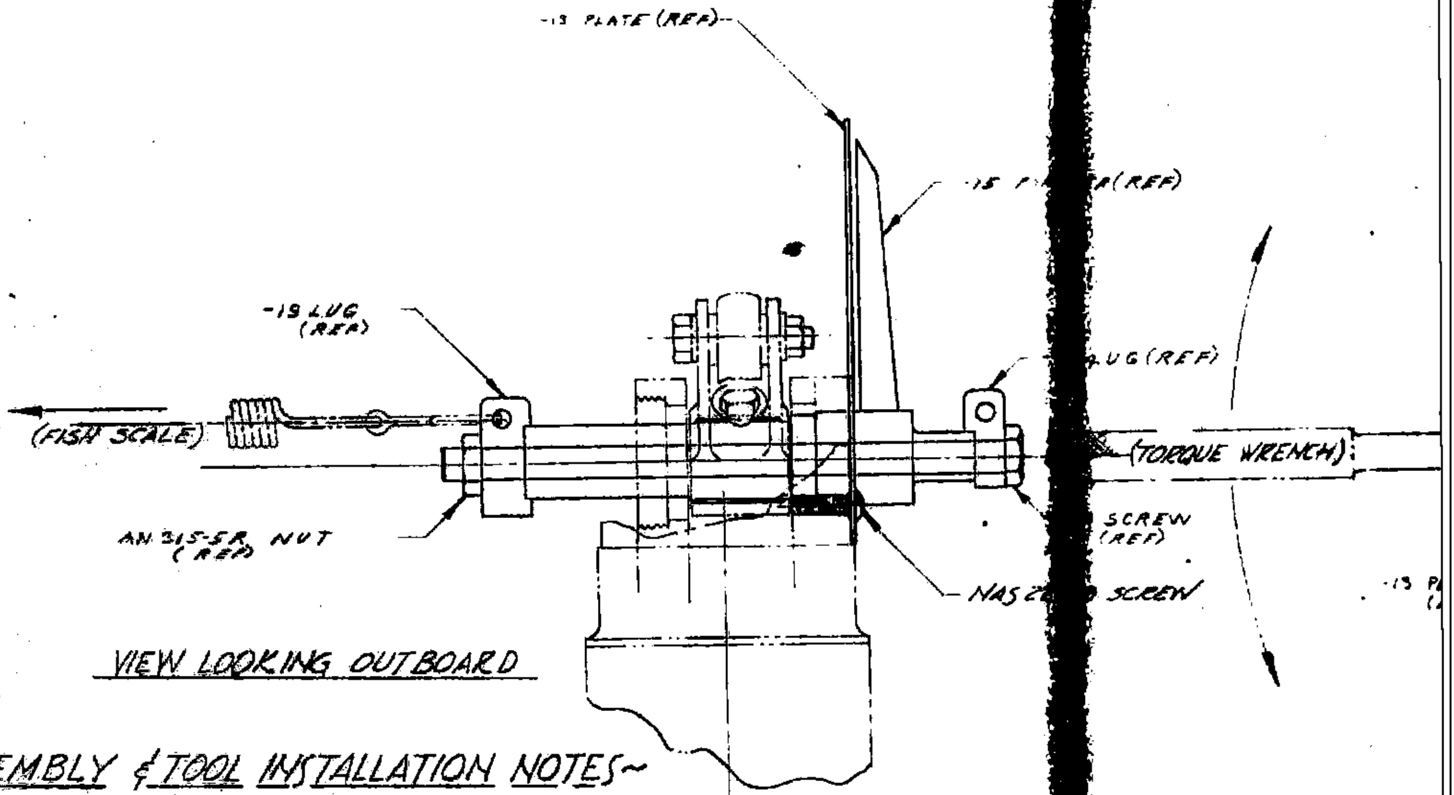
FORM 150-A

FIGURE 10



CALCULATED BY	J.M.	11-8-56	LOAD-DEFLECTION CURVES GRIP ASSY. - SIDE CONTROL STICK TVA 47623 ON 8-49520	PAGE
TRACED BY	J.M.	11-8-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	MODEL
APPROVED BY				F-102A
APPROVED BY				

FIGURE 11



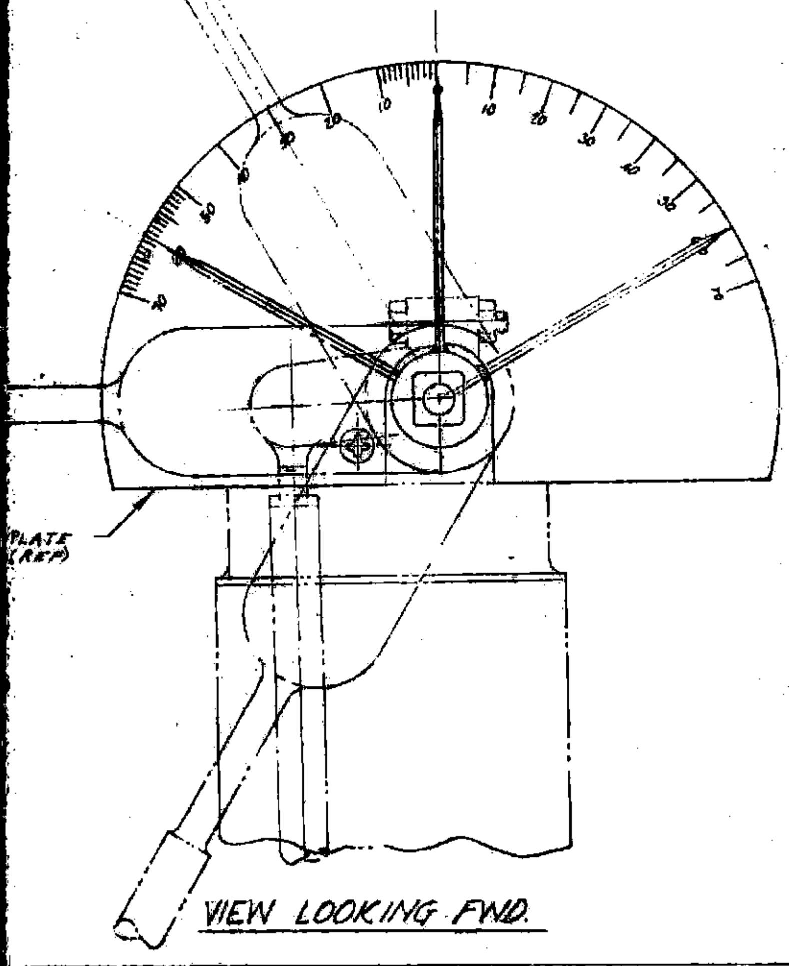
VIEW LOOKING OUTBOARD

ISASSEMBLY & TOOL INSTALLATION NOTES~

EFERENCE: SEE TVA47623 ON 8-04393 SHEET 2 FOR STICK HEAD
& GRIP ASSEMBLY DETAILS.)

WITH RIG PIN INSTALLED IN STICK BASE, PERFORM THE FOLLOWING.
REMOVE C'SK SCREWS THAT RETAIN -13 CONTROL STICK HEAD COVER.
REMOVE -13 CONTROL STICK HEAD COVER.
REMOVE -39 BEARING RETAINING NUT FROM HOUSING.
REMOVE AN365-428 NUT & AN960D416L WASHER FROM NAS 464-4A-15 BOLT.
REMOVE NAS 464-4A-15 BOLT
REMOVE -7 GRIP FROM HEAD ASSY. (NOTE: -15 CRANK WILL FALL FREE
BOUT ROD END BEARING WHEN GRIP IS REMOVED.)
NSERT ROUND END OF TOOL #SE1049 THRU AFT BEARING IN STICK HOUSING
POSITION -15 CRANK AND TOOL #SE1049 SUCH THAT WHEN TOOL & CRANK
PLINES ARE ENGAGED, THE GROOVE IN TOOL SPLINE WILL ALIGN WITH
LAMP UP BOLT HOLE IN CRANK & THAT -15 CRANK IS IN SAME ANGULA
OSITION AS BEFORE GRIP WAS REMOVED.
NSERT TOOL #SE1049 THRU -15 CRANK & FWD HOUSING BEARING TO DEPTH S
S TO ALLOW NAS 464-4A-15 BOLT INSTALLATION.
INSTALL NAS 464-4A-15 BOLT, AN960D416L WASHER & AN365-428 NUT.
POSITION DIAL #SE1049-13 ON AFT SIDE OF COLUMN TO STRADDLE TOOL SHAN
AND SUCH TO ALLOW INSERTION OF NAS 221-9 DIAL FASTENING SCREW IN
EXISTING TAPPED HOLE IN COLUMN HOUSING.
INSERT & TIGHTEN NAS 221 SCREW. DIAL POINTER MAY HAVE TO BE SLIGH
ENT TO ALIGN POINTER WITH ZERO DIAL READING WITH SYSTEM IN NEUTR
[illegible]
FOLLOW INSTRUCTIONS FOR ELEVON CHECKOUT ON SHTS 1-3 OF THIS TVA.
FOR ELEVATOR CHECKOUT INSTALL TOOL PARTS #SE1049-17-19 & 21 AS SHOWN.
INSERT CALIBRATED SPRING SCALE HOOK IN FWD HOOK HOLE AS SHOWN
FOR CHECKING AND STICK FORCES & IN AFT HOOK HOLE FOR AFT FORCES.
9. AFTER BOTH CHECKS ARE COMPLETED, REINSTALL GRIP AS REVERSE
PROCEDURE OF INSTRUCTIONS GIVEN ABOVE FOR DISASSEMBLY.

AKK KAR-1111-01 AK53-1819 SHT 1



VIEW LOOKING FWD.

1	NAS220-9	SCREW

CONTROL [illegible]
INSTRUCTIONS
MODEL 8-10
AIRPLANE

FULL

FIGURE [illegible]

TWA 47687 ON
8-04059

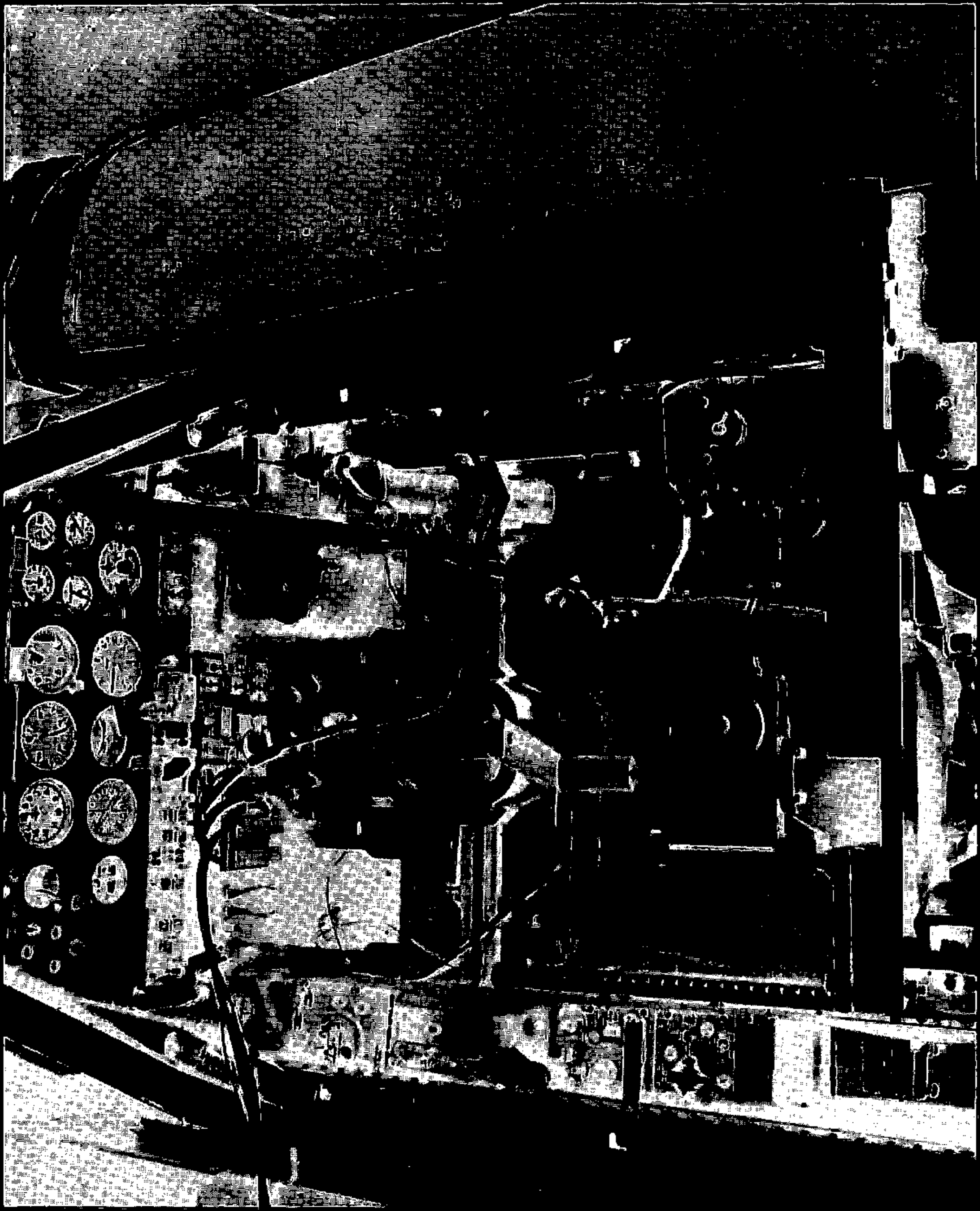


Figure 13 CONTROL TEST - AFT LOADS

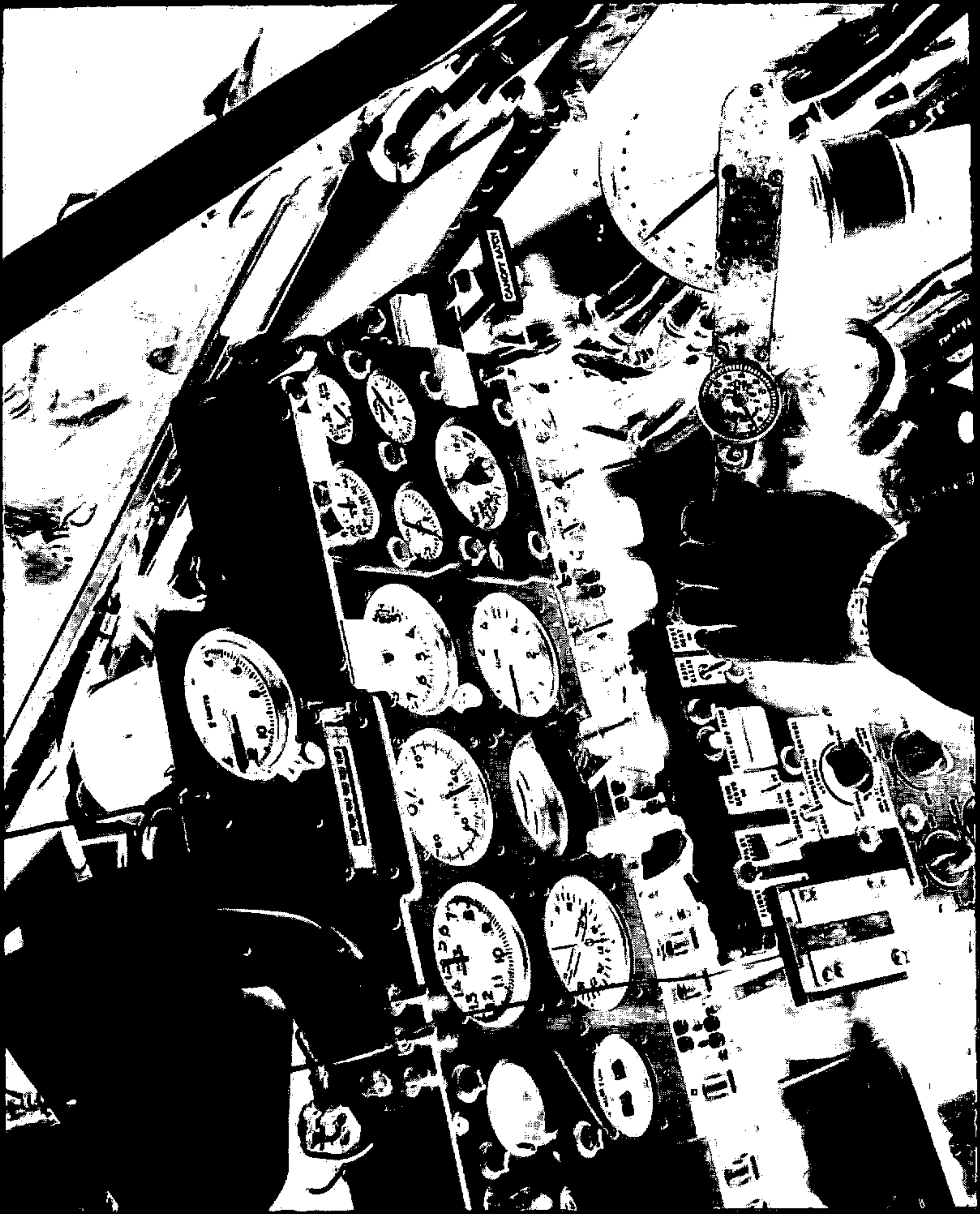


Figure 15 CONTROL TEST - TORSION LOAD

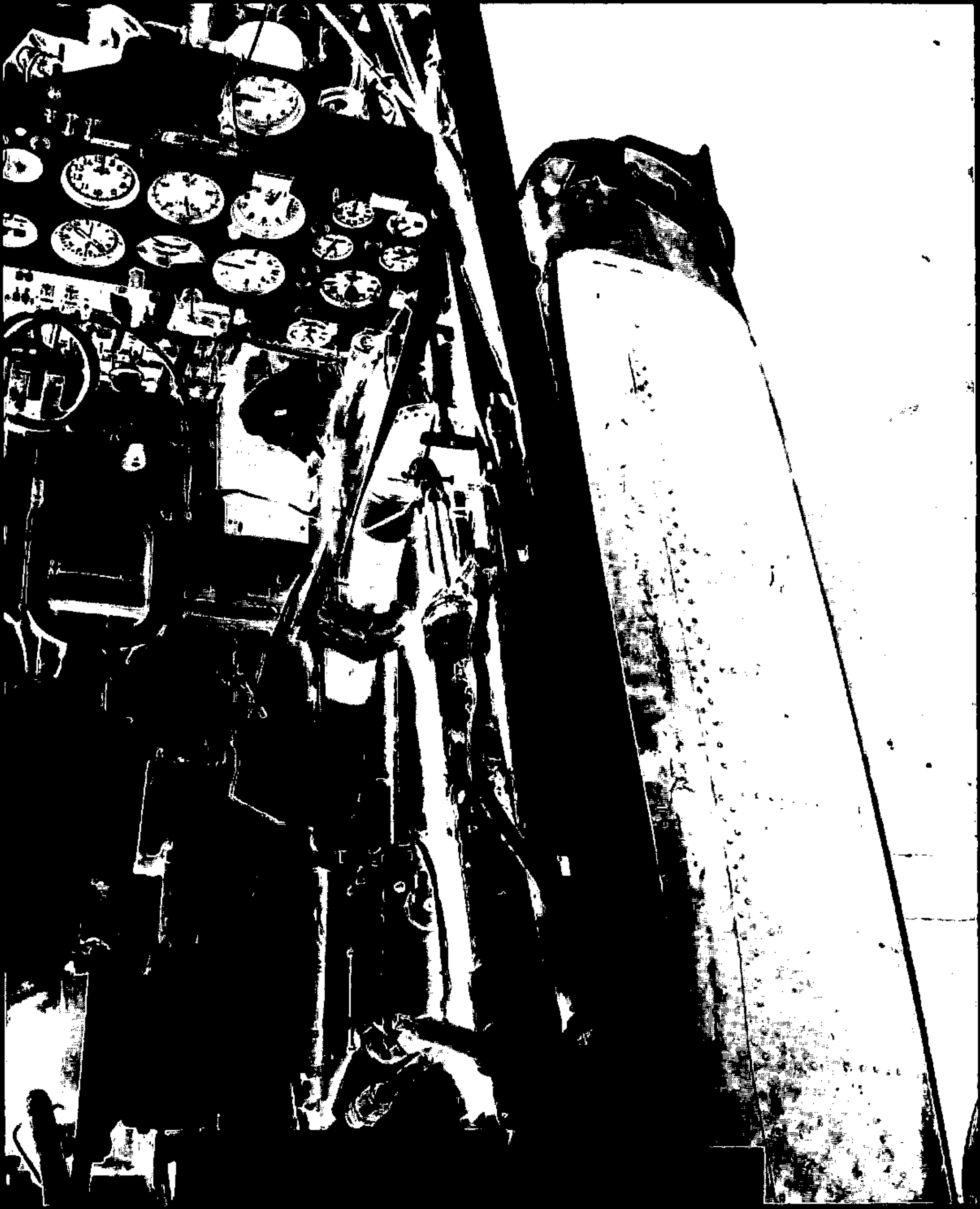
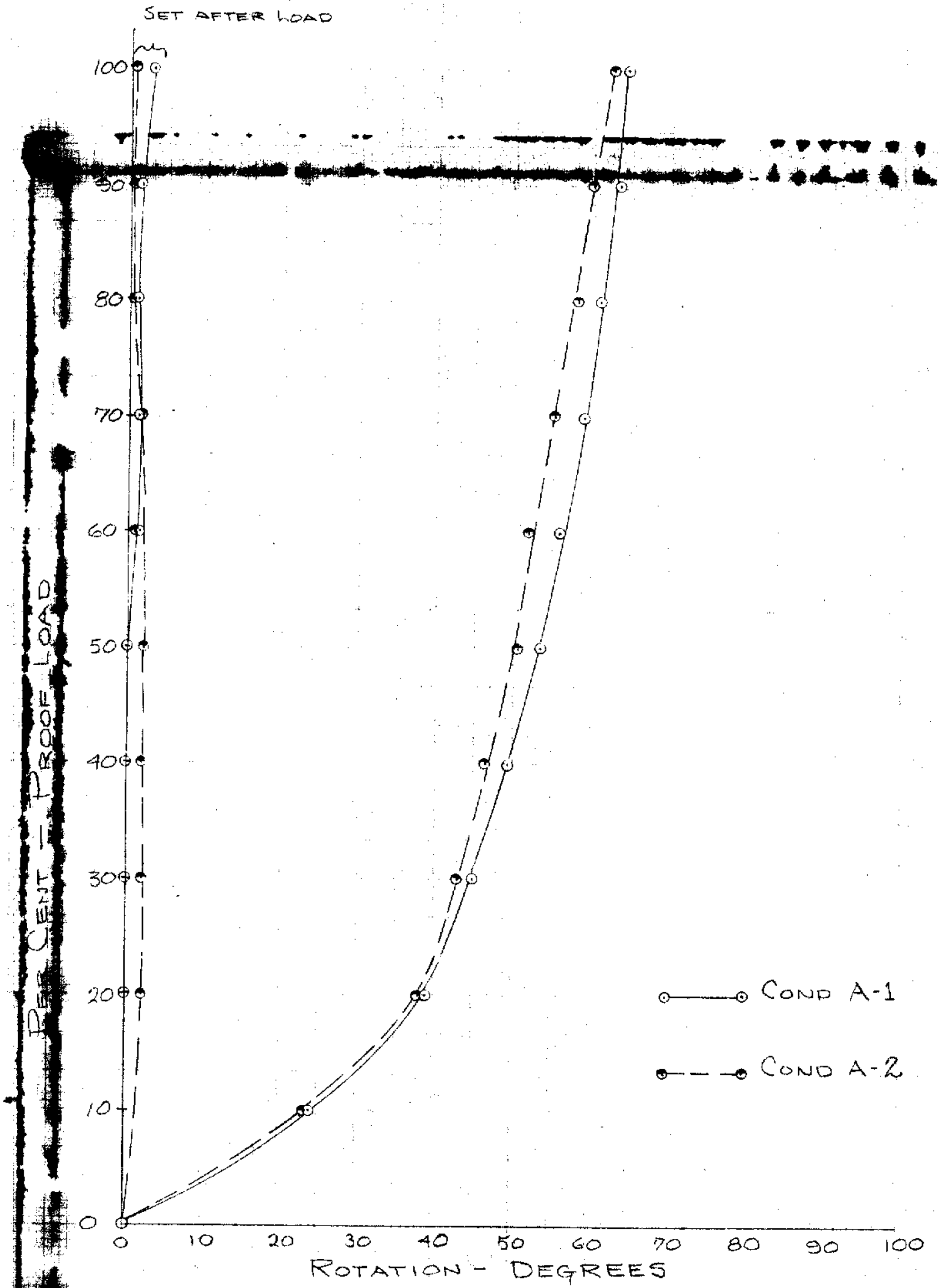


Figure 14 CONTROL TEST - FORWARD LOADS



CA[illegible]	[illegible]	[illegible]	CONDITIONS A-1 AND A-2 LOAD DEFLECTION CURVES AILERON SYSTEM	PAGE
TR[illegible]	[illegible]	[illegible]		DOC [illegible]
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	MODEL F-102A
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APPROVED BY				

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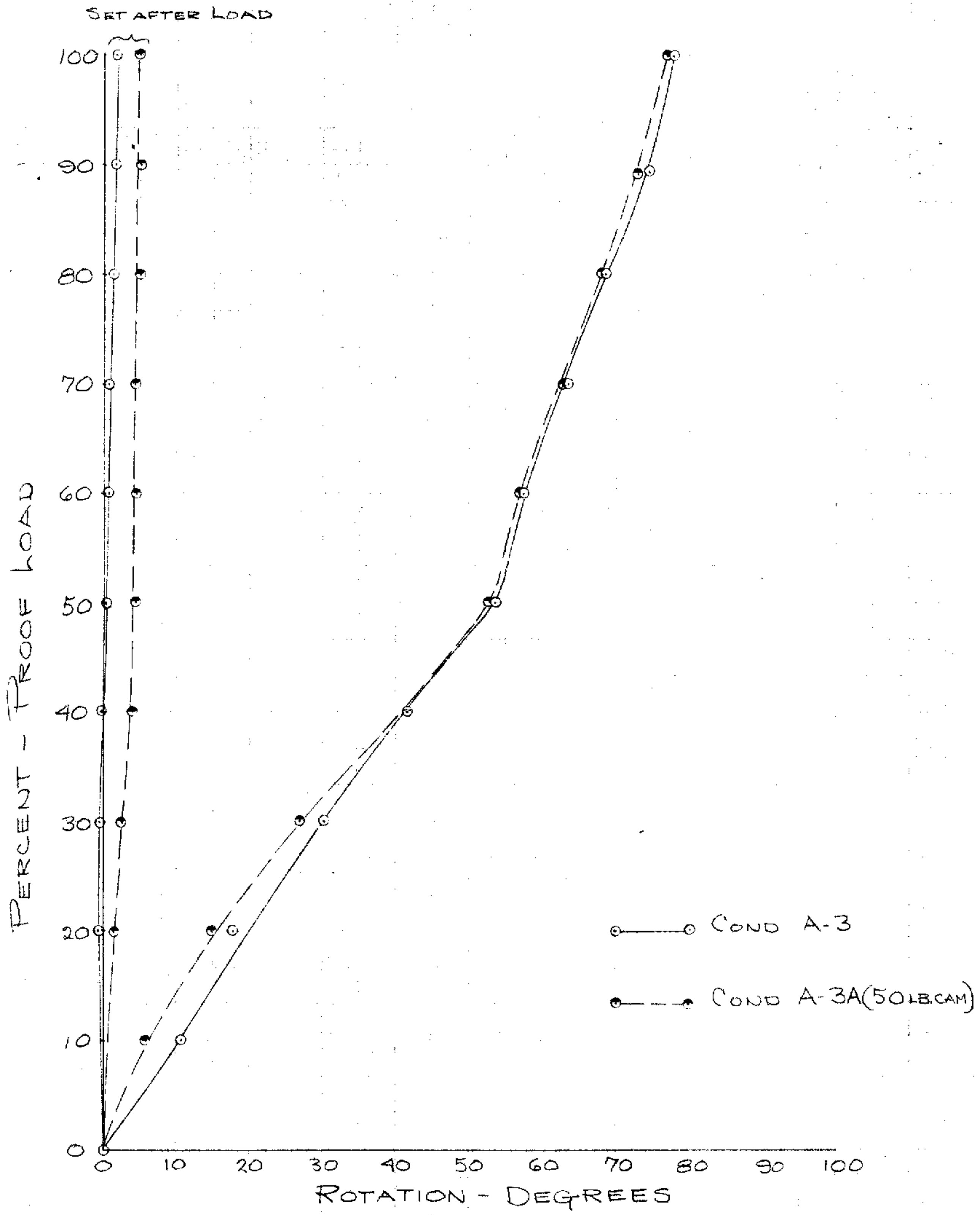
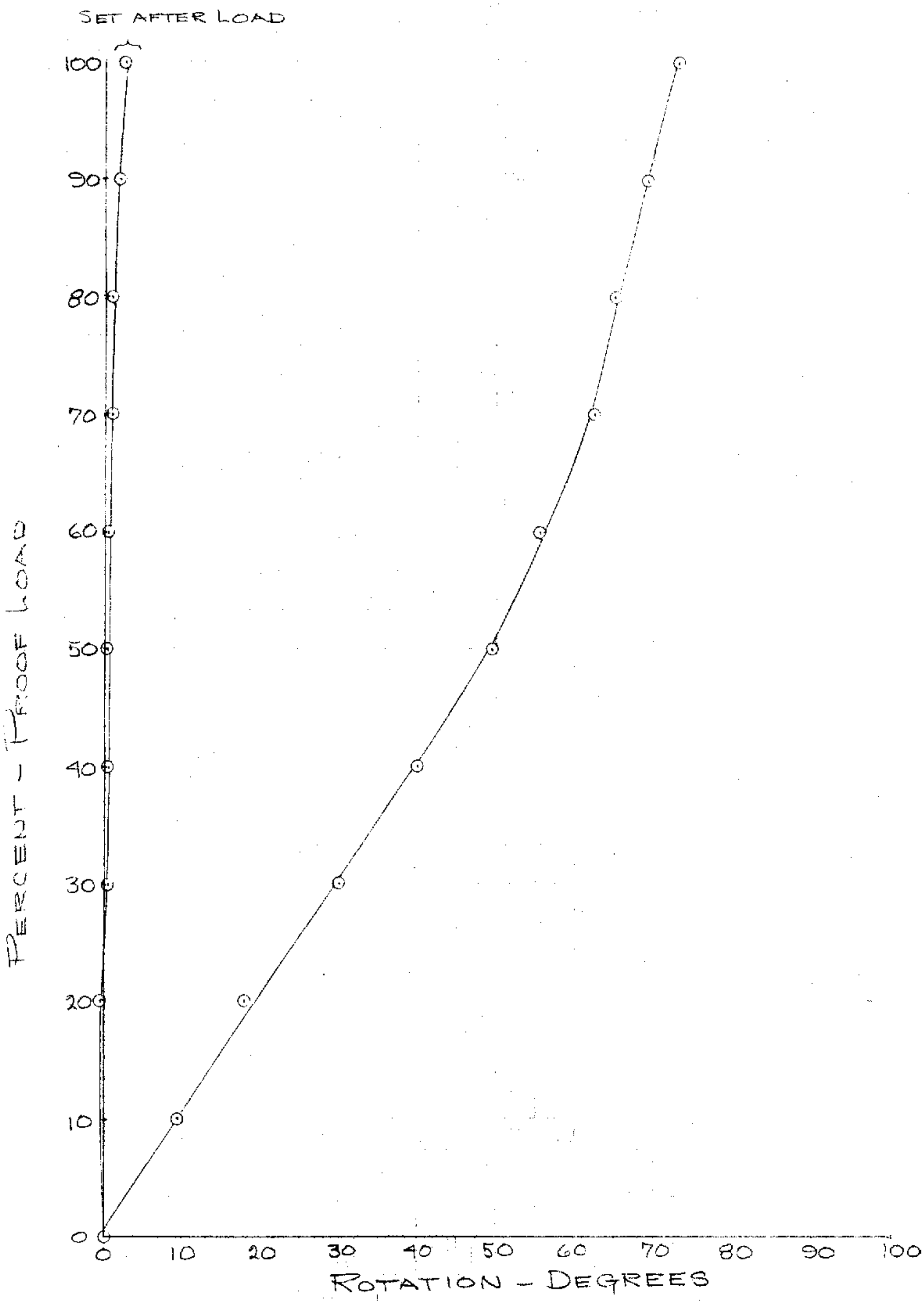
FIGURE 16



CALCULATED BY	J.M.	11-20-56	CONDITIONS A-3 AND A-3A LOAD-DEFLECTION CURVES AILERON SYSTEM	PAGE
TRACED BY	J.M.	11-20-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	
APPROVED BY				MODEL
APPROVED BY				F-102A

FORM 150-A

FIGURE 17



CALCULATED BY	J.M.	11-20-56
TRACED BY	J.M.	11-20-56
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APPROVED BY		
APPROVED BY		

CONDITION A-4
LOAD-DEFLECTION CURVES
AILERON SYSTEM

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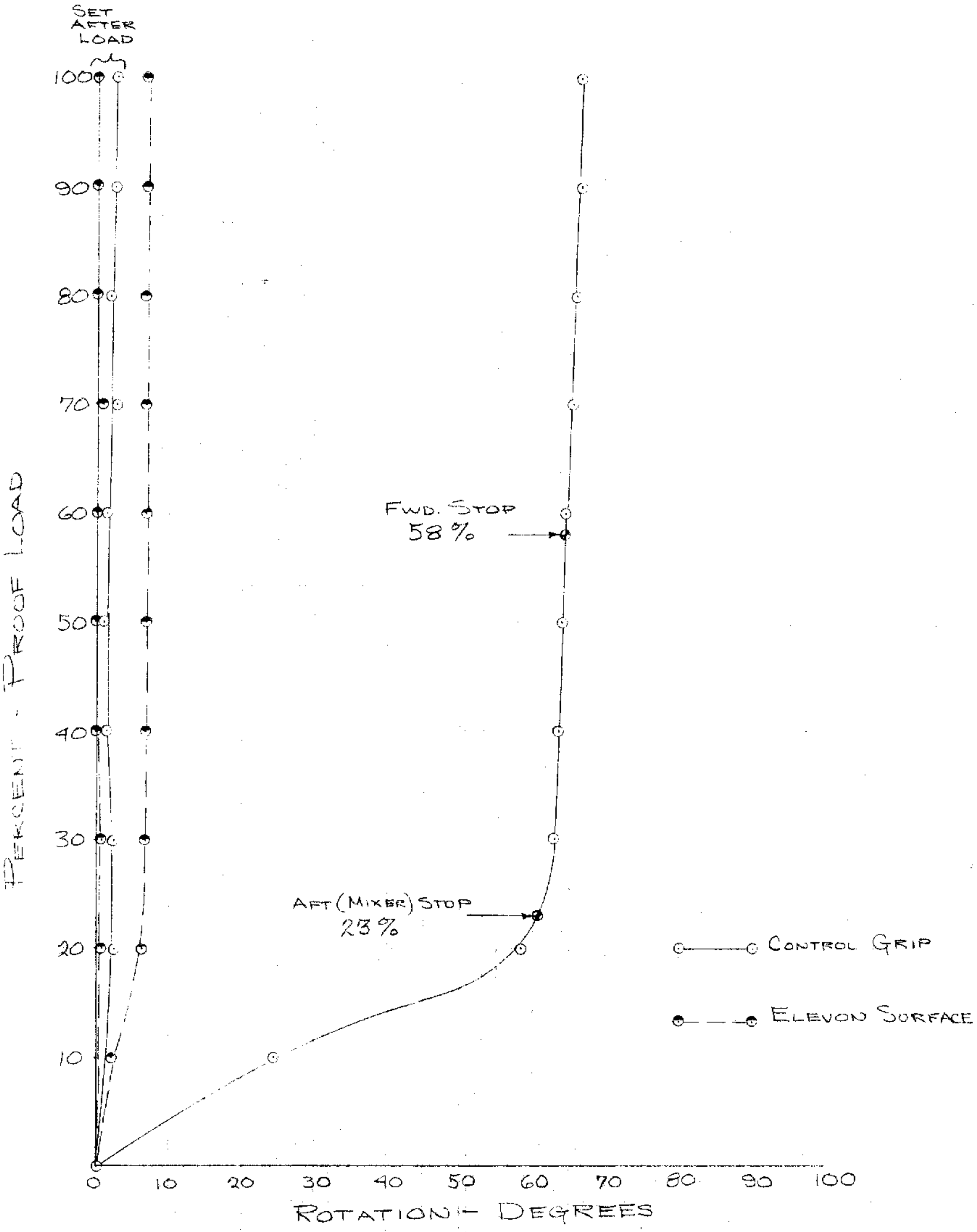
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO, CALIFORNIA

PAGE
DOC. NO.
MODEL F-102A

FORM 150-A

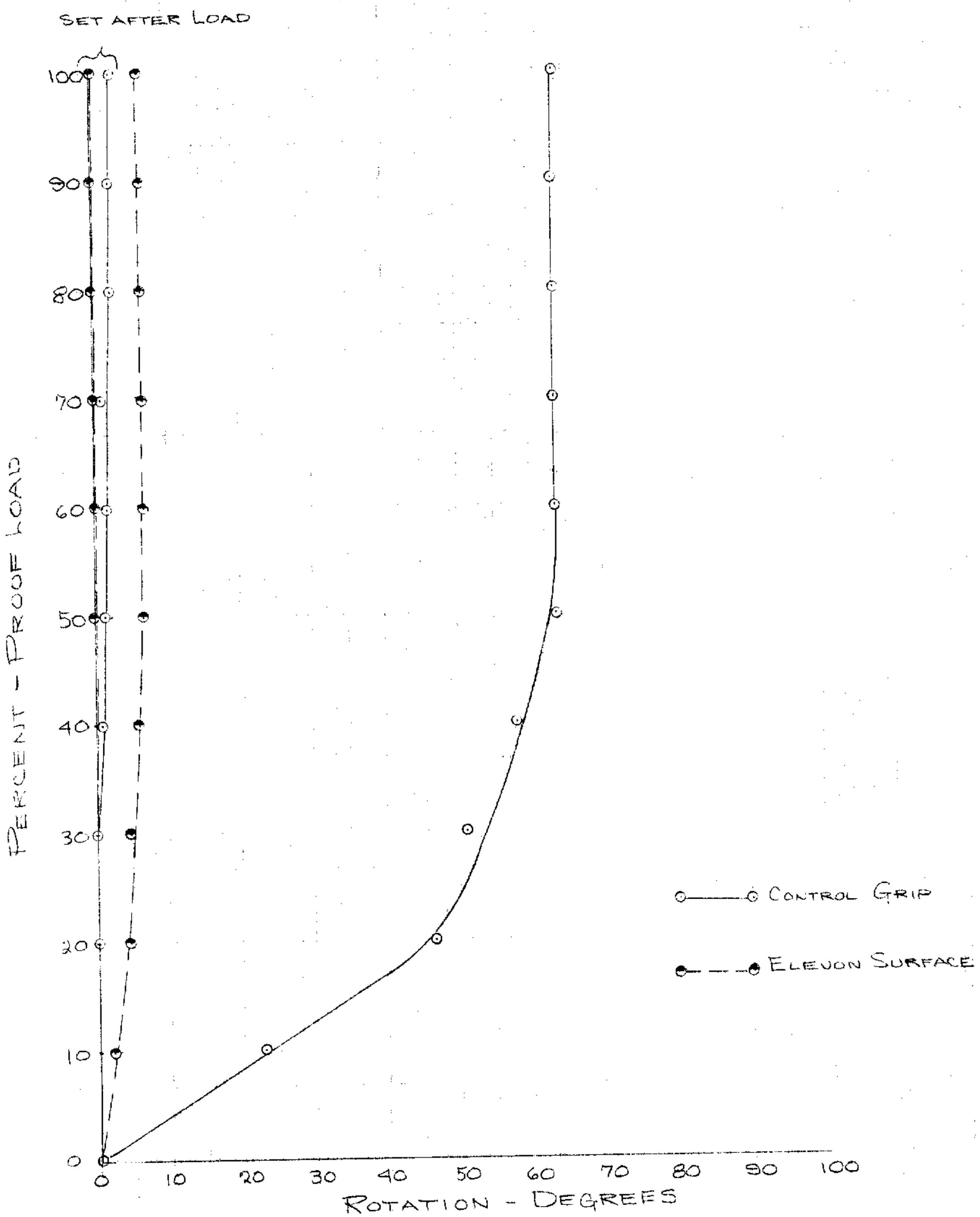
FIGURE 18



CALCULATED BY	J.M.	11-20-56	Condition A-5 Load-Deflection Curves Aileron System	PAGE
TRACED BY	J.M.	11-20-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	
APPROVED BY				MODEL F-102A
APPROVED BY				

FORM 150-A

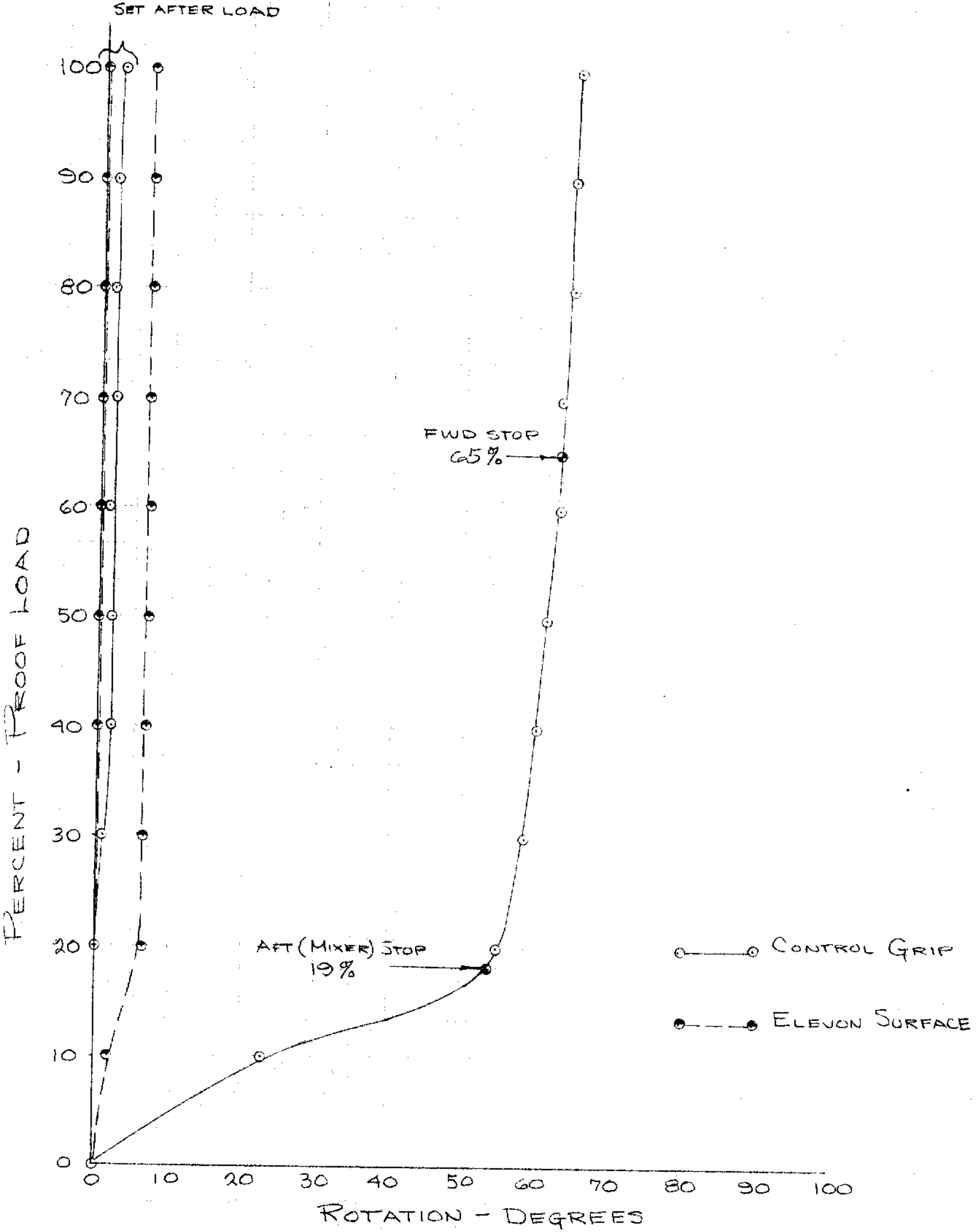
Figure 19



CALCULATED BY	J M	11-20-56	Condition A-5A Load-Deflection Curves Aileron System	PAGE
TRACED BY	J M	11-20-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	MODEL F-102A
APPROVED BY				
APPROVED BY				

FORM 150-A

Figure 20



CALCULATED BY	J.M.	11-20-56
TRACED BY	J.M.	11-20-56
CHECKED BY		
APPROVED BY		
APPROVED BY		

CONDITION A-6
LOAD-DEFLECTION CURVES
AILERON SYSTEM

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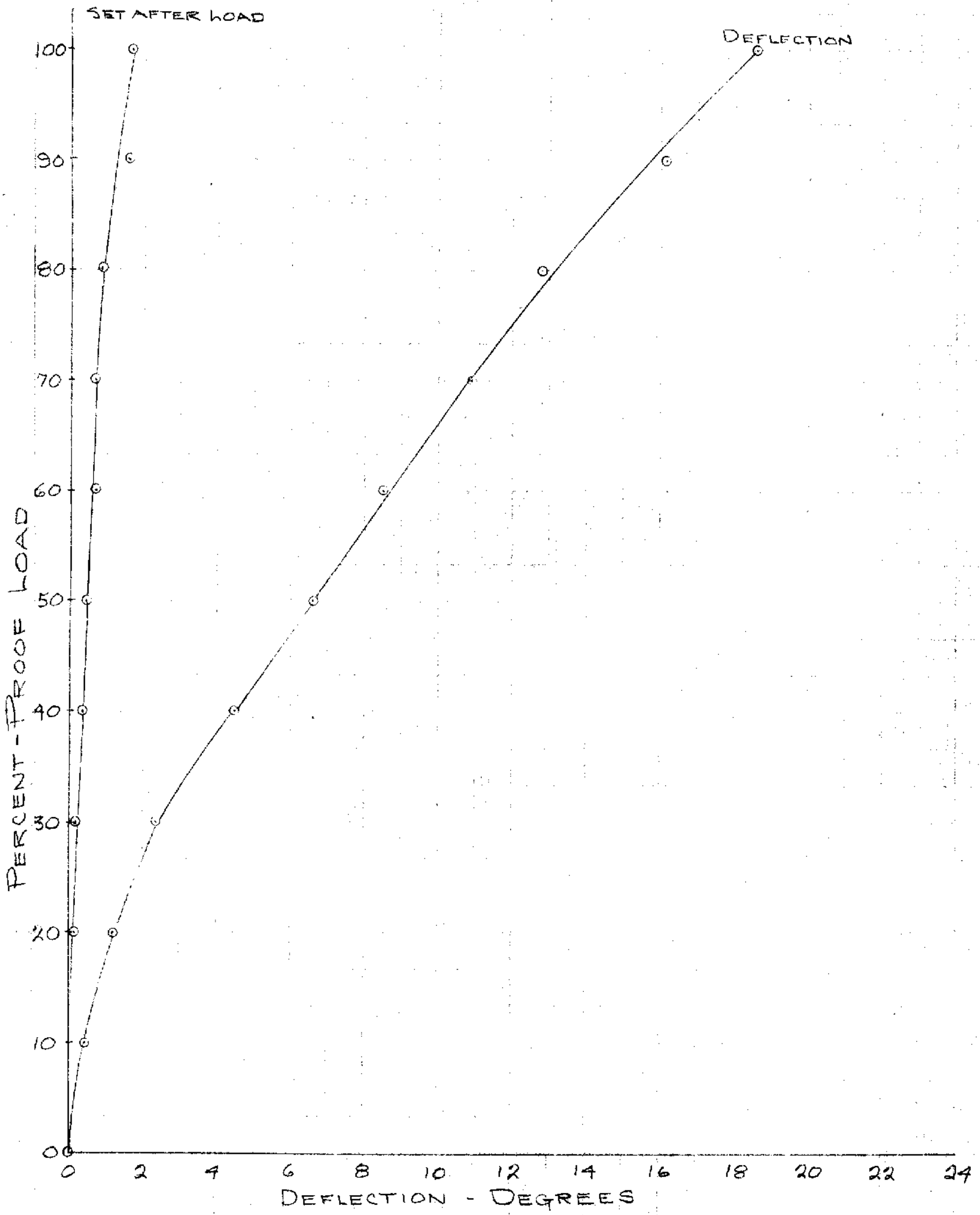
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO, CALIFORNIA

PAGE
DOC. NO.
MODEL
F-102A

FORM 150-A

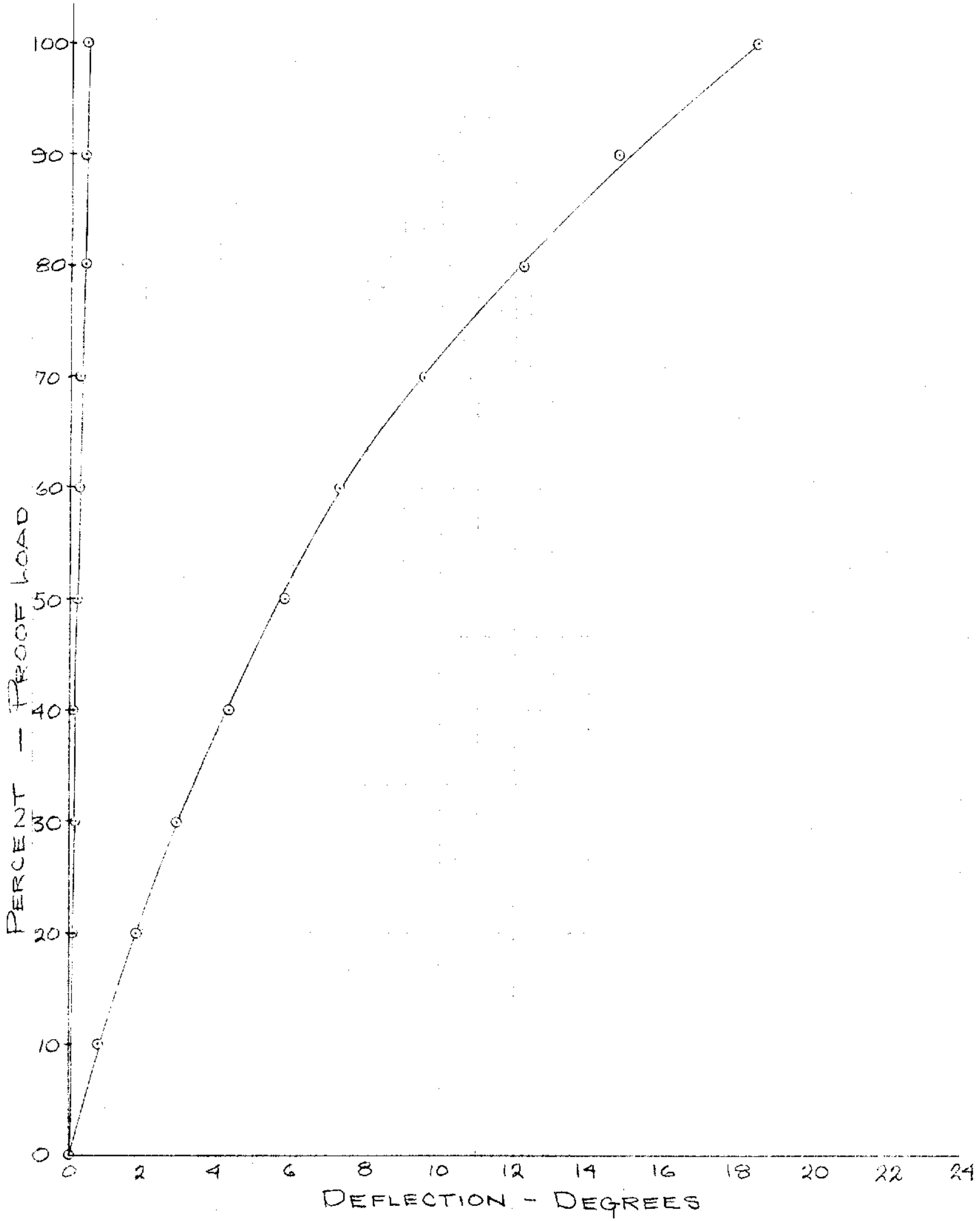
FIGURE 21



CALCULATED BY	J. M.	11-20-56	CONDITION B-1 LOAD - DEFLECTION CURVES ELEVATOR SYSTEM	PAGE
TRACED BY	J. M.	11-20-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	MODEL F-102 A
APPROVED BY				
APPROVED BY				

FORM 150-A

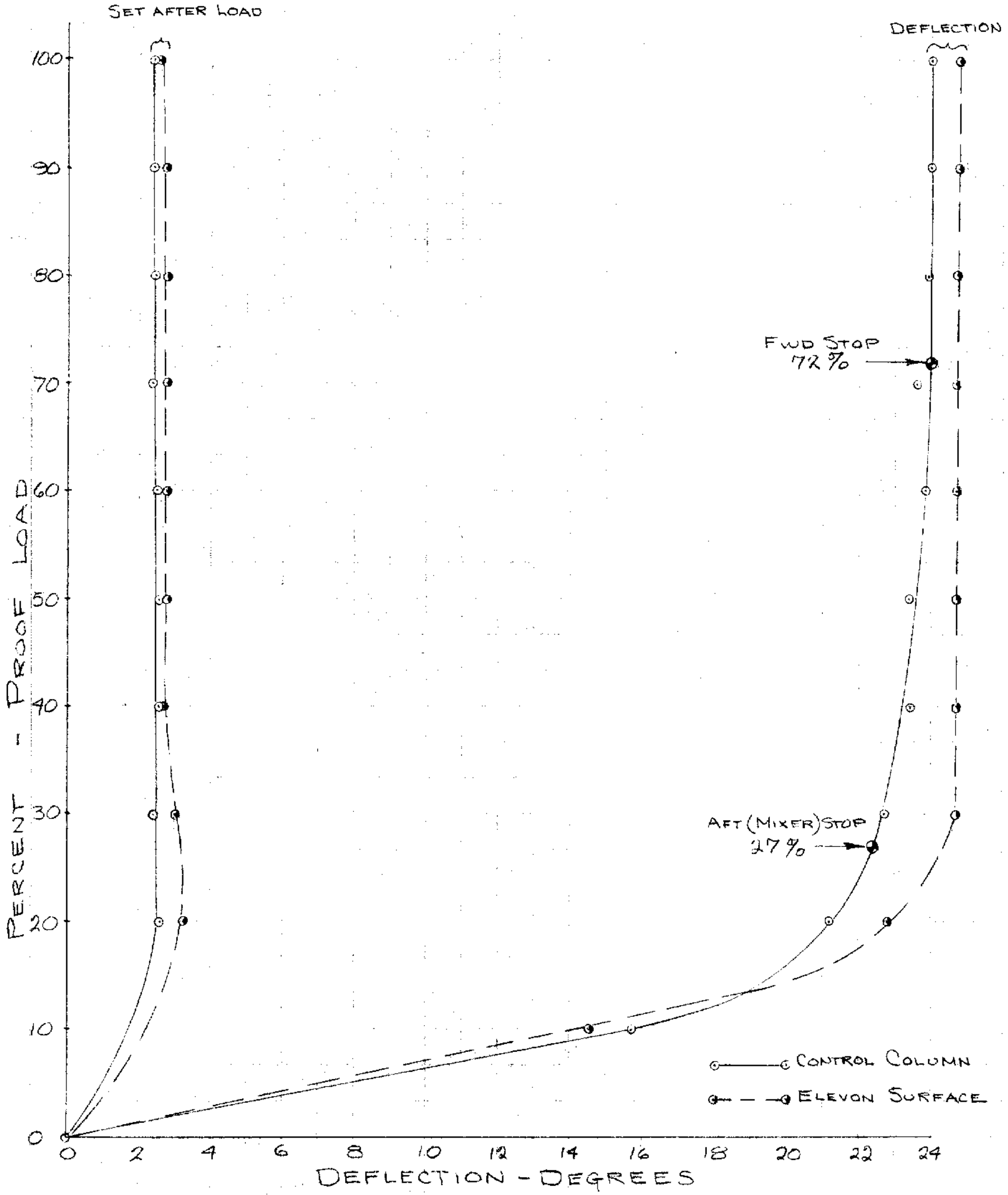
FIGURE 22



CALCULATED BY	J.M.	11-20-56	Condition B-2 Load-Deflection Curves Elevator System	PAGE
TRACED BY	J.M.	11-20-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	MODEL F-102A
APPROVED BY				
APPROVED BY				

FORM 150-A

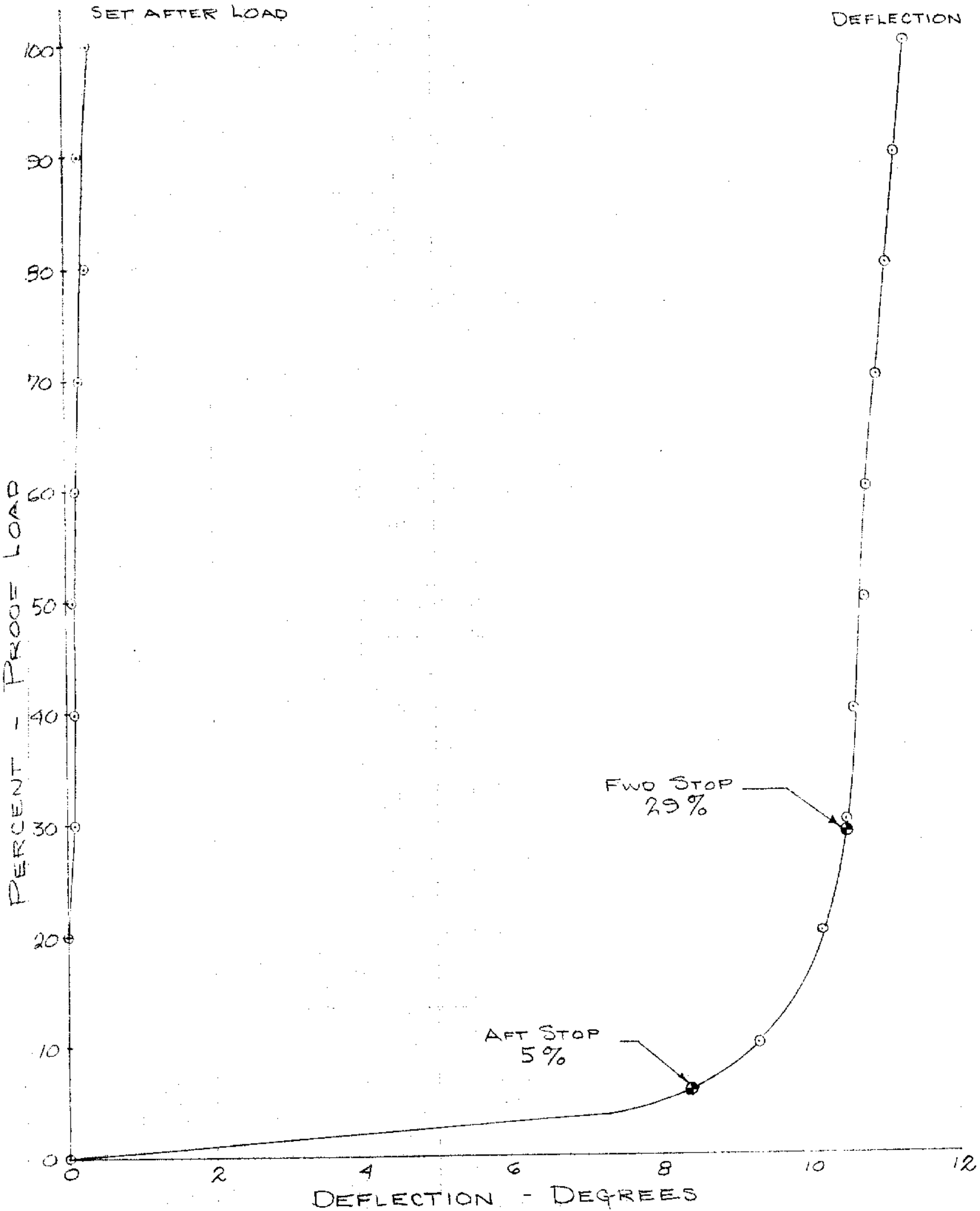
Figure 23



CALCULATED BY	J.M.	11-20-56	CONDITION B-3 LOAD-DEFLECTION CURVES ELEVATOR SYSTEM	PAGE
TRACED BY	J.M.	11-20-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	
APPROVED BY				MODEL F-102A
APPROVED BY				

FORM 150-A

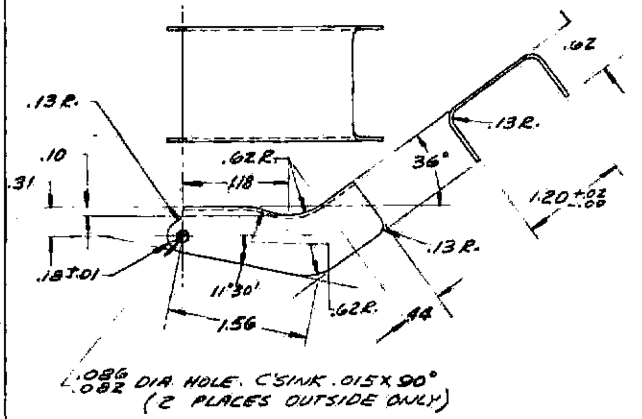
FIGURE 24



CALCULATED BY	J M	11-20-56	CONDITION B-4 LOAD-DEFLECTION CURVES ELEVATOR SYSTEM	PAGE
TRACED BY	J M	11-20-56		DOC. NO.
CHECKED BY			**CONVAIR** A DIVISION OF GENERAL DYNAMICS CORPORATION SAN DIEGO, CALIFORNIA	MODEL
APPROVED BY				F-102A
APPROVED BY				

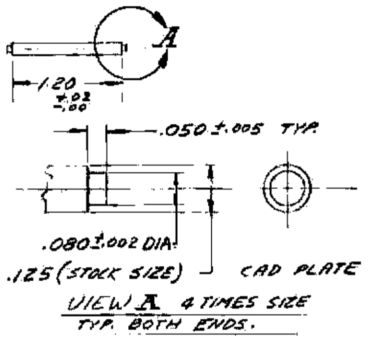
FORM 150-A

FIGURE 25



PAINT RED - SEE NOTE #14

TVA 47623-35



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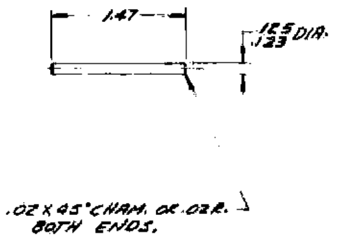

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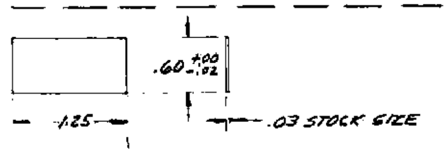

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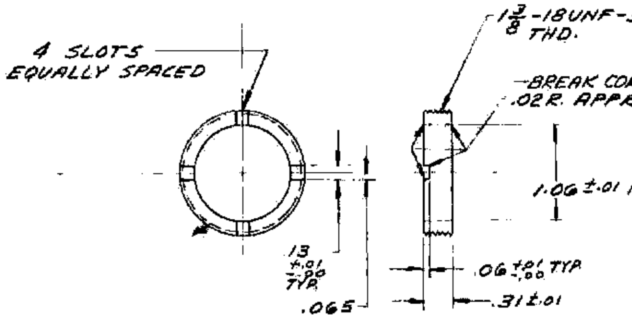
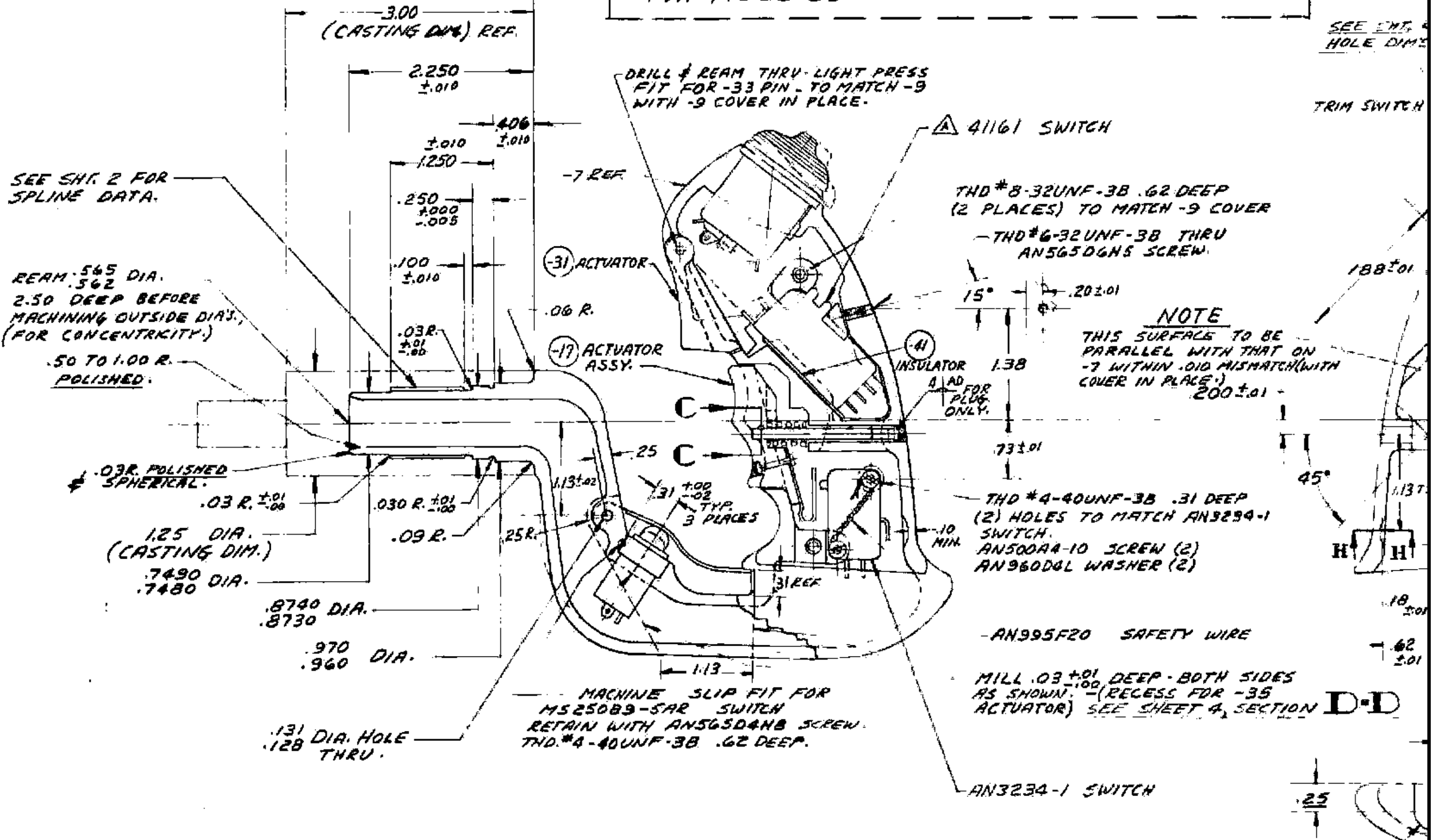
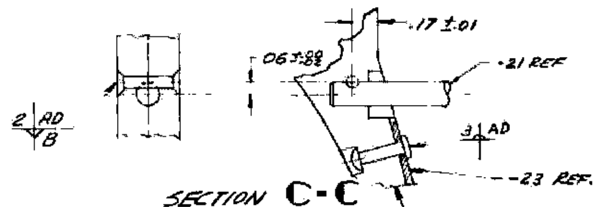
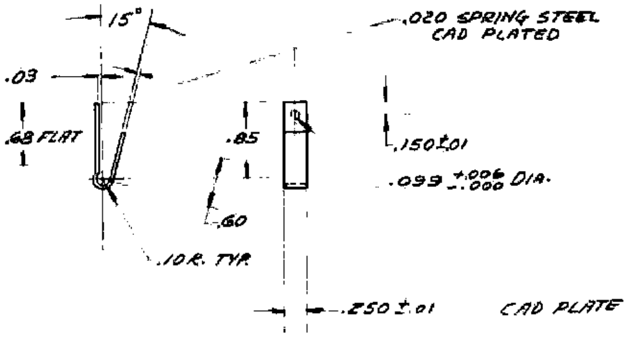


TVA 47623-39


3.00 (CASTING DIM.) REF.
2.250 ±.010
.406 ±.010
±.010 1.250
SEE SHT. 2 FOR SPLINE DATA.
-7 REF.
.250 +.000 -.005
REAM .565/.562 DIA. 2.50 DEEP BEFORE MACHINING OUTSIDE DIA'S. (FOR CONCENTRICITY.)
.100 ±.010
.06 R.
.03 R. +.01 -.00
.50 TO 1.00 R. POLISHED.
.03 R. POLISHED SPHERICAL.
.03 R. +.01 -.00
1.25 DIA. (CASTING DIM.)
.7490/.7480 DIA.
.030 R. +.01 -.00
.09 R.
.8740/.8730 DIA.
.970/.960 DIA.
.131/.128 DIA. HOLE THRU.
DRILL & REAM THRU - LIGHT PRESS FIT FOR -33 PIN - TO MATCH -9 WITH -9 COVER IN PLACE.
A 41161 SWITCH
(31) ACTUATOR
(17) ACTUATOR ASSY.
(41) INSULATOR
C
C
.25
1.13 ±.02
.25 R.
.31 +.00 -.02 TYP. 3 PLACES
.31 REF.
1.13
MACHINE SLIP FIT FOR MS25089-5AR SWITCH RETAIN WITH AN565D4H8 SCREW. THD. #4-40UNF-3B .62 DEEP.
THD #8-32UNF-3B .62 DEEP (2 PLACES) TO MATCH -9 COVER
THD #6-32UNF-3B THRU AN565D6H5 SCREW.
15°
.20 ±.01
NOTE
THIS SURFACE TO BE PARALLEL WITH THAT ON -7 WITHIN .010 MISMATCH (WITH COVER IN PLACE.)
1.38
4 AD FOR PLUG ONLY.
.200 ±.01
.73 ±.01
THD #4-40UNF-3B .31 DEEP (2) HOLES TO MATCH AN3234-1 SWITCH.
AN500A4-10 SCREW (2)
AN960D4L WASHER (2)
.10 MIN.
AN995F20 SAFETY WIRE
MILL .03 +.01 -.00 DEEP - BOTH SIDES AS SHOWN. (RECESS FOR -35 ACTUATOR) SEE SHEET 4, SECTION D-D
AN3234-1 SWITCH
SEE SHT. 4 HOLE DIMS
TRIM SWITCH
.188 ±.01
45°
1.13
H
H
.18 ±.01
.62 ±.01
.25
.25 R.



SECTION C-C
2 TIMES SIZE



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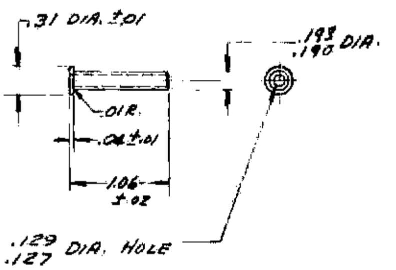
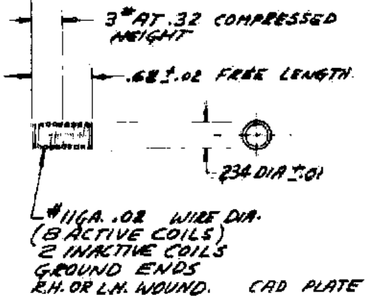
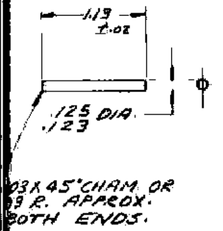


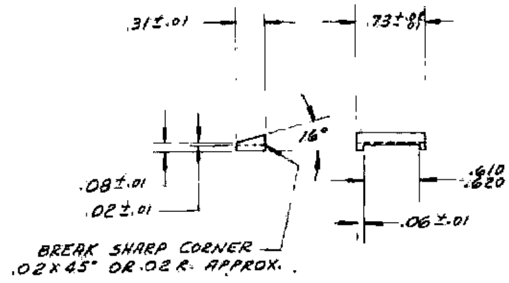
TVA 47623-27



TVA 47623-25



A 47623-33



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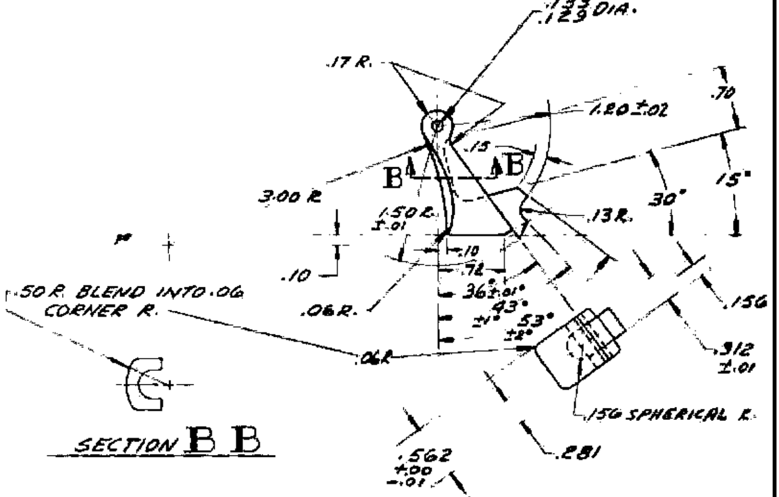
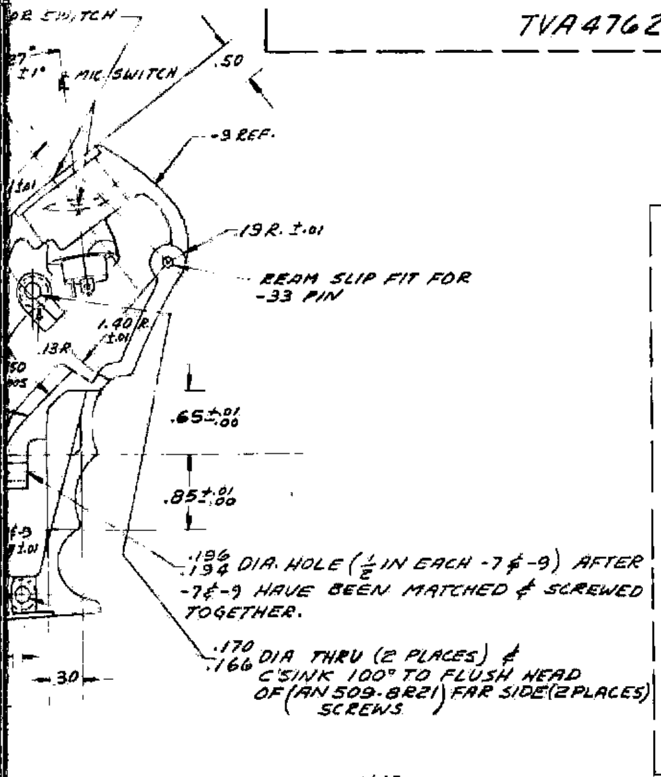
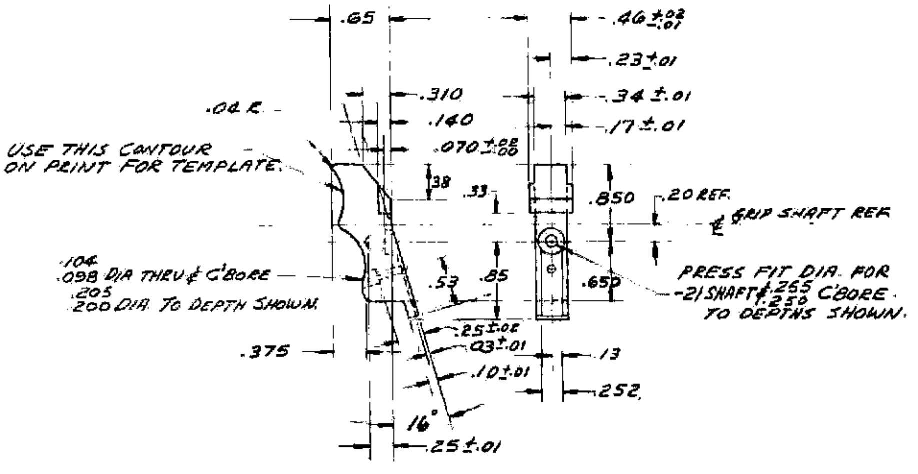


SECTION B B

PAINT RED - SEE NOTE #14

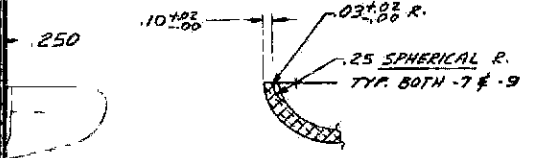
TVA 47623-31


OR SWITCH
27° ±1°
MIC SWITCH
.50
-9 REF.
±.01
.19 R. ±.01
REAM SLIP FIT FOR -33 PIN
1.40 R. ±.01
.13 R.
.65 +.01 -.00
.85 +.01 -.00
.196 .194 DIA. HOLE (1/2 IN EACH -7 & -9) AFTER -7 & -9 HAVE BEEN MATCHED & SCREWED TOGETHER.
.170 .166 DIA THRU (2 PLACES) & C'SINK 100° TO FLUSH HEAD OF (AN 509-8R21) FAR SIDE (2 PLACES) SCREWS.
.30


.65
.46 +.02 -.01
.23±.01
.310
.34±.01
.04 R.
.140
.17±.01
.070 +.02 -.00
USE THIS CONTOUR ON PRINT FOR TEMPLATE.
.38
.33
.850
.20 REF.
℄ GRIP SHAFT REF.
.104 .098 DIA THRU & C'BORE
.205 .200 DIA. TO DEPTH SHOWN.
.53
.85
.650
PRESS FIT DIA. FOR -21 SHAFT & .265 .250 C'BORE TO DEPTHS SHOWN.
.25±.02
.375
.03±.01
.13
.10±.01
.252
16°
.25±.01

PAINT RED - SEE NOTE #14

TVA 47623-19



SECTION H-H

EWO 3605 ACA 8-1111-01

GOVT		
CHECKER		
CR ENGR	Mac Donald	4/[illegible]
MATERIAL	[illegible]	4/[illegible]
TOOLING		
STR ENGR	[illegible]	[illegible]
WEIGHTS		
DESIGN		
DFTSMN	J. OLSON	3-21-50

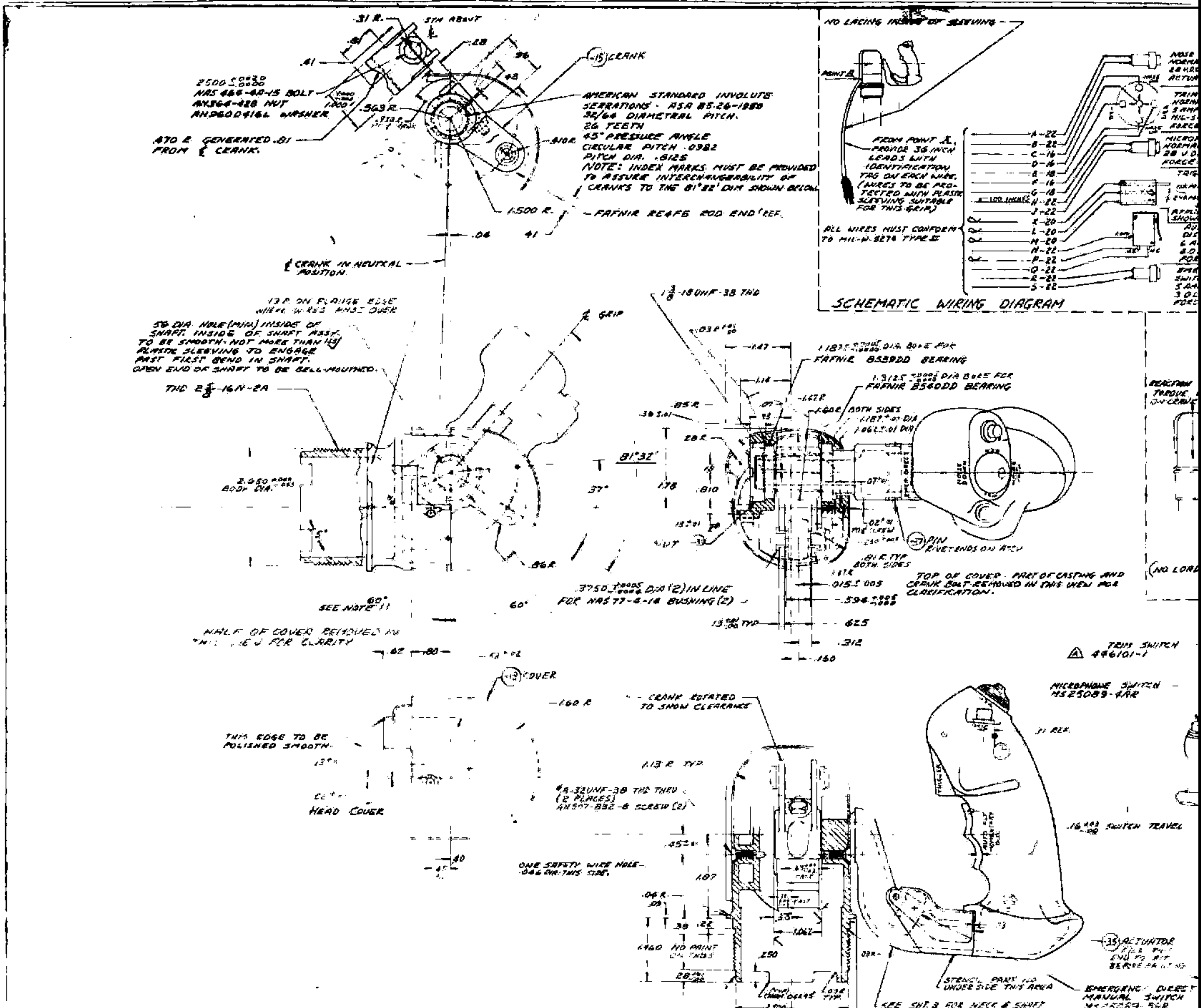
GRIP ASSY.- SIDE CONTROL STICK.

SCALE FULL & NOTED REL

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO CALIFORNIA

DWG SIZE D TVA 47623 ON 8-49520
SHEET 3

T 4-10-56 FIGURE 3


.31 R.
5TH ABOUT ℄
.81
.28
.41
.96
.48
15 CRANK
2500 +.0020 -.0000
NAS 464-4A-15 BOLT
AN364-428 NUT
AN960D416L WASHER
.563 R.
AMERICAN STANDARD INVOLUTE SERRATIONS - ASA B5.26-1950
32/64 DIAMETRAL PITCH.
26 TEETH
45° PRESSURE ANGLE
CIRCULAR PITCH .0982
PITCH DIA. .8125
NOTE: INDEX MARKS MUST BE PROVIDED TO ASSURE INTERCHANGEABILITY OF CRANKS TO THE 81°32' DIM SHOWN BELOW.
.470 R. GENERATED .81 FROM ℄ CRANK.
.410R.
1.500 R.
FAFNIR RE4FB ROD END (REF.
.04
41
℄ CRANK IN NEUTRAL POSITION.
NO LACING INSIDE OF SLEEVING
POINT B
FROM POINT A, PROVIDE 36 INCH LEADS WITH IDENTIFICATION TAG ON EACH WIRE. (WIRES TO BE PROTECTED WITH PLASTIC SLEEVING SUITABLE FOR THIS GRIP.)
ALL WIRES MUST CONFORM TO MIL-W-5274 TYPE II
A-22
B-22
C-16
D-16
E-18
F-16
G-18
H-22
J-22
K-20
L-20
M-20
N-22
P-22
Q-22
R-22
S-22
SCHEMATIC WIRING DIAGRAM
.13 R. ON FLANGE EDGE WHERE WIRES PASS OVER
.59 DIA. HOLE (MIN) INSIDE OF SHAFT. INSIDE OF SHAFT ASSY. TO BE SMOOTH - NOT MORE THAN 1/32 PLASTIC SLEEVING TO ENGAGE PAST FIRST BEND IN SHAFT. OPEN END OF SHAFT TO BE BELL-MOUTHED.
THD 2 5/8-16N-2A
2.450 BODY DIA.
℄ GRIP
1 3/8-18UNF-3B THD
.147
1.1875 DIA. BORE FOR FAFNIR B539DD BEARING
1.3125 DIA. BORE FOR FAFNIR B540DD BEARING
.14
.447 R
.60R BOTH SIDES
.85 R.
.36
.28 R.
81°32'
37°
.178
.810
NUT
PIN RIVET ENDS ON ASSY
.86R.
.3750 DIA (2) IN LINE FOR NAS 77-6-14 BUSHING (2)
.015±.005
.594
.625
.312
.160
TOP OF COVER - PART OF CASTING AND CRANK BOLT REMOVED IN THIS VIEW FOR CLARIFICATION.
60° SEE NOTE 11
60°
HALF OF COVER REMOVED IN THIS VIEW FOR CLARITY
.62
.80
13 COVER
.160 R
THIS EDGE TO BE POLISHED SMOOTH.
HEAD COVER
CRANK ROTATED TO SHOW CLEARANCE
.113 R. TYP.
#8-32UNF-3B THD THRU (2 PLACES) AN507-832-8 SCREW (2)
ONE SAFETY WIRE HOLE - .046 DIA. THIS SIDE.
.45
.187
.06 R.
.03
.38
.22
.460 NO PAINT ON THDS
.250
.062
.28
2.300 DIA.
2.940 DIA.
.40
SEE SHT. 3 FOR NECK & SHAFT PORTION OF GRIP
STENCIL PART NO. UNDER SIDE THIS AREA
TRIM SWITCH 496101-1
MICROPHONE SWITCH MS25089-4AR
.16 SWITCH TRAVEL
35 ACTUATOR
EMERGENCY DIRECT MANUAL SWITCH MS25089-5AR
(NO LOAD

5 EXTERIOR SURFACE OF ENTIRE GRIP ASSY. (EXCEPT SWITCH BUTTONS) SHALL BE PAINTED WITH #3725 BLACK LACQUER-SPEC. TT-C-595 ALL SWITCHES (EXCEPT TRIM SWITCH WHICH ALSO SHALL BE BLACK) TO BE PAINTED WITH #3115 RED LACQUER-SPEC. TT-C-595, UNLESS THEY ARE MADE OF RED PLASTIC SUITABLE FOR THIS GRIP.

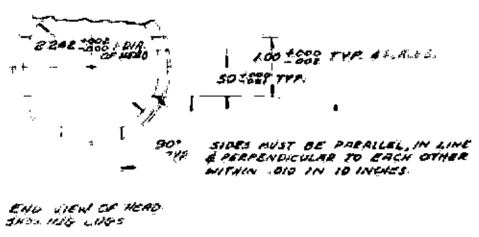
6 RECESSED HOLES FOR PINS, SCREWS ETC. TO BE FILLED WITH A SUITABLE SEALING COMPOUND AFTER ASSY.

7 ALL LETTERING TO BE APPROX .16 HIGH, RECESSED AND "WHITE PAINT" FILLED EXCEPT CONVAIR PART NO. WHICH SHALL BE STENCILED IN WHITE LETTERS APPROX. .19 HIGH ON BOTTOM OF GRIP.

THE HEAD COVER SHALL BE APPROX .03 THICK MADE OF [redacted] LAMINATED PLASTICS. SEE B/M

CRANK MUST TRAVEL 60° IN BOTH DIRECTIONS WITH A MAX. FRICTION OF 1/4 IN LB TORQUE. OVERTRAVEL MUST BE PROVIDED FOR AS FOLLOWS: 5° OVERTRAVEL IN THE DOWN POSITION 5° OVERTRAVEL IN THE UP POSITION (WITH COVER ON) AND 37° WITH COVER OFF (THIS IS TO ALLOW FOR ROD END ADJUSTMENTS.)

1 WIRES TO HAVE SUFFICIENT SLACK WITHIN GRIP SO THAT SWITCHES CAN BE REPLACED IF NECESSARY.


2.242 DIA. OF HEAD
1.00 TYP. 4 PLACES
.50 TYP.
90° TYP.
SIDES MUST BE PARALLEL, IN LINE & PERPENDICULAR TO EACH OTHER WITHIN .010 IN 10 INCHES.

END VIEW OF HEAD SHOWING LUGS

[illegible]

[illegible]

[illegible]

[illegible] (THOSE DO NOT [illegible]) [illegible] 29 [illegible] 8000D.

[illegible] IN CONTACT WITH [illegible]

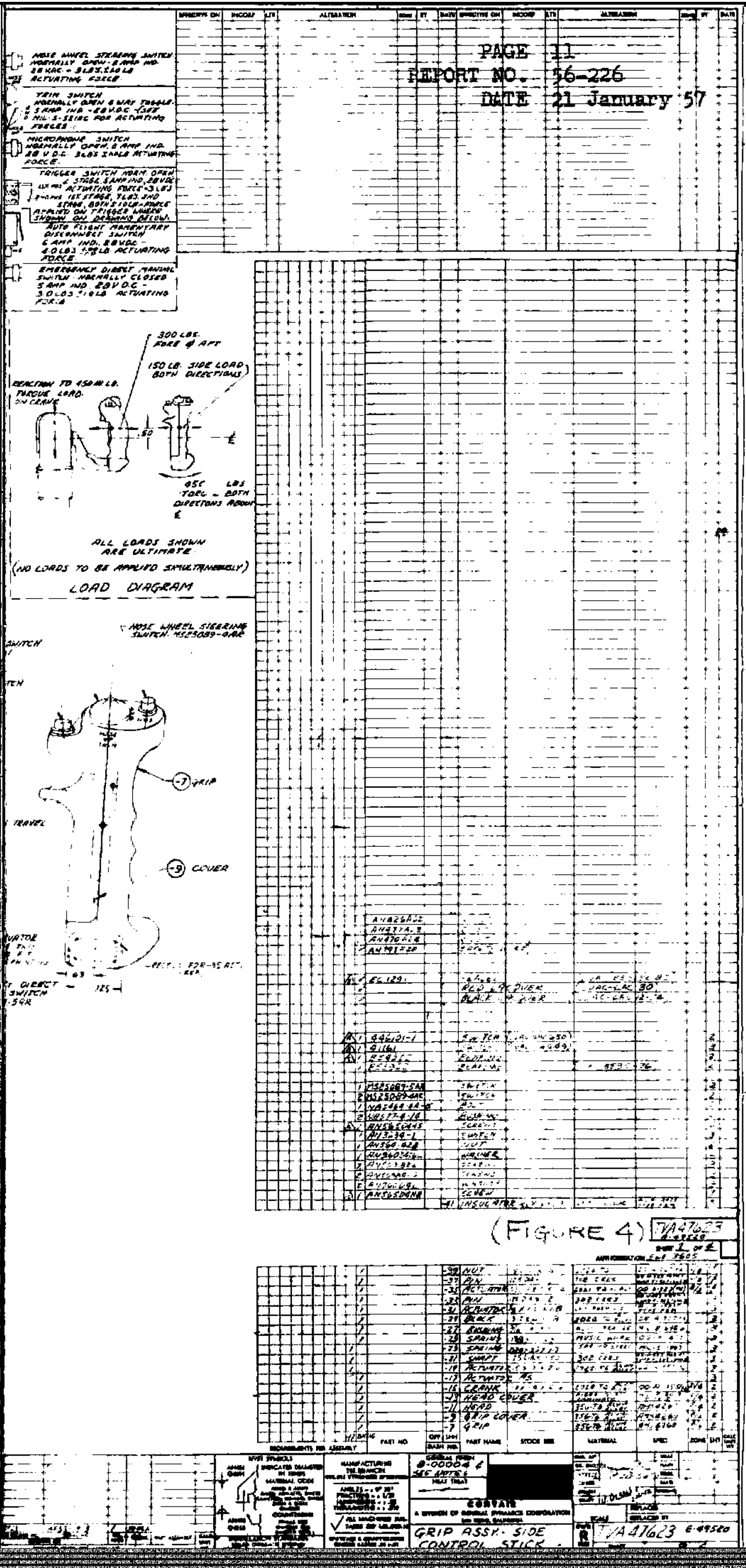
4 THE GRIP SHAPE AND SIZE SHALL BE GOVERNED BY A CONVAIR FURNISHED MODEL.

- AFTER TESTING RUN DI-ELECTRIC TEST BY APPLYING 600 V. RMS @ 60~ ACROSS EACH LEAD AND THREADED PORTION OF GRIP ASSY FOR 1 MINUTE.

2 △ MASON ELECTRIC CORP LOS ANGELES 62, CALIF

1 [illegible] -7, -9 AND -[illegible] TO MEET [illegible] PER MIL-E-5272

NOTES:


NOSE WHEEL STEERING SWITCH NORMALLY OPEN - 8 AMP IND. 28 VDC - 3 LBS ±1/2 LB ACTUATING FORCE
TRIM SWITCH NORMALLY OPEN 4 WAY TOGGLE 5 AMP IND - 28 V.D.C. MIL-S-5818C FOR ACTUATING FORCES
MICROPHONE SWITCH NORMALLY OPEN. 8 AMP IND. 28 V D.C. 3 LBS ±1/2 LB ACTUATING FORCE.
TRIGGER SWITCH NORM. OPEN 2 STAGE 5 AMP IND, 28 VDC ACTUATING FORCE - 3 LBS 1ST STAGE, 7 LBS 2ND STAGE, BOTH ±1 LB - FORCE APPLIED ON TRIGGER WHERE SHOWN ON DRAWING BELOW.
AUTO FLIGHT MOMENTARY DISCONNECT SWITCH 6 AMP IND. 28 VDC - 4.0 LBS ±1/2 LB ACTUATING FORCE
EMERGENCY DIRECT MANUAL SWITCH NORMALLY CLOSED 5 AMP IND 28 V D.C - 3.0 LBS ±1 LB ACTUATING FORCE
300 LBS. FORE & AFT
150 LB. SIDE LOAD BOTH DIRECTIONS
REACTION TO 450 IN LB. TORQUE LOAD ON CRANK
.50
℄
450 IN LBS TORQ. - BOTH DIRECTIONS ABOUT ℄
ALL LOADS SHOWN ARE ULTIMATE
(NO LOADS TO BE APPLIED SIMULTANEOUSLY)
LOAD DIAGRAM
NOSE WHEEL STEERING SWITCH MS25089-4AR
SWITCH
TCH
-7 GRIP
TRAVEL
-9 COVER
UATOR
DIRECT SWITCH
.63
.125
(FIGURE 4)
VA47623
E-49520
SHEET 1 OF 2
EFFECTIVITY ON
INCORP
ALTERATION
PART NO.
PART NAME
STOCK SIZE
MATERIAL
SPEC
ZONE
SHT
REQUIRED PER ASSEMBLY
GENERAL FINISH 8-00004 SEE NOTE 6
HEAT TREAT
CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO, CALIFORNIA
SCALE
REPLACES
REPLACED BY
GRIP ASSY. - SIDE CONTROL STICK
VA47623 E-49520
R
J. OLSON

Qty	Part No.	Part Name
	AN426AD[illegible]	
	AN937A[illegible]	
	AN476[illegible]	
	AN491[illegible]	
1	EC129[illegible]	[illegible]
		RED [illegible]
		BLACK [illegible]
1	446101-1	SWITCH
1	91161	SWITCH
1	[illegible]	BEARING
1	[illegible]	BEARING
1	MS25089-5AR	SWITCH
2	MS25089-4AR	SWITCH
1	NAS464-4A-[illegible]	BOLT
2	NAS77-4-14	BUSHING
	AN565[illegible]	SCREW
1	AN13239-L	SWITCH
1	AN364-428	NUT
1	AN960[illegible]	WASHER
2	[illegible]	[illegible]
2	[illegible]	[illegible]
2	[illegible]	WASHER
1	AN565DC[illegible]	SCREW
	-41	INSULATOR

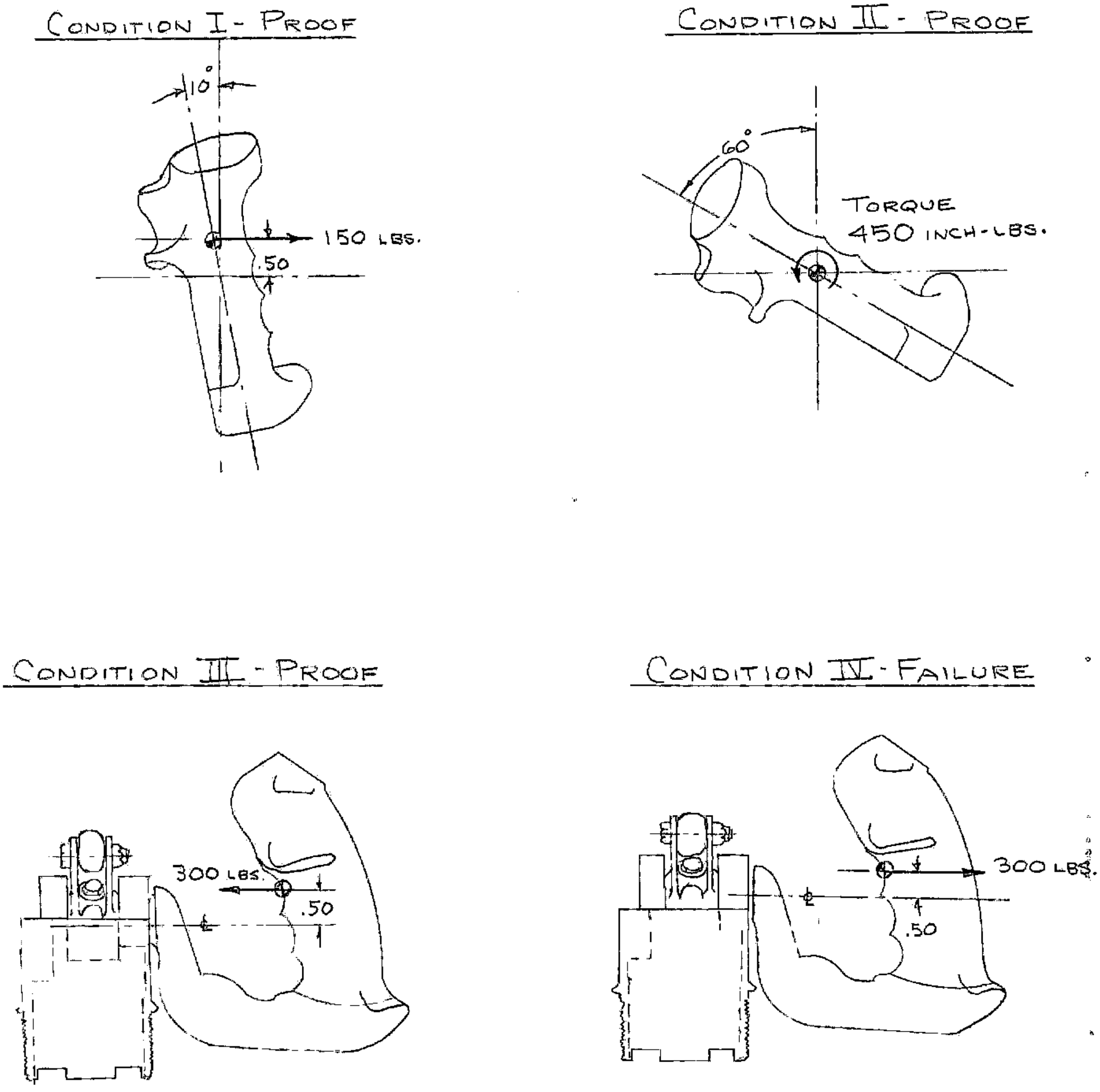
Dash No.	Part Name
-39	NUT
-37	PIN
-35	ACTUATOR
-33	PIN
-31	ACTUATOR
-29	BLOCK
-27	BUSHING
-25	SPRING
-23	SPRING
-21	SHAFT
-19	ACTUATOR
-17	ACTUATOR
-15	CRANK
-13	HEAD COVER
-11	HEAD
-9	GRIP COVER
-7	GRIP

[redacted]

DESIGN ULTIMATE LOADS

GRIP ASSY. - SIDE CONTROL STICK

TVA 47633 ON 8-49520


CONDITION I - PROOF

CONDITION II - PROOF

CONDITION III - PROOF

CONDITION IV - FAILURE

FIGURE 5

AFFTC-TR-57-25
NOVEMBER 1957
ASTIA DOCUMENT
NO. AD-131466

[redacted]

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ARDC

F-102A
CASE XX WING
PHASE IV PERFORMANCE

LEROY G. COOPER
Captain, USAF
Project Engineer

ROBERT M. WHITE
Captain, USAF
Project Pilot

CONVAIR - SECURITY LOG
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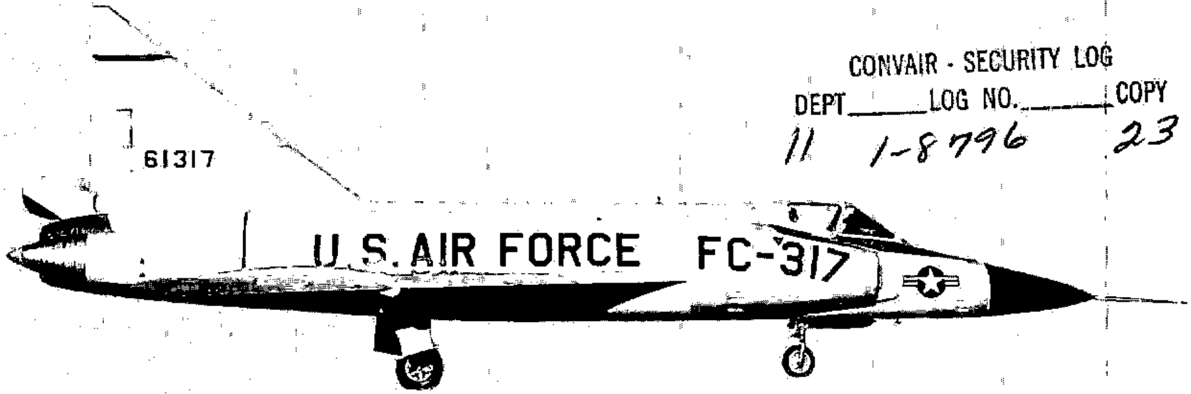

AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE

58 SARS-0084C

[redacted]

Unclassified WC0S3858

57-F-1589
#23

MEMO

CONVAIR—SAN DIEGO

F. D. APPLEGATE

DATE 2/16 6-105

TO D H Bennett Frick

Pls fix SOC charts OK

Jim

Pls return report & letter

Jim

~~CONFIDENTIAL~~

HEADQUARTERS
WRIGHT AIR DEVELOPMENT CENTER
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE
Wright-Patterson Air Force Base, Ohio

SH Bennett - action
Applegst

IN REPLY ADDRESS
COMMANDER, ATTENTION:
WCLSD-4

30 December 1957

SUBJECT: F-102A, Preparation of Standard Aircraft Characteristics For

THRU:
Air Force Plant Representative
Convair Division
General Dynamics Corporation
San Diego 12, California

TO:
Convair Division
General Dynamics Corporation
San Diego 12, California

Forwarded is one (1) copy of Air Force Flight Test Center Technical Report AFFTC-TR-57-25 entitled "F-102A Case XX Phase IV Performance" dated November 1957 to be used as a basis for a revision to the F-102A Standard Aircraft Characteristics.

FOR THE COMMANDER:

Joseph A. Ellis
JOSEPH A. ELLIS
Chief, Design Branch
Aircraft Laboratory

1 Incl
AFFTC-TR-57-25, Phase IV
Nov57 (1cy) (C)

Cy furnished:
SAAMA
AFPRO

CONVAIR - SECURITY LOG
DEPT ____ LOG NO. ____ COPY
11 1-8796 1

57WCOS-3858

IND.
AFPR Office, SAAMA, Convair - A Div. of Gen. Dynamics
San Diego 12, Calif. 24 Jan 58
To: Convair - A Div. of Gen. Dynamics, San Diego
Attn: Contr. Adm. Model 8
~~Parts Sales~~
~~GLP Group~~ Project Office
Forwarded
for GEORGE E. BOSTWICK, Colonel, USAF
AF Plant Representative

When Inclosure(s) One is (are) withdrawn or not attached, the classification of confidential on this WADC Ltr will be canceled (or changed to unclassified) in accordance with Par AFR 205-1

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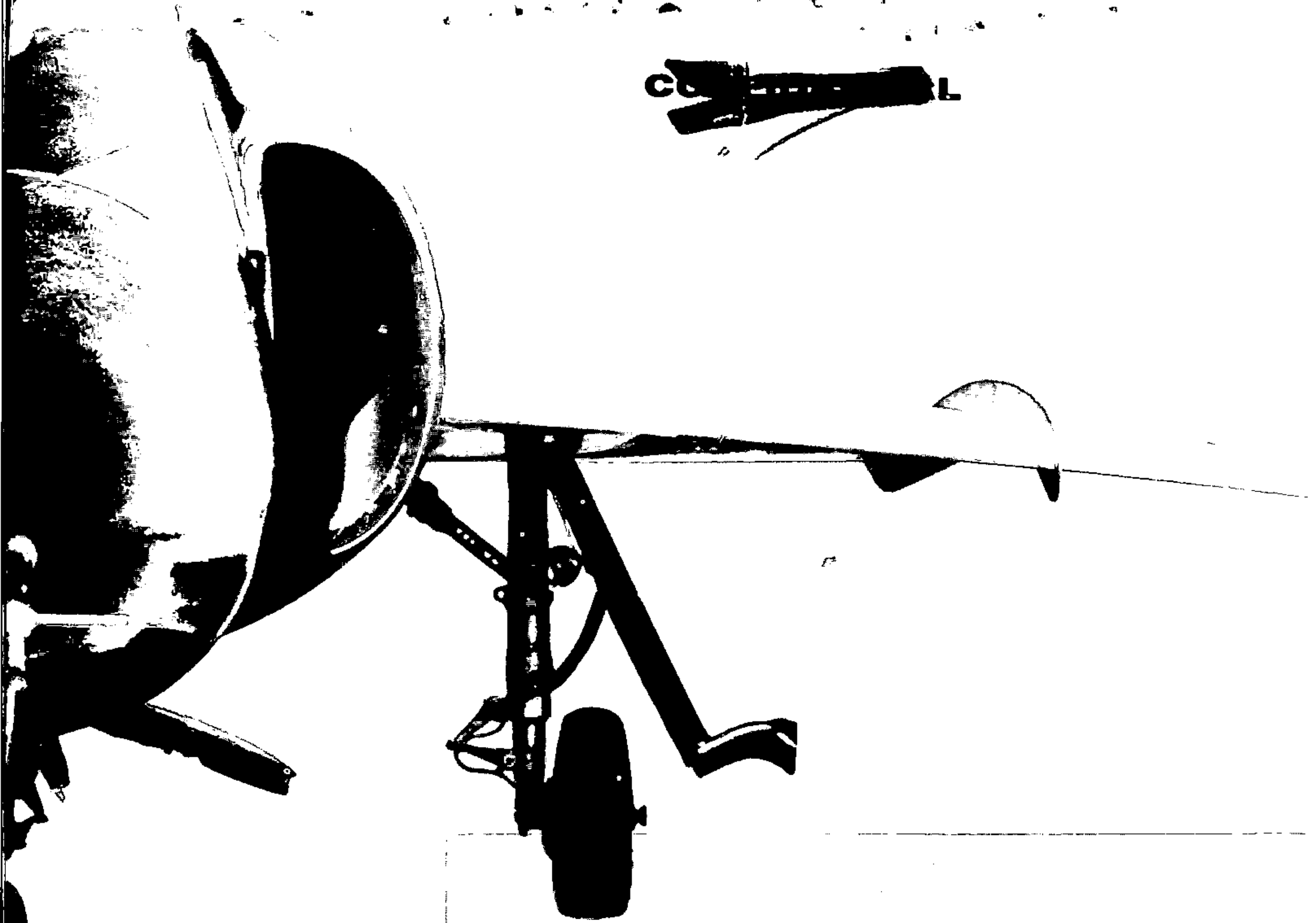
[redacted]



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AFFTC-TR-57-25
November 1957
ASTIA Document
No. AD-131466

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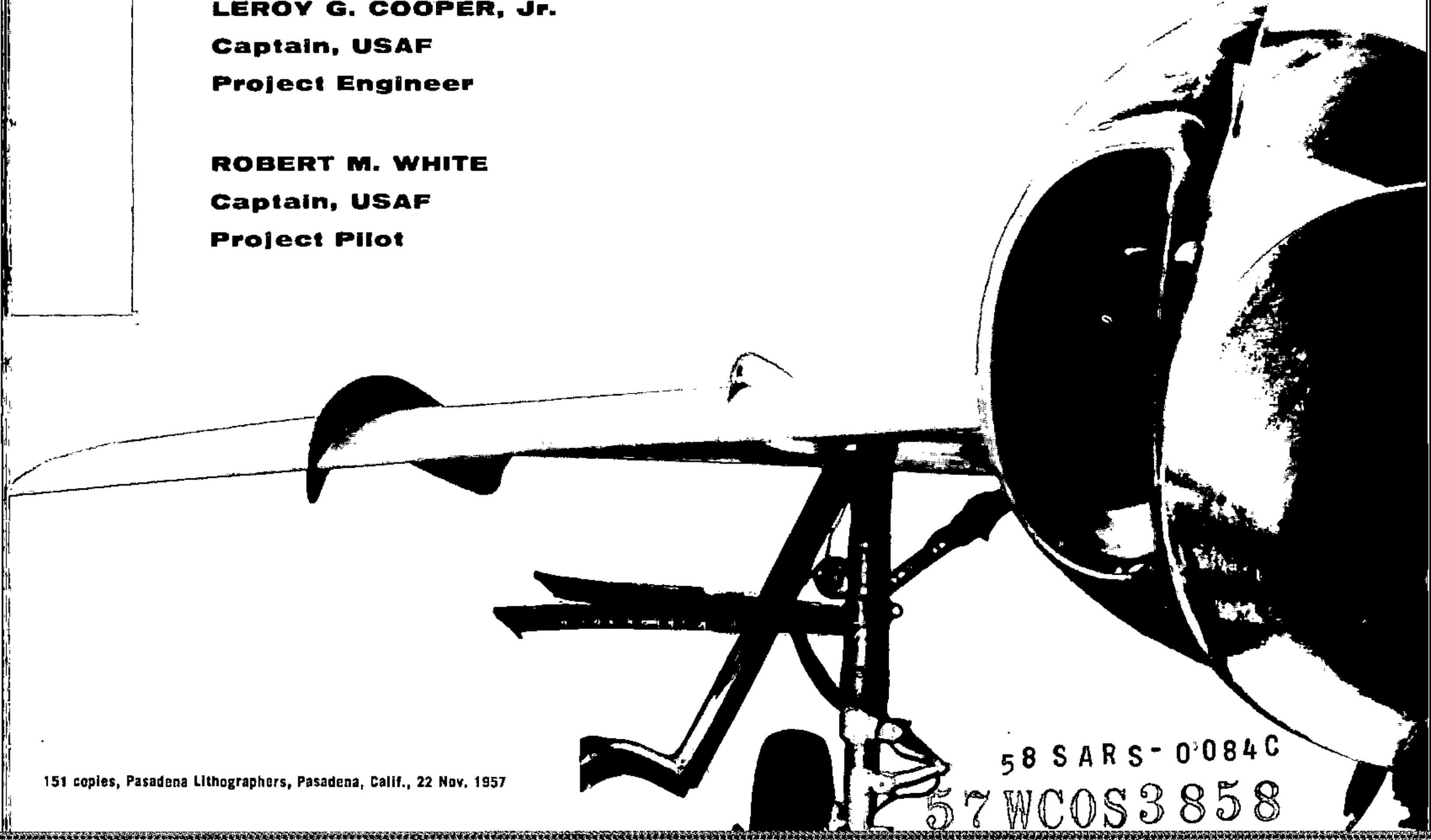
F-102A CASE XX WING PHASE IV PERFORMANCE

LEROY G. COOPER, Jr.
Captain, USAF
Project Engineer

ROBERT M. WHITE
Captain, USAF
Project Pilot



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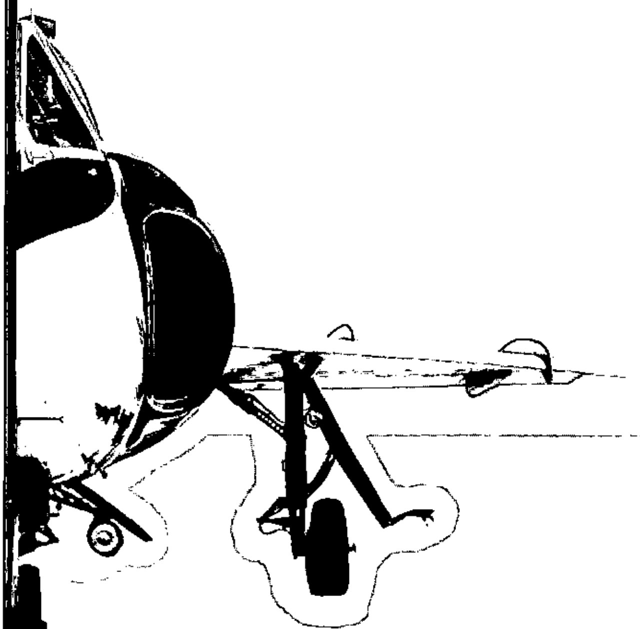
57WCOS3858



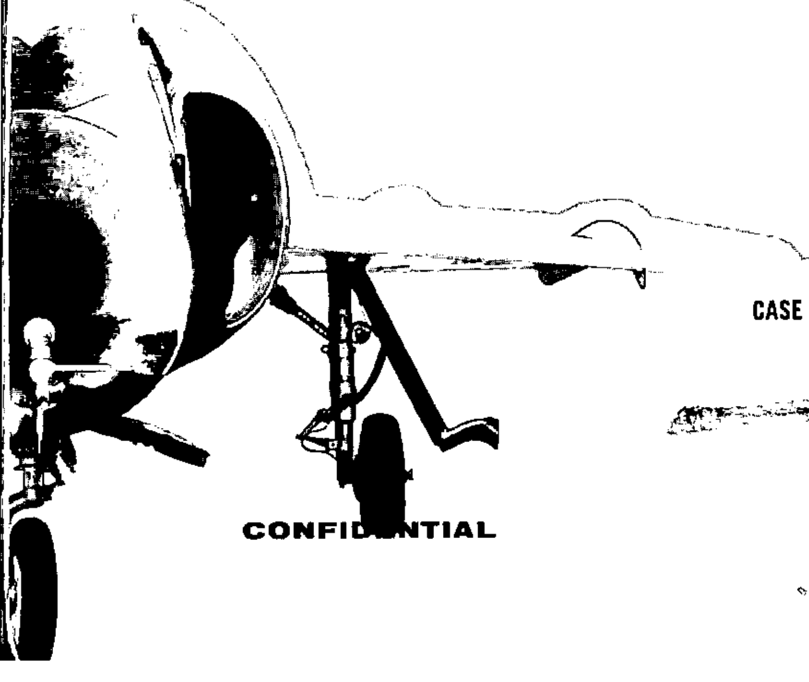
CASE X

SUMMARY

The purpose of this program was to determine the performance gain of an F-102A with a Case XX wing over that of the Case X wing in service use at this time, as well as to provide sufficient data to revise the pilot's operating handbook. The leading edge of the Case XX wing is more highly cambered than that of the Case X wing. Instead of having reflexed wing tips as in the Case X, the "droop" continues to the extreme tip.

The test program consisted of 11 hours and 35 minutes flown in F-102A S/N 56-1317 (Case XX wing) and 4 hours and 30 minutes flown in F-102A S/N 56-1001 (Case X wing). The same J57-P23 engine (S/N P607268) was used in both aircraft for these tests. The engine pressure ratio was trimmed to the trim curve dated 26 April 1957 as included in TO-102-1 throughout the program.

The maximum combat ceiling is extended to 56,000 feet, an increase of 4,200 feet, and the military power combat ceiling increased by 1,000 feet to a height of 44,200 feet by the Case XX wing. Level flight maximum speed has not been appreciably



CASE XX

CO[redacted]

changed at the medium altitudes, but a measurable increase is noted at extreme altitudes. At 50,000 feet the maximum speed has been increased by .06 Mach number. The range capability appears to have been decreased by 5 percent at low and medium altitudes, but remains the same at extreme altitudes.

The handling qualities at low indicated airspeeds show a distinct improvement. In the power approach configuration the onset of initial and heavy buffet is approximately 15 knots IAS lower than in the Case X wing aircraft. Lateral control is noticeably improved at approach airspeeds and lower.

This report has been reviewed and approved

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APPENDIX III * Data Corrected for Instrument Error

*This data has been published as a separate volume and can be obtained from ASTIA (see inside cover).

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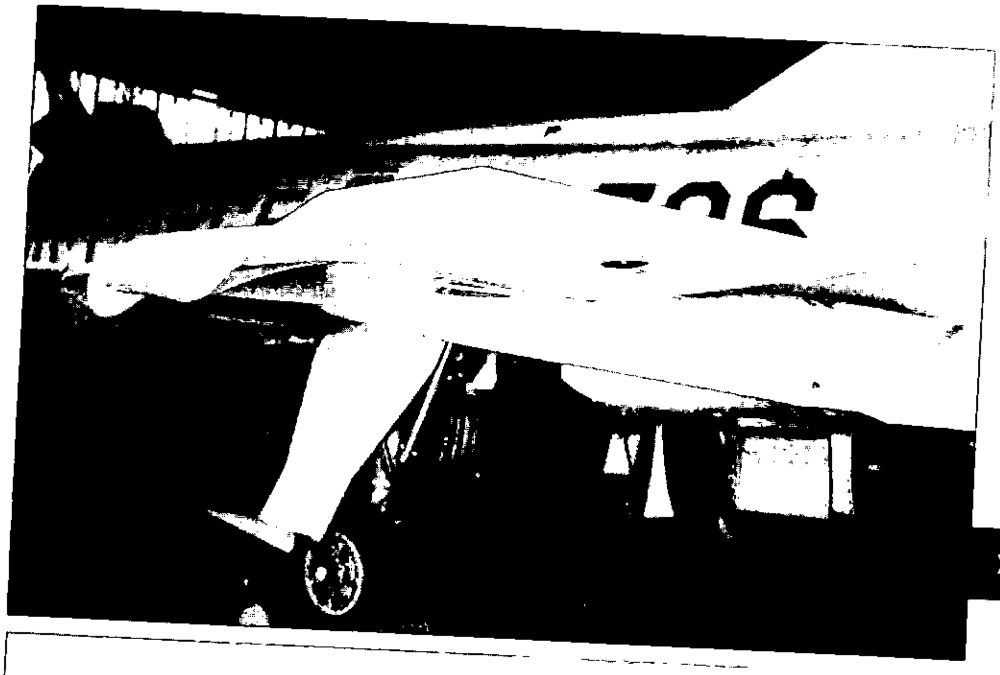

X

XX

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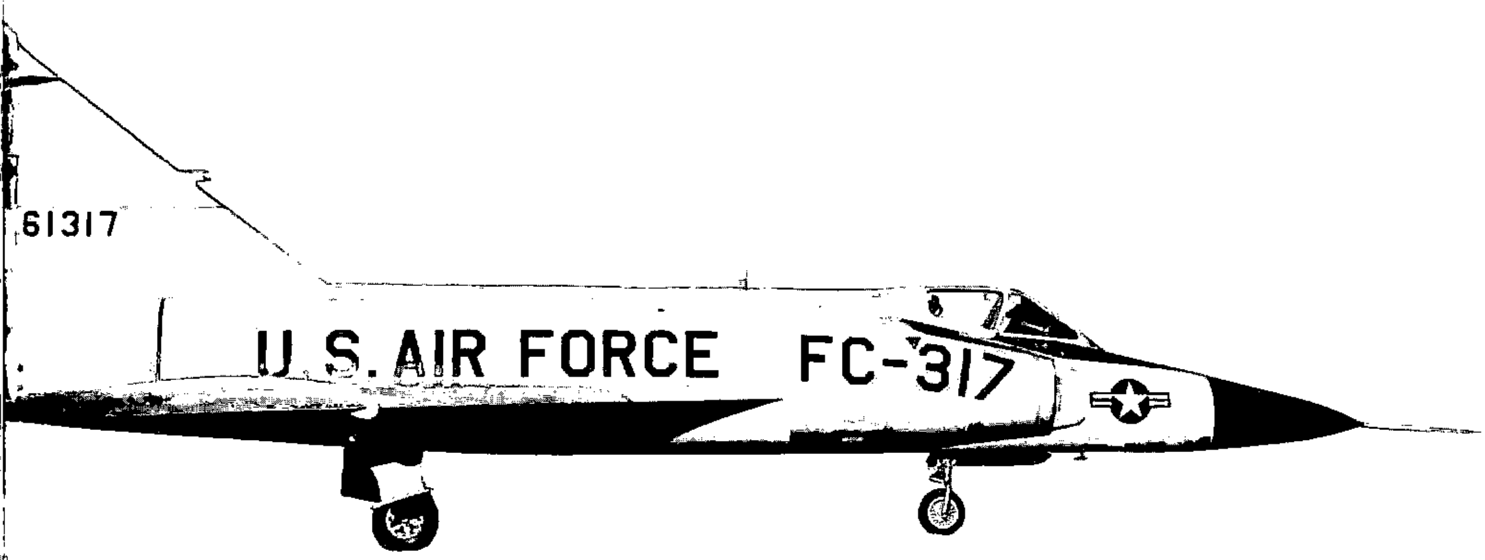

INTRODUCTION

This report presents the results of a limited Phase IV performance flight test program on F-102A S/N 56-1317 and F-102A S/N 56-1001 to evaluate the effects of change in wing camber on performance and to provide data to revise the pilot's operating handbook.

These flight tests were conducted at the Air Force Flight Test Center, Edwards Air Force Base, California, from 11 July to 9 August 1957. The program required 11:35 hours on 56-1317 (Case XX wing), and 4:30 hours on 56-1001 (Case X wing), for a total of 17 flights. The Case X wing aircraft was maintained by the Air Force and the Case XX wing aircraft was maintained by the Contractor at Convair's EAFB facility.

F-102A S/N 56-1317 is a production aircraft except for a Case XX wing replacing the standard Case X wing of earlier models. The armament was replaced by instrumentation, and the aircraft was ballasted to obtain as near a combat loading as possible. F-102A S/N 56-1001 is a production aircraft with a Case X wing and has only minor additions of test instrumentation in the cockpit for these tests. The planform and cross section of the Case X and Case XX wings are identical except for the leading edge camber and the tip design. The Case X camber extends from 6.34 percent of the local semispan while the Case XX extends from 15 percent of the local semispan and has a larger "droop." The tips of the Case X wing are reflexed up, while those of the Case XX wing continue the camber to the extreme tip. The elevon area of 56-1317 has been increased from 67.15 square feet to 70.06 square feet. Both aircraft compared here differ from the Phase IV and VI test aircraft in that the main gear has been skewed forward and the larger vertical fin installed.

In order to eliminate any variable due to differences in thrust, fuel consumption, etc, between two different engines, the same engine was used in both aircraft throughout the entire program. The thrust and engine pressure ratio trim setting were checked frequently to make certain they remained the same.

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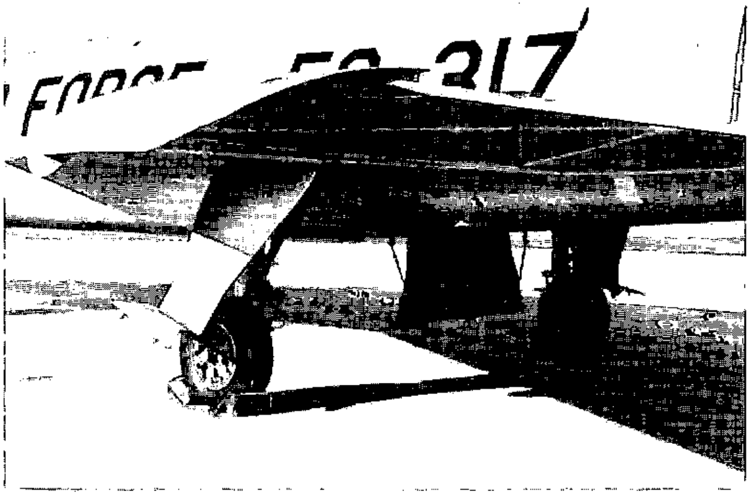


TEST RESULTS

cockpit evaluation

Both aircraft tested have representative cockpits and contain many improvements suggested in AFFTC Phase IV and Phase VI reports. The general arrangement is satisfactory except for the following deficiencies:

1. The rudder trim switch is difficult to locate and operate. The switch should be made more accessible by moving it to the left console just aft of the throttle.
2. The landing gear warning light in the landing gear handle is difficult to see.
3. The landing gear warning light comes on when the gear is not down and locked at altitudes below 10,000 feet and at indicated airspeeds below 275 knots. The sensing should be changed so the light is actuated at airspeeds encountered on final approach, or approximately 200 knots or below.

[redacted]

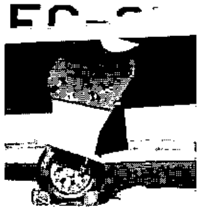
take-off

The take-off characteristics have been improved with the increased camber and larger elevon areas of the Case XX wing. The larger elevons enable the pilot to rotate to a take-off attitude approximately 10 knots IAS lower than with the Case X wing resulting in a lower minimum take-off speed. The lower drag at high angles of attack with the increased camber of the Case XX wing enable a slightly faster acceleration immediately after take-off.

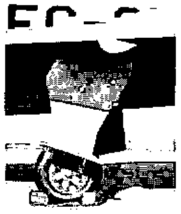
The take-off characteristics of 56-1317 and 56-1001 are improved over the earlier F-102 aircraft used in Phase IV and VI tests because of the skewed landing gear. The straight landing gear of the early aircraft resulted in a greater nose down moment and a higher speed was required prior to the elevons becoming sufficiently effective to rotate the airplane. Skewing of the gear forward on later aircraft has reduced the rotation speed appreciably. A significant reduction in minimum take-off speed is realized as a result.

The minimum speeds for rotation with full back stick in the ground roll and subsequent take-off are tabulated for each of the aircraft tested along with the information from the prior Phase IV tests (AFFTC-TR-56-23).

Phase IV
Unskewed Gear
Small Elevon
Case X Wing



56-1001
Skewed Gear
Small Elevon
Case X Wing



56-1317
Large Elevon
Skewed Gear
Case XX Wing

	Phase IV	56-1001	56-1317
Nose wheel off	140 knots IAS* / 2-5	115 knots IAS	105 knots IAS
Take-off	155 knots IAS	135 knots IAS	130 knots IAS
CG	....	29.15 percent MAC	29.8 percent MAC

*Lower nose wheel off speed can be obtained by applying aft stick at approximately 125 knots.

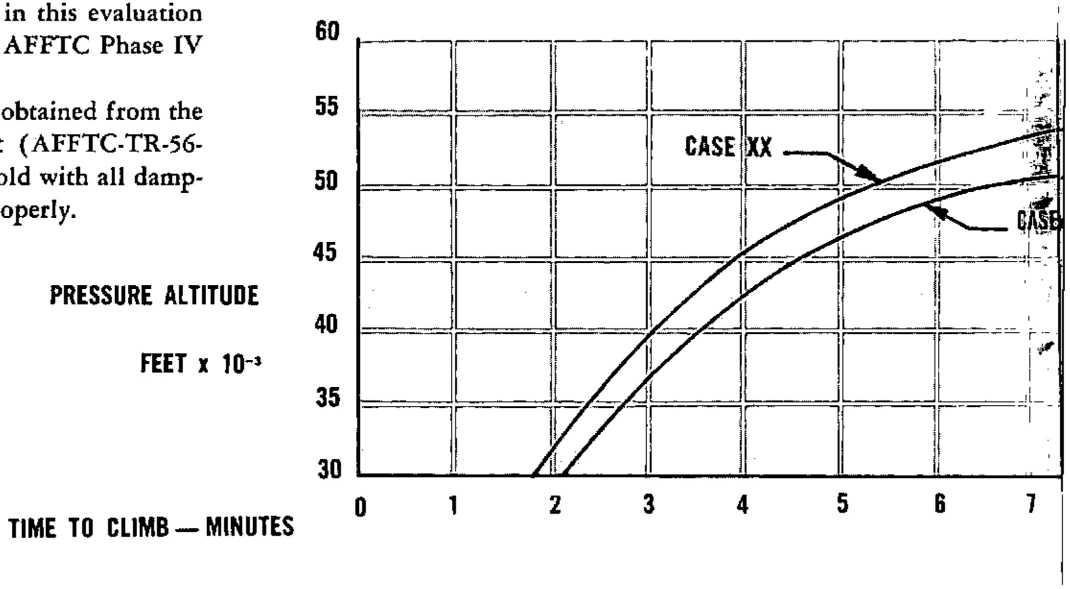
climb performance

The climb performance of the F-102A has been substantially increased by the addition of the Case XX wing. The maximum power combat ceiling has been increased 4,200 feet to a height of 56,000 feet, and the time to climb to 50,000 feet has been decreased by more than 1 minute. The military power climb ceiling has been increased by 1,000 feet to a ceiling of 44,200 feet. Climb data is presented in Figures 1 through 4, Appendix I, and summarized in the following table.

The maximum power combat ceiling obtained by an F-102A in the AFFTC Phase IV performance test was 56,000 feet. However, this aircraft had the engine pressure ratio trimmed higher than is permitted at this time. The decrease in thrust resulting from the lower trim in use at present is the reason that the rate of climb measured on 56-1001 in this evaluation is lower than that presented in the AFFTC Phase IV performance test report.

The climb schedules used were obtained from the AFFTC Phase IV performance test (AFFTC-TR-56-23). These schedules were easy to hold with all dampers and the trim servo operating properly.

MAXIMUM POWER CLIMB PERFORMANCE
(Times are from Sea Level)



Altitude ft.	Aircraft No. 56-1317 (Case XX Wing) Mach Number	Rate of Climb ft/min	Time to Climb min.	Weight lbs.	Aircraft No. 56-1001 (Case X Wing) Rate of Climb ft/min.	Time to Climb min.	Weight lbs.	Mach Number
Sea Level	.96	19,000	0	28,110	....	....	....	....
10,000	.96	18,000	.5	27,100	....	....	....	....
20,000	.96	15,400	1.1	26,600	....	....	....	....
30,000	.96	11,400	1.8	26,200	10,000	2.1*	26,100	.95
40,000	.96	7,000	3.0	25,800	5,900	3.5	25,800	.95
50,000	.96	2,800	5.2	25,400	1,200	6.3	25,400	.95
52,800	.96	1,800	6.1	25,200	0	11.1	25,000	.95
55,000	.95	1,000	8.1	25,000	....	....	....	....
57,000	.94	0	13.7	24,800	....	....	....	....

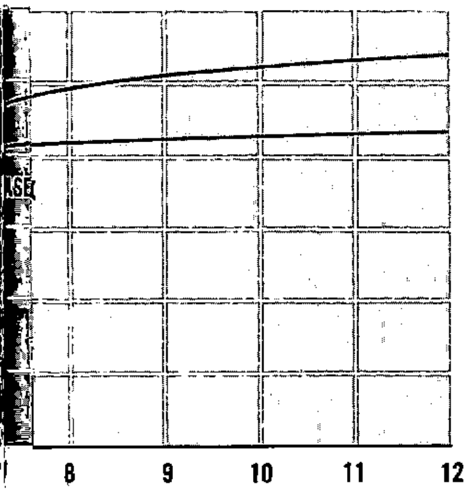
* Time from previous Phase IV tests with Case X wing, corrected to current trim schedule.

level flight performance

Test data indicates the Case XX wing to be .01 Mach number slower in level flight maximum speed at 35,000 feet. This is within the inaccuracies of measurement and repeatability, and is not considered at all significant.

The Case XX wing increases the level flight maximum speed of the aircraft by .03 Mach number at 45,000 feet and .06 Mach number at 50,000 feet. (Maximum speed at 50,000 feet is 1.05 Mach number with the Case XX wing.) Maximum speed data is presented in Figure 5.

The increase in camber has penalized the Case XX wing approximately 5 percent in range at altitudes below 40,000 feet, but little difference can be noted at higher altitudes. A comparison of the specific range parameters with those obtained in the Phase IV tests indicates a reduction of 3 percent at 45,000 feet and 10 percent at 40,000 and 30,000 feet. However, the specific fuel consumption of the engine used in this evaluation appeared to be 3 to 5 percent greater than that obtained during the Phase IV tests as noted on Figure 8, Appendix I, of this report and Figure 10 of AFFTC-TR-56-23. This difference in fuel consumption is responsible for the difference in specific range at 45,000 feet and one-half of the difference at lower altitudes,



qualitative handling characteristics

The most noticeable difference between the Case XX wing and the Case X wing is further improvement of the low speed handling characteristics of the Case XX wing. In the power approach configuration the onset of initial and heavy buffet has been decreased by approximately 15 knots IAS. At a weight of 24,000 pounds, the initial buffet occurs at approximately 140 knots, and heavy buffet is encountered at approximately 125 knots. The traffic pattern, final approach, and landing can be comfortably flown 10 knots slower than with the Case X wing.

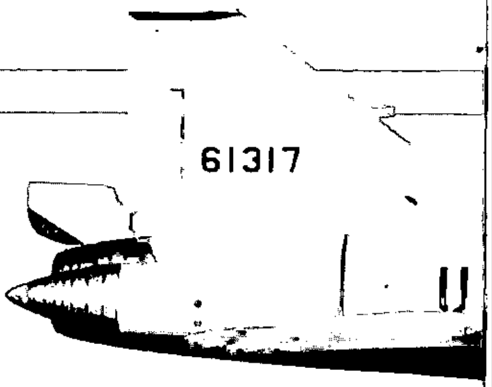
Stalls are still difficult to obtain because of the high rates of sink encountered. One distinct improvement in stall approaches is the increased lateral control down to 90 knots, the minimum airspeed investigated. Due to high rates of sink and control deterioration, it is recommended that airspeeds below 125 knots be avoided.

There is no noticeable degradation of stability and control with the Case XX wing. Longitudinal control has been increased as discussed under "Take-Off" largely because of the increase in elevon area. The qualitative evaluation did not disclose a noticeable improvement in lateral control except at low airspeeds. However, a quantitative evaluation would probably show that the larger elevons improve lateral control throughout the speed range.

MC-1 partial pressure suit

It was again very apparent during this series of tests that the ease and facility with which the F-102A is flown is greatly impaired by the necessity of wearing the partial pressure suit under conditions of high altitude flight. The pressure suit restricts the pilot's freedom of movement within the cockpit to the extent that the aft console circuit breakers cannot be reached and the IFF control console cannot be seen. With the canopy in the open position the pilot is unable to reach the canopy closing handle. The pilot's field of vision to the rear is so restricted by the pressure suit that the wing tips (normally visible) cannot be seen. Visibility during steep turns is compromised by the limited freedom of movement imposed by the pressure suit. Pilots have considerable difficulty in seeing another aircraft well enough to fly wing position.

The pressure suit is uncomfortable under the best conditions, and is almost intolerable in warm weather. This is a particularly serious problem in tactical use, since pilots on stand-by are required to wear the suit for considerable periods of time on the ground.



CONCLUSIONS

THE INCORPORATION OF THE CASE XX WING on an F-102A in place of the Case X wing has produced an increase in maximum power combat ceiling of 4,200 feet, and an increase of .06 Mach number level flight maximum speed at 50,000 feet. Nautical air miles per pound of fuel are decreased 5 percent at low and medium altitudes, but the Case XX wing provides the same range parameters as the Case X wing if a higher cruising altitude is used. The low speed handling characteristics are improved.

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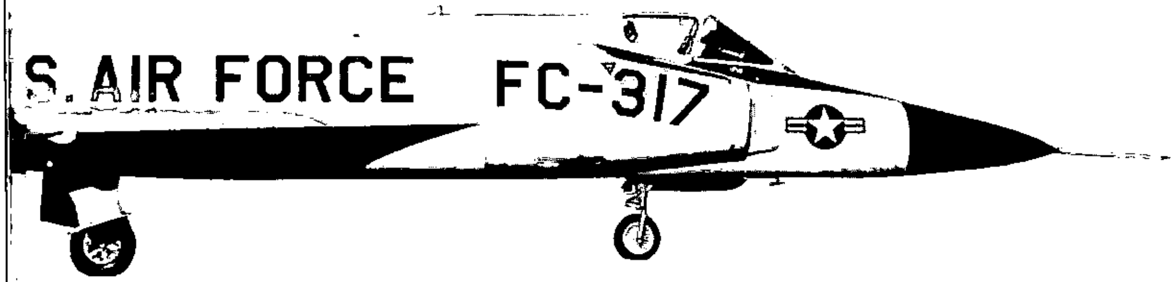


RECOMMENDATIONS

IT IS RECOMMENDED THAT:

1. The Case XX wing be provided on future production F-102A aircraft.

2. The performance figures in the pilot's operating handbook be adjusted to reflect the performance data included in this report.

[redacted]

APPENDIX I

data analysis methods

references

1. Flight Test Engineering Manual, AFFTC-TR No. 6273 (Revised) May 1951.
2. Flight Research Branch Technical Note R-4, Kenneth Lush.
3. Phase IV Performance Test on F-102A, AFFTC-TR-56-23, January 1957.

thrust computation

In-flight thrust computation was made using the methods and procedures outlined in reference 1. The nozzle coefficient was checked with a static thrust run. The extrapolations of Pratt and Whitney curves No. 17818, dated 31 October 1955, and No. 17821, revised 11 June 1956, were used for higher in-flight values. Air flow for computing ram drag was obtained from Pratt and Whitney's curve No. 17145 using a ram efficiency of 96 percent in all subsonic speeds.

The engine was trimmed as recommended by TO-102-1 at the start of the tests. This trim was rechecked several times during the program. This data is shown in Figure 15 of this Appendix with the recommended trim curve used. An in-flight bias curve was then obtained from maximum and military power tests and presented in Figure 13 and 14 of this Appendix. Standard day pressure ratios for correction to standard conditions were taken from the bias curve directly.

atmosphere used

To facilitate the comparison between this test and previous F-102 tests, all performance data was reduced using the old NACA standard day atmosphere, which has a standard lapse rate of 2 degrees centigrade per 1,000 to 35,332 feet altitude and is then constant (—55 degrees C) throughout the altitudes reached in this test.

drag and lift computations

Drag data from level flight speed power, maximum speed points and climb data were computed as outlined in reference 1. The angle of climb was computed and used in obtaining lift and drag from the climb data.

Maximum speed was determined using Figures 9 (corrected net thrust versus Mach number) and 13 (in-flight bias curve) to correct the test day point to a standard day. A series of successive approximations was required.

■ continuous climbs

Continuous climbs were corrected to standard day conditions with the procedures outlined in reference 1. The data was corrected to the thrust indicated by the in-flight bias curve. Exhaust gas temperatures were corrected to the standard day thrust from the bias curves, but no attempt was made to correct for a possible overtemperature condition because of the inaccuracies inherent in the exhaust gas temperature indications.

■ fuel flow

Fuel flow data was corrected to standard conditions using Figure 8 of this Appendix. The slope of the fuel flow curve was employed to correct the test point rather than obtaining standard values directly from the curve.

■ airspeed calibrations

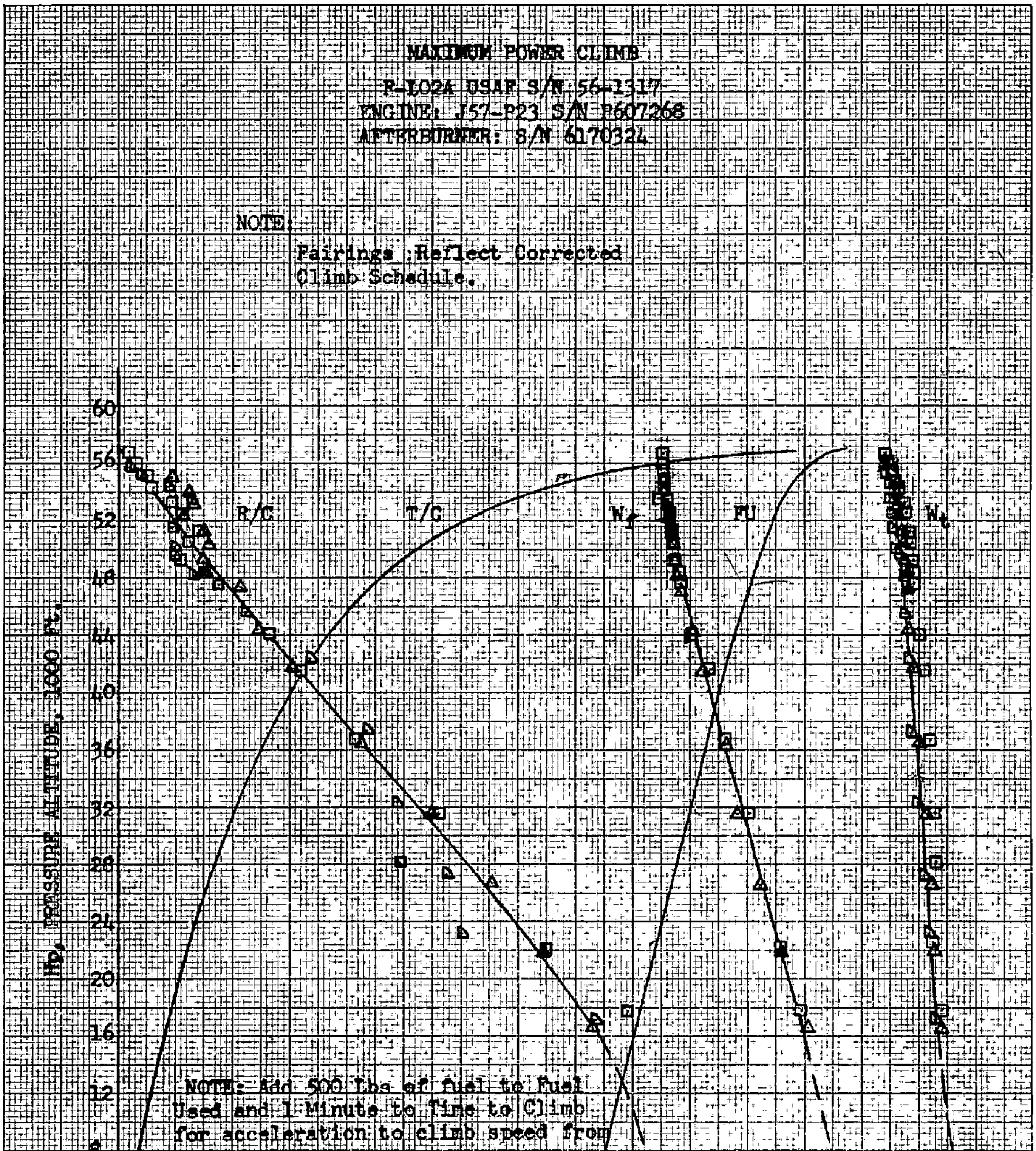
Standard production airspeed systems were used in both aircraft in this evaluation. A calibrated pacer aircraft was used to obtain airspeed and altitude position error data to verify the calibration curves obtained during the Phase IV performance flight tests.

■ ambient temperatures

Ambient temperatures were determined from Rawinsonde data obtained from the base weather detachment before and after the flight.

performance plots

Figure No.
1 Power Climbs 10
5 Level Flight Performance 18
11 Drag Polars 24
12 Drag Rise 25
13 Pressure Ratios 26
16 Static Thrust Calibration 29
17 Airspeed Calibration 30



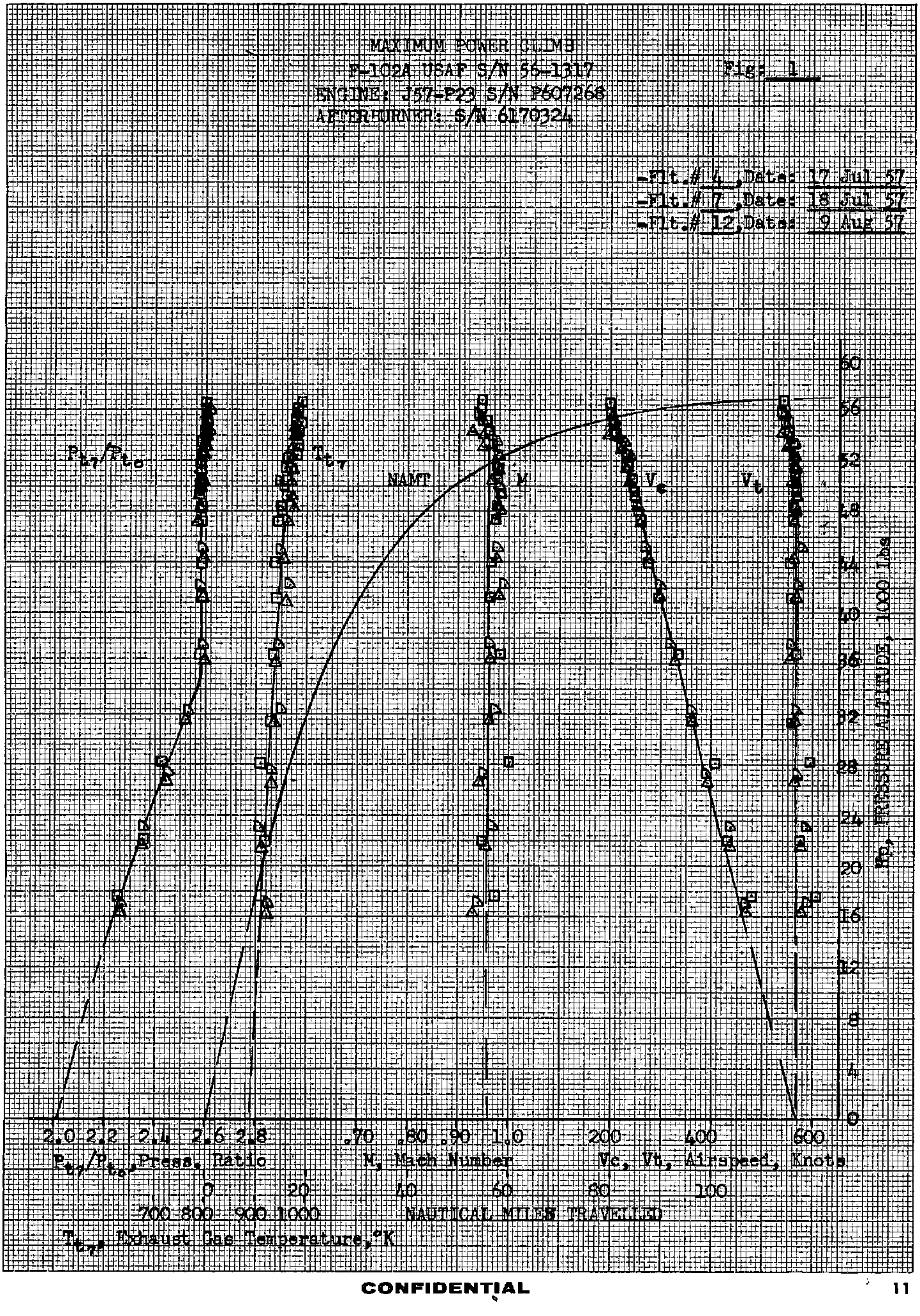
MAXIMUM POWER CLIMB

F-102A USAF S/N 56-1317
ENGINE: J57-P23 S/N P607268
AFTERBURNER: S/N 6170324

NOTE:

Fairings Reflect Corrected Climb Schedule.

NOTE: Add 500 Lbs of fuel to Fuel Used and 1 Minute to Time to Climb for acceleration to climb speed from brake release.



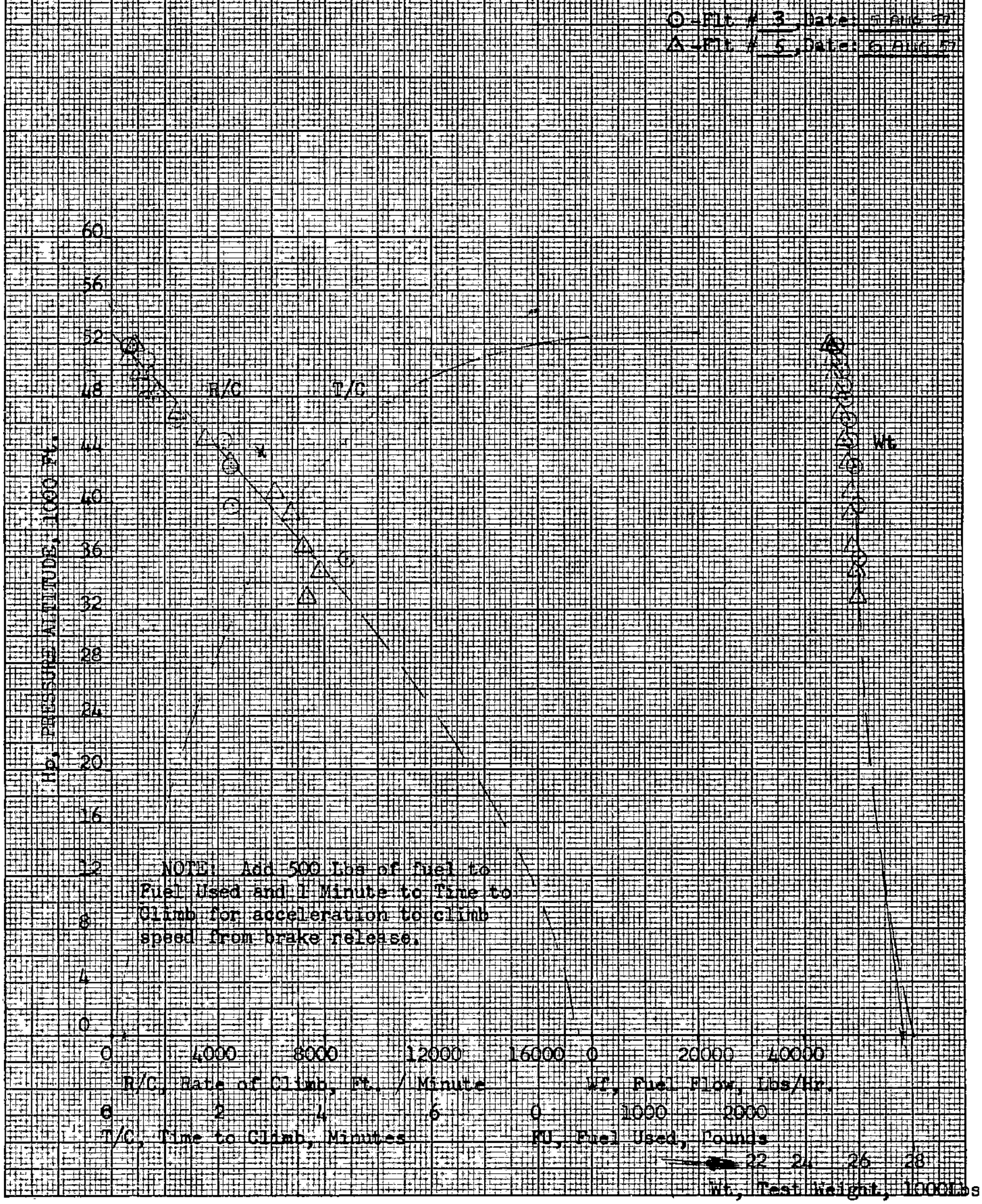
MAXIMUM POWER CLIMB
F-102A USAF S/N 56-1317
ENGINE: J57-P23 S/N P607268
AFTERBURNER: S/N 6170324

Fig: 1

28110
500
27610
25800
1810

MAXIMUM POWER CLIMB
F-102A USAF S/N 56-1001
ENGINE: J57-P23 S/N p607268
AFTERBURNER: S/N 6170324

FIG: 2



NOTE: Add 500 Lbs of fuel to Fuel Used and 1 Minute to Time to Climb for acceleration to climb speed from brake release.

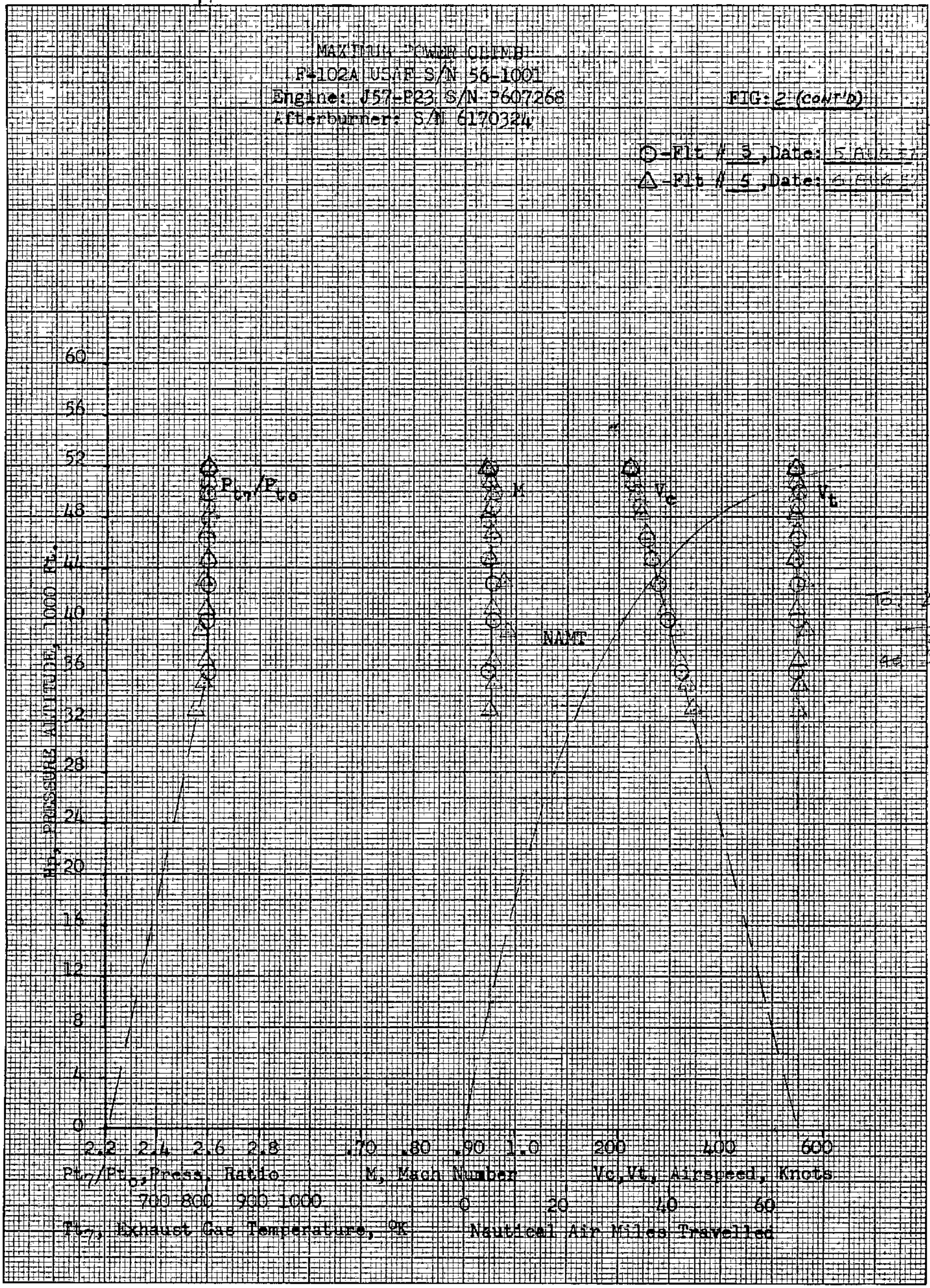


FIG: 2 (CONT'D)

X

XY

28163
500
27663
2587
1863

28110
27610
25895
1715

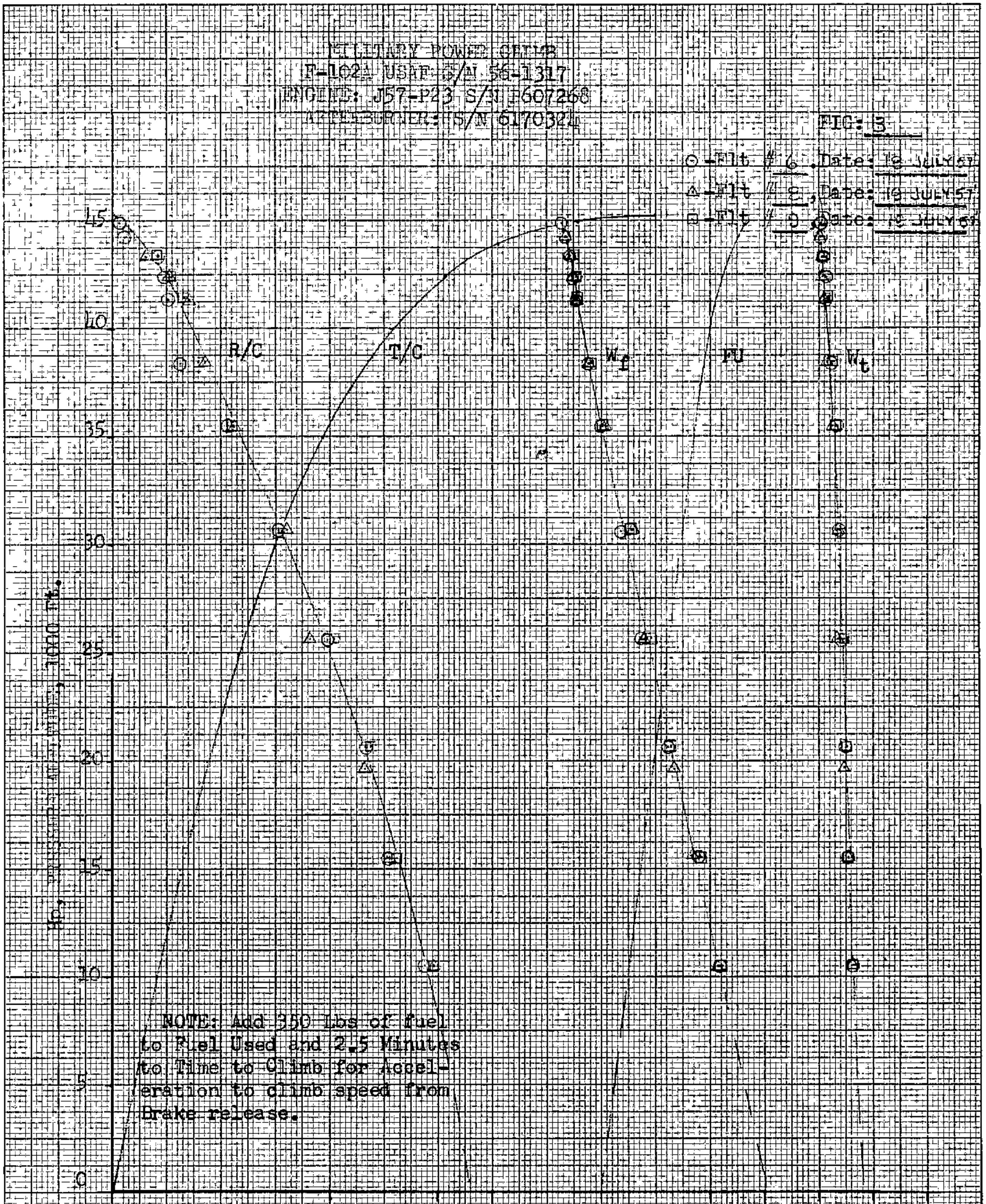
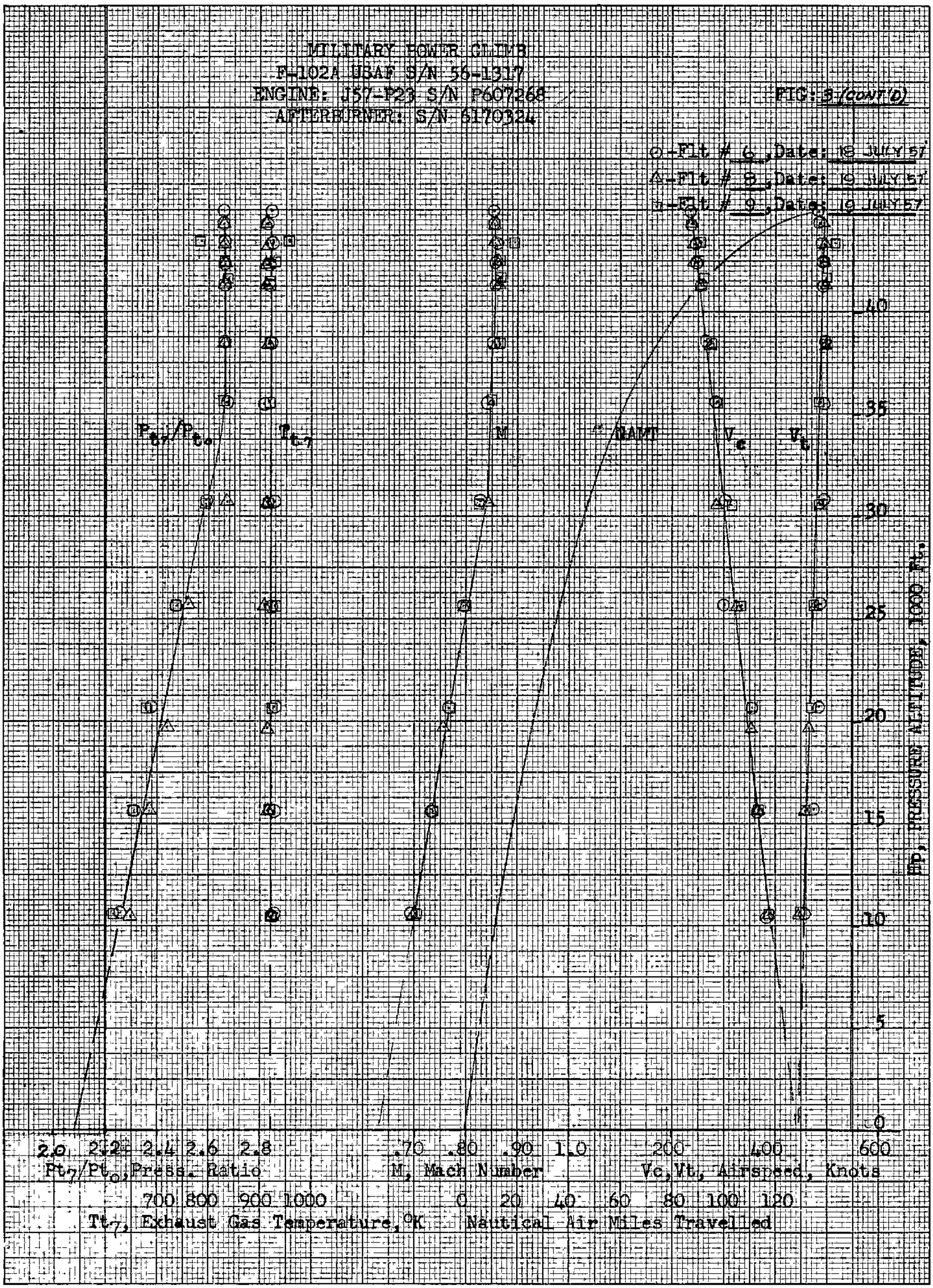


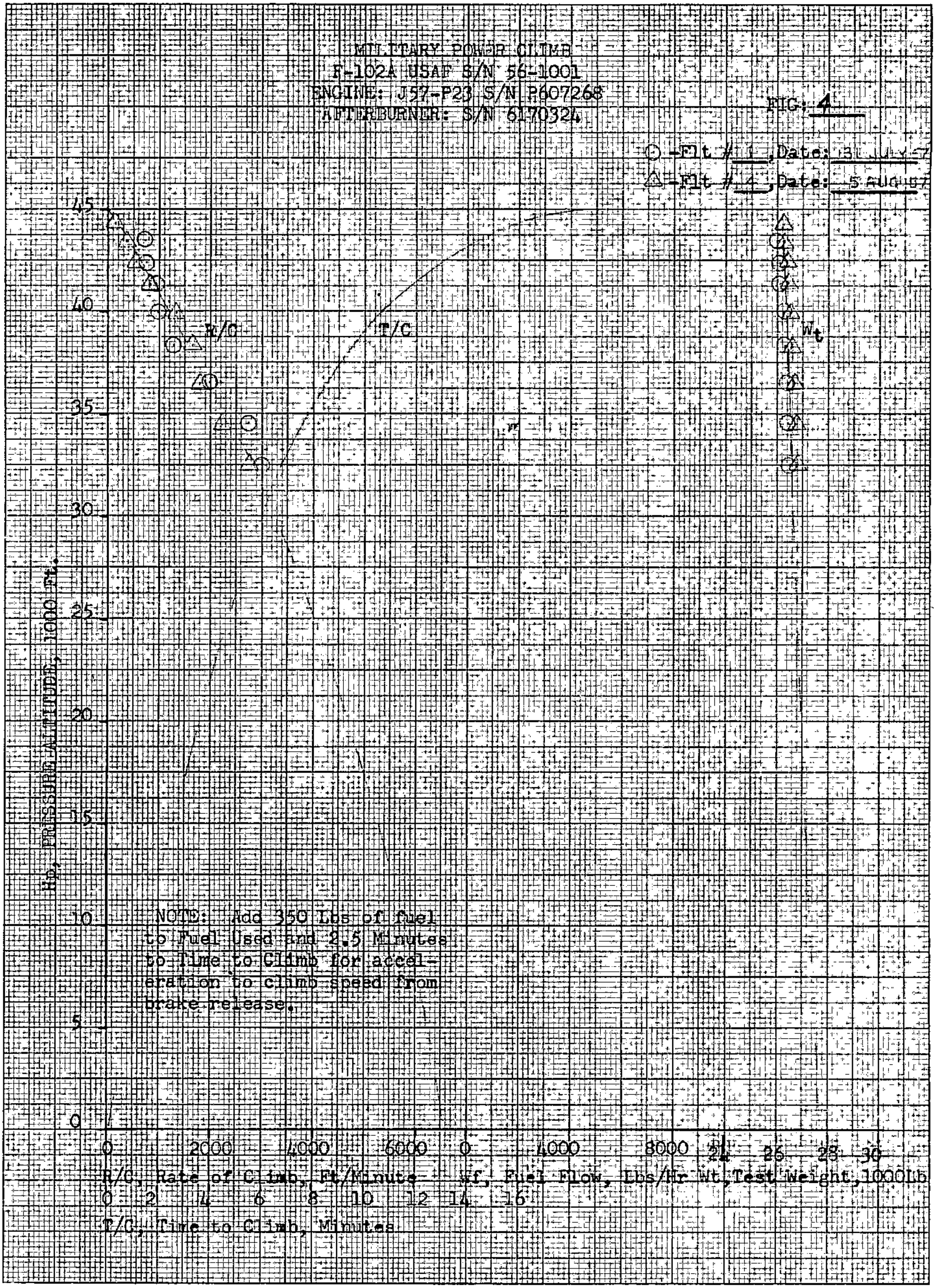
FIG: 3

NOTE: Add 350 Lbs of fuel to Fuel Used and 2.5 Minutes to Time to Climb for Acceleration to climb speed from Brake release.



MILITARY POWER CLIMB
F-102A USAF S/N 56-1317
ENGINE: J57-P23 S/N P607268
AFTERBURNER: S/N 6170324

FIG: 3 (CONT'D)



MILITARY POWER CLIMB
F-102A USAF S/N 56-1001
ENGINE: J57-P23 S/N P607268
AFTERBURNER: S/N 6170324

FIG: 4

NOTE: Add 350 Lbs of fuel to Fuel Used and 2.5 Minutes to Time to Climb for acceleration to climb speed from brake release.

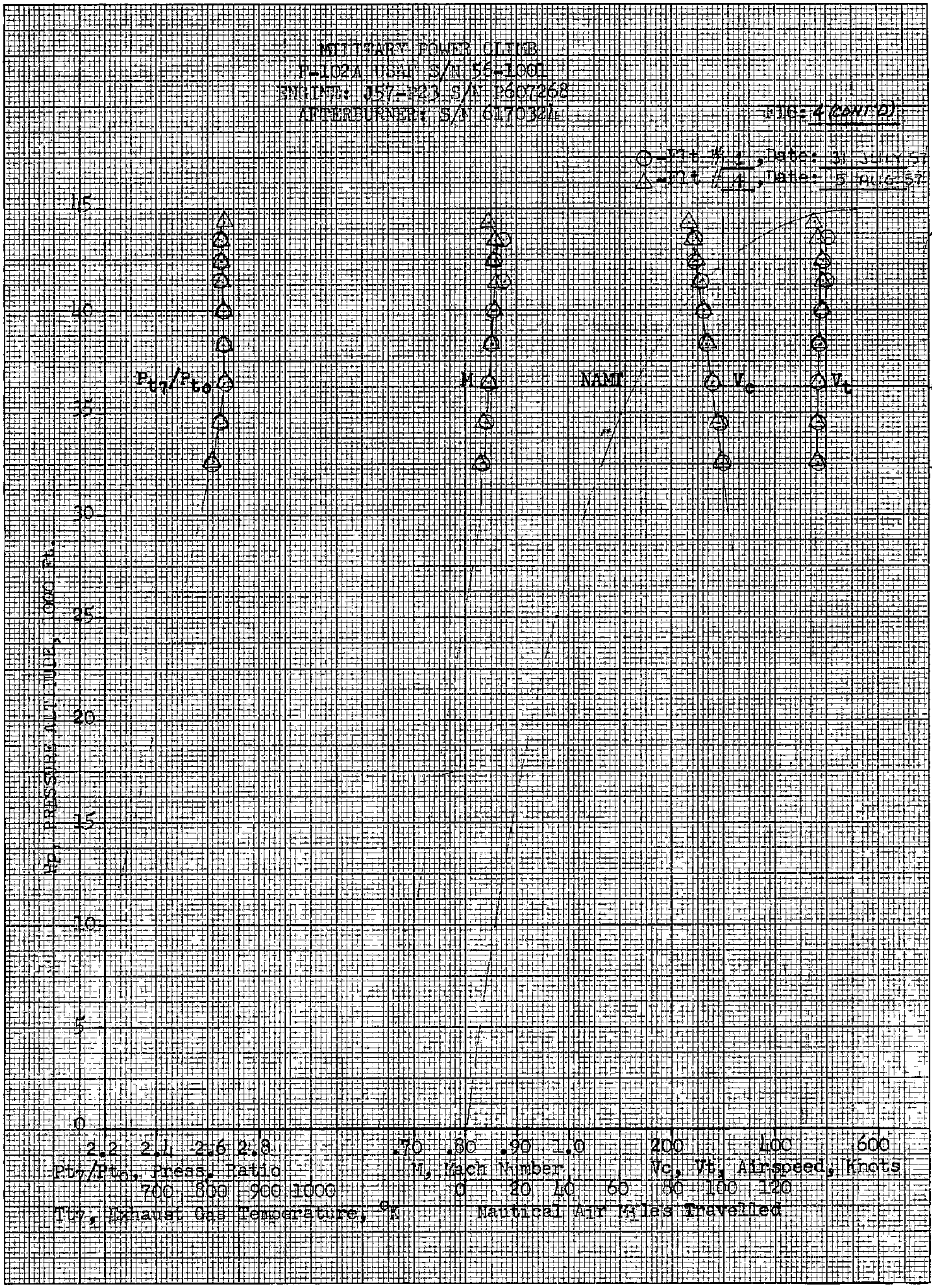


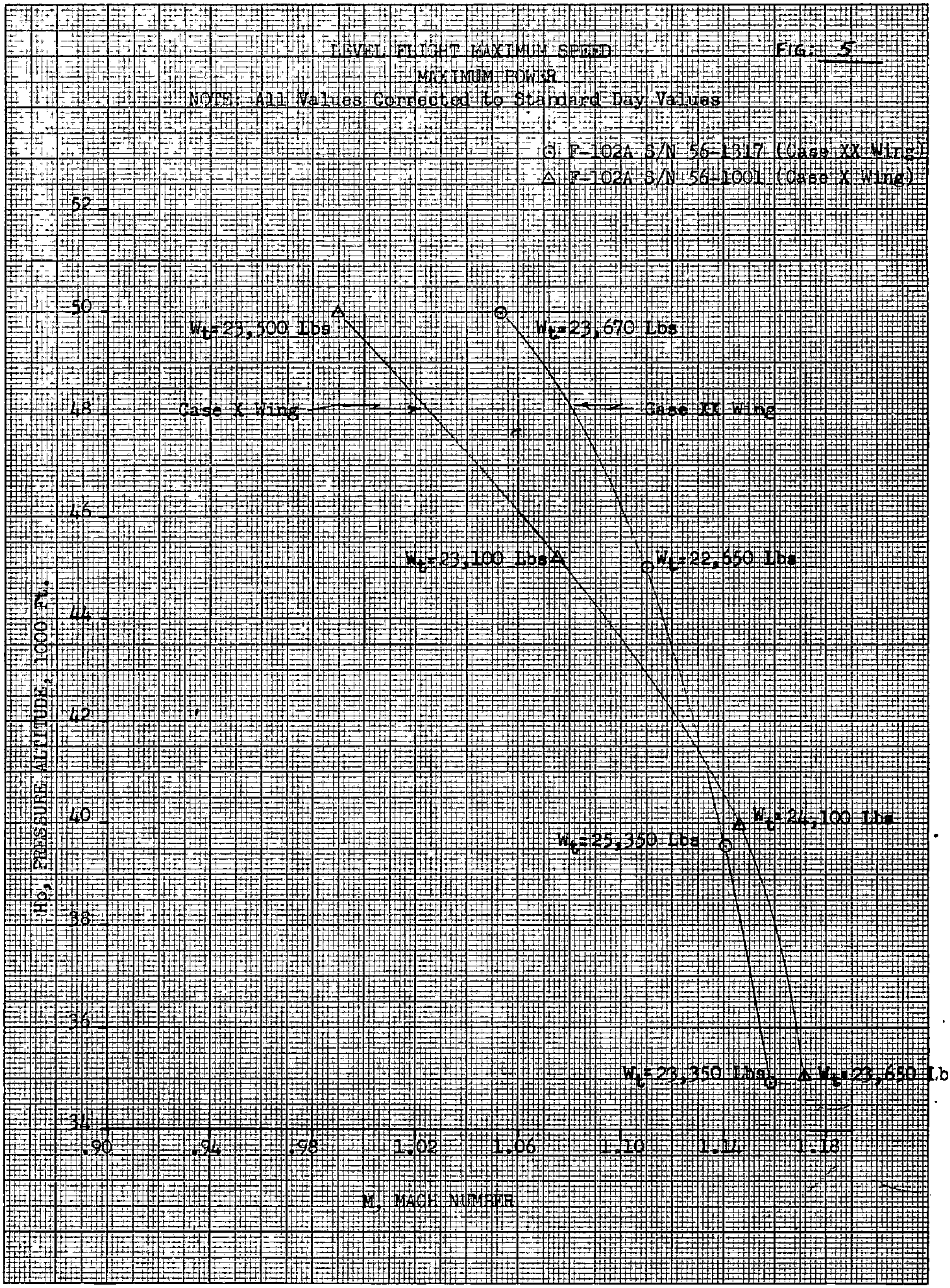
FIG. 4 (CONT'D)

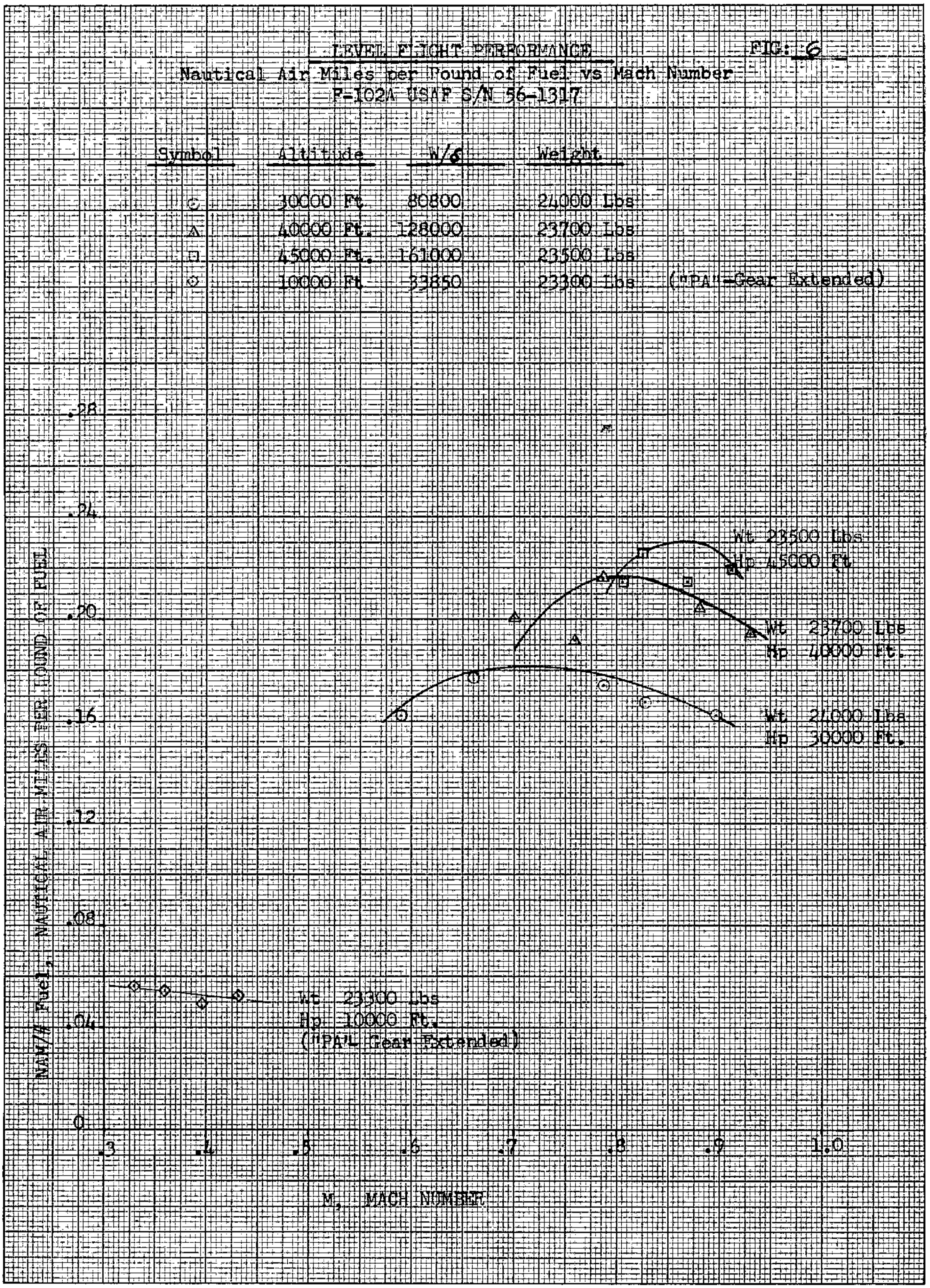
C_{D_z} &

C_{L_z}

FIG. 5

LEVEL FLIGHT MAXIMUM SPEED
MAXIMUM POWER
NOTE: All Values Corrected to Standard Day Values



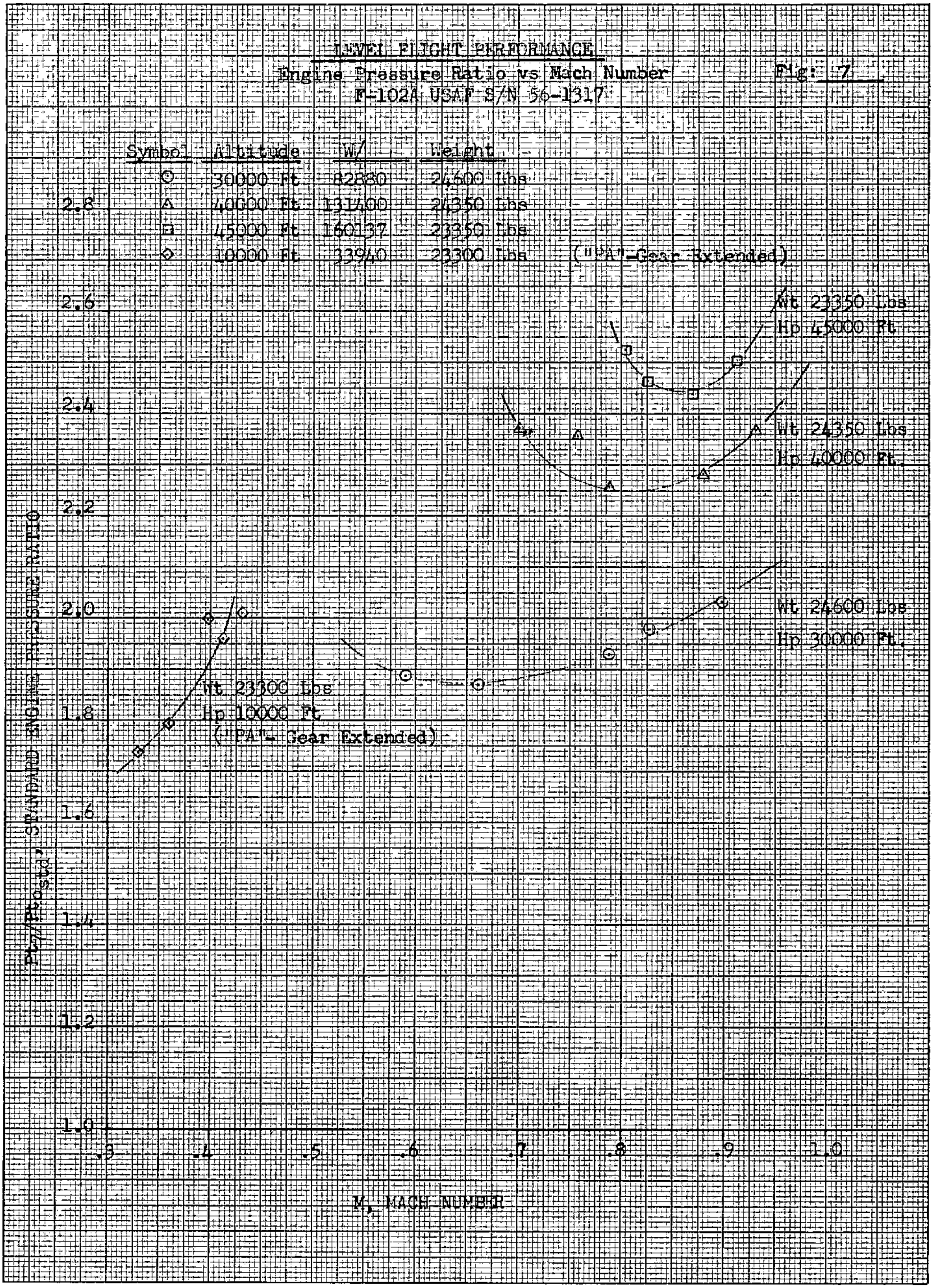


LEVEL FLIGHT PERFORMANCE

FIG: 6

Nautical Air Miles per Pound of Fuel vs Mach Number
F-102A USAF S/N 56-1317

Symbol	Altitude	W/δ	Weight	
⊙	30000 Ft	80800	24000 Lbs	
△	40000 Ft.	128000	23700 Lbs	
□	45000 Ft.	161000	23500 Lbs	
◇	10000 Ft	33850	23300 Lbs	("PA"–Gear Extended)



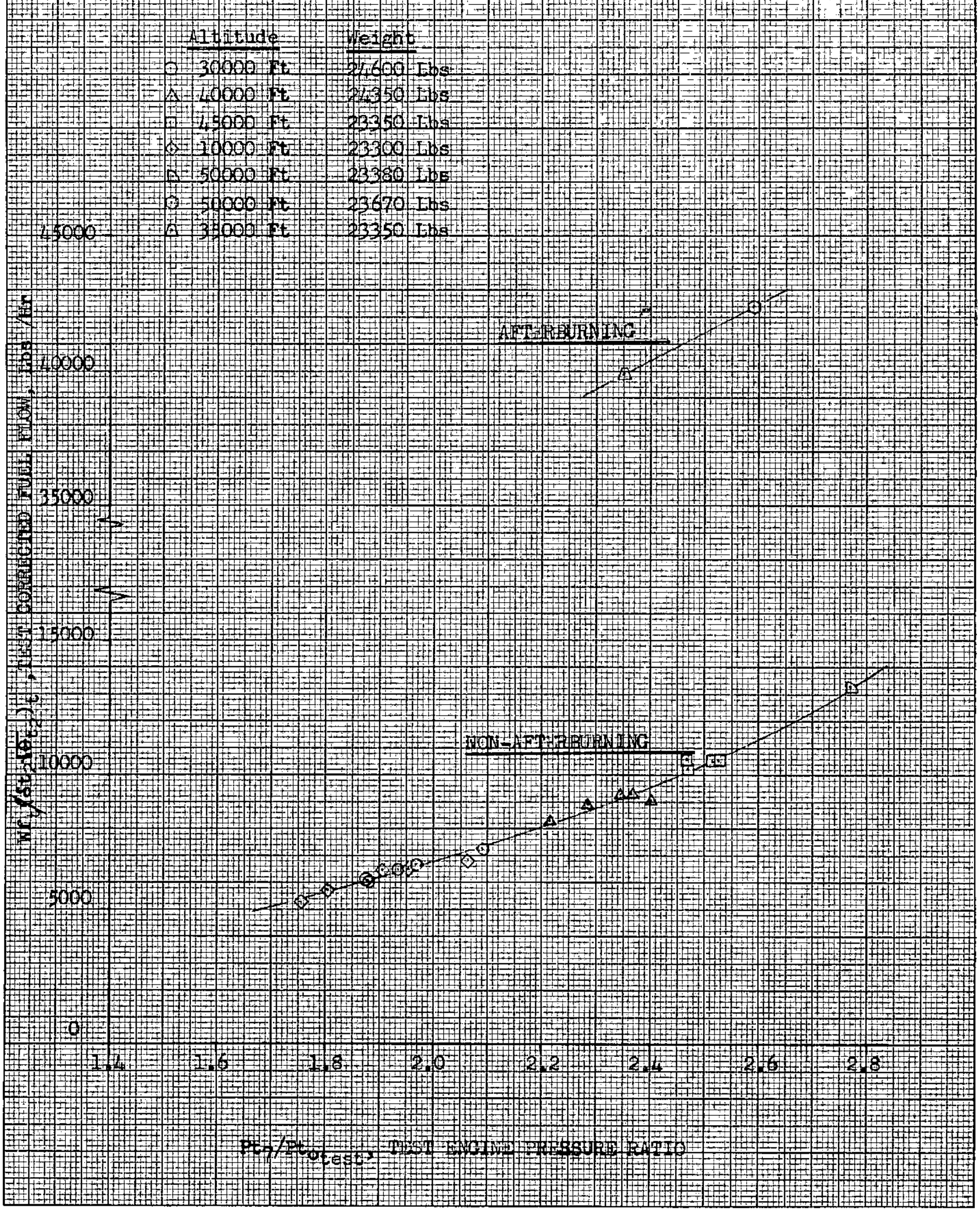
LEVEL FLIGHT PERFORMANCE
Engine Pressure Ratio vs Mach Number
F-102A USAF S/N 56-1317

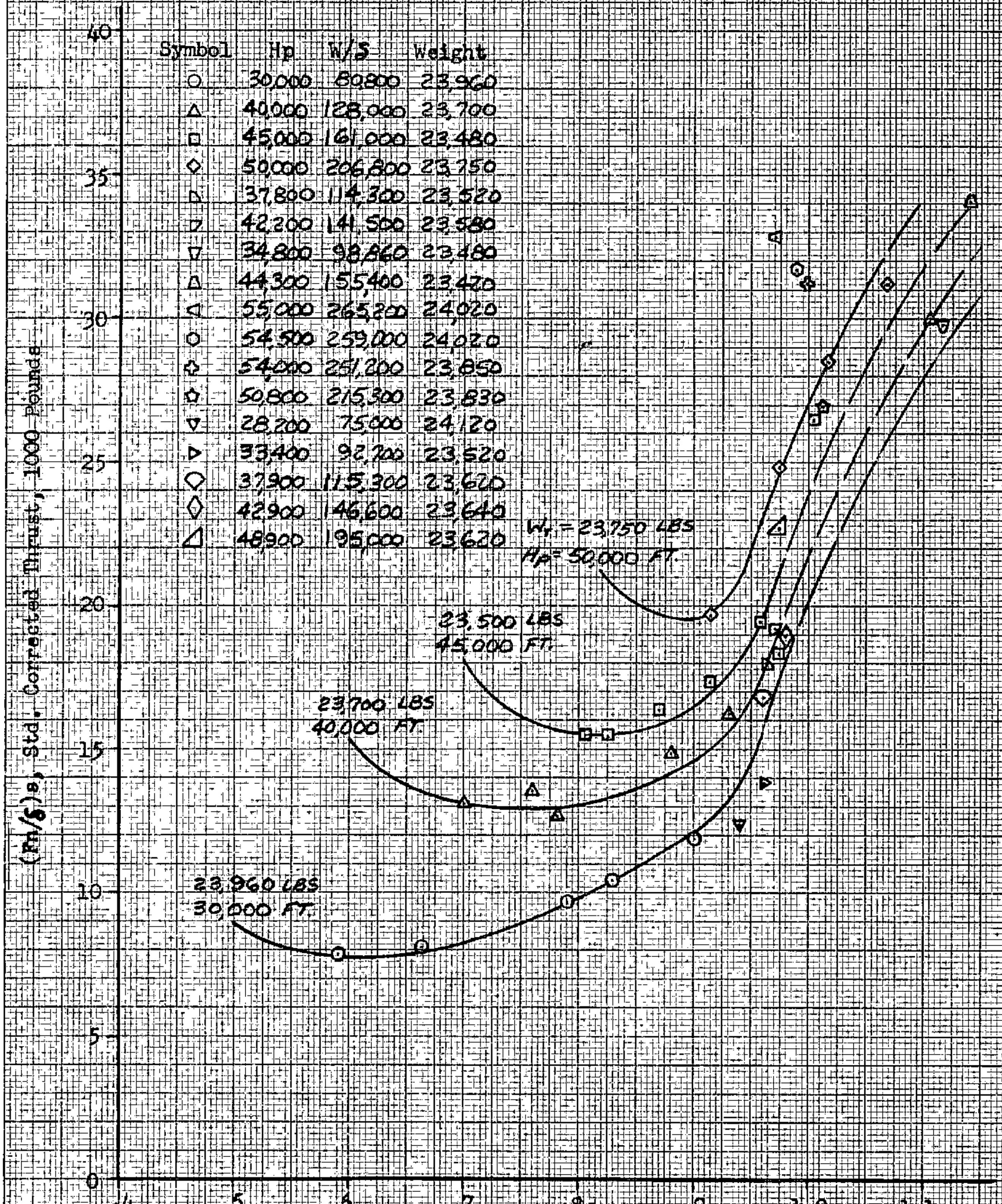
Fig: 7

Symbol	Altitude	W/	Weight	
⊙	30000 Ft	82880	24600 Lbs	
△	40000 Ft	131400	24350 Lbs	
⊡	45000 Ft	160137	23350 Lbs	
◇	10000 Ft	33940	23300 Lbs	("PA"-Gear Extended)

FUEL CONSUMPTION FIG 8

Corrected Fuel Flow vs Pressure Ratio
F-102A USAF S/N 56-1317





LEVEL FLIGHT PERFORMANCE
Corrected Thrust vs. Mach
F-102A USAF S/N 56-1317
Engine: J-57-P23 S/N P607268
Afterburner: S/N 6170324

Fig: 9

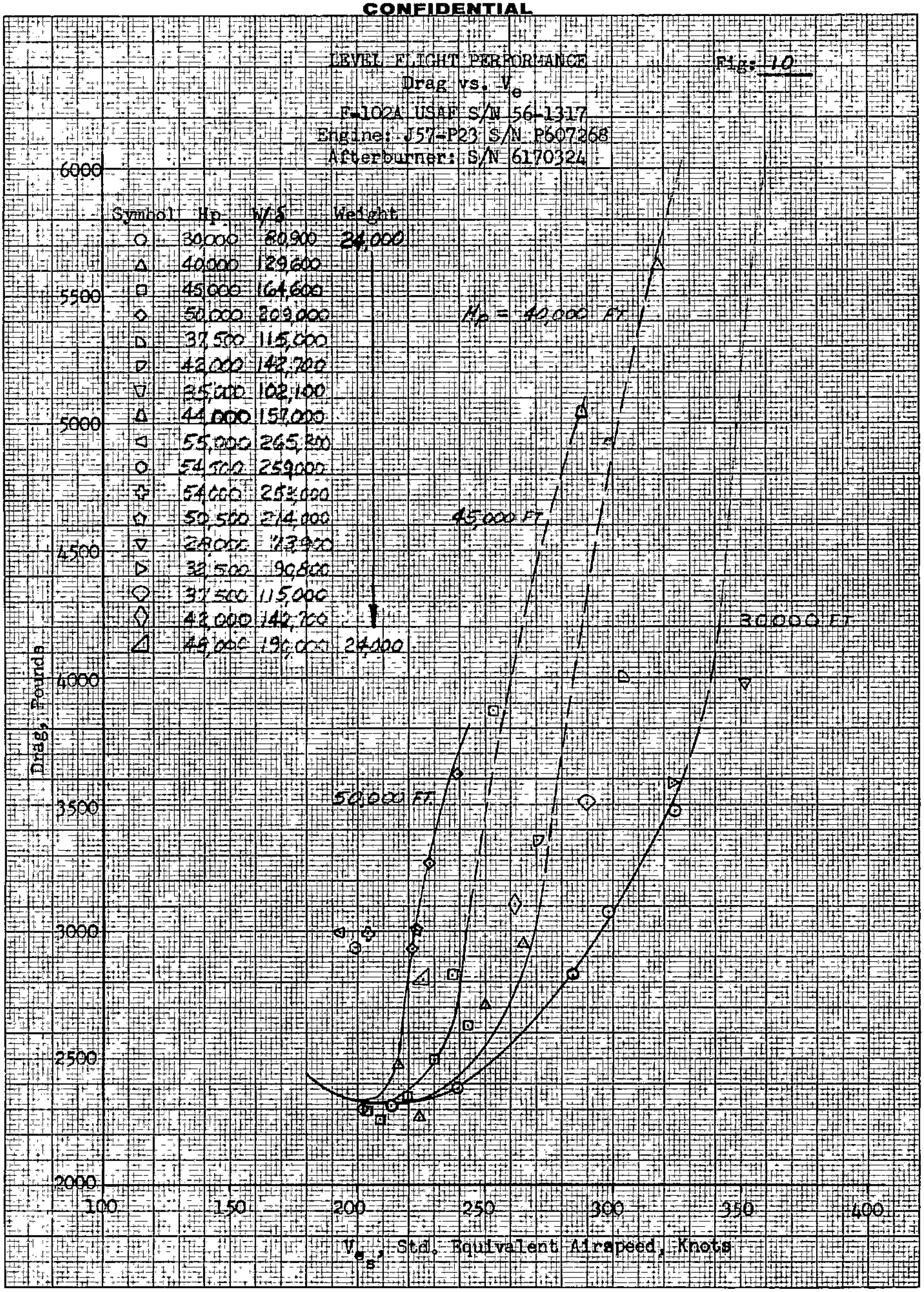


Fig: 10

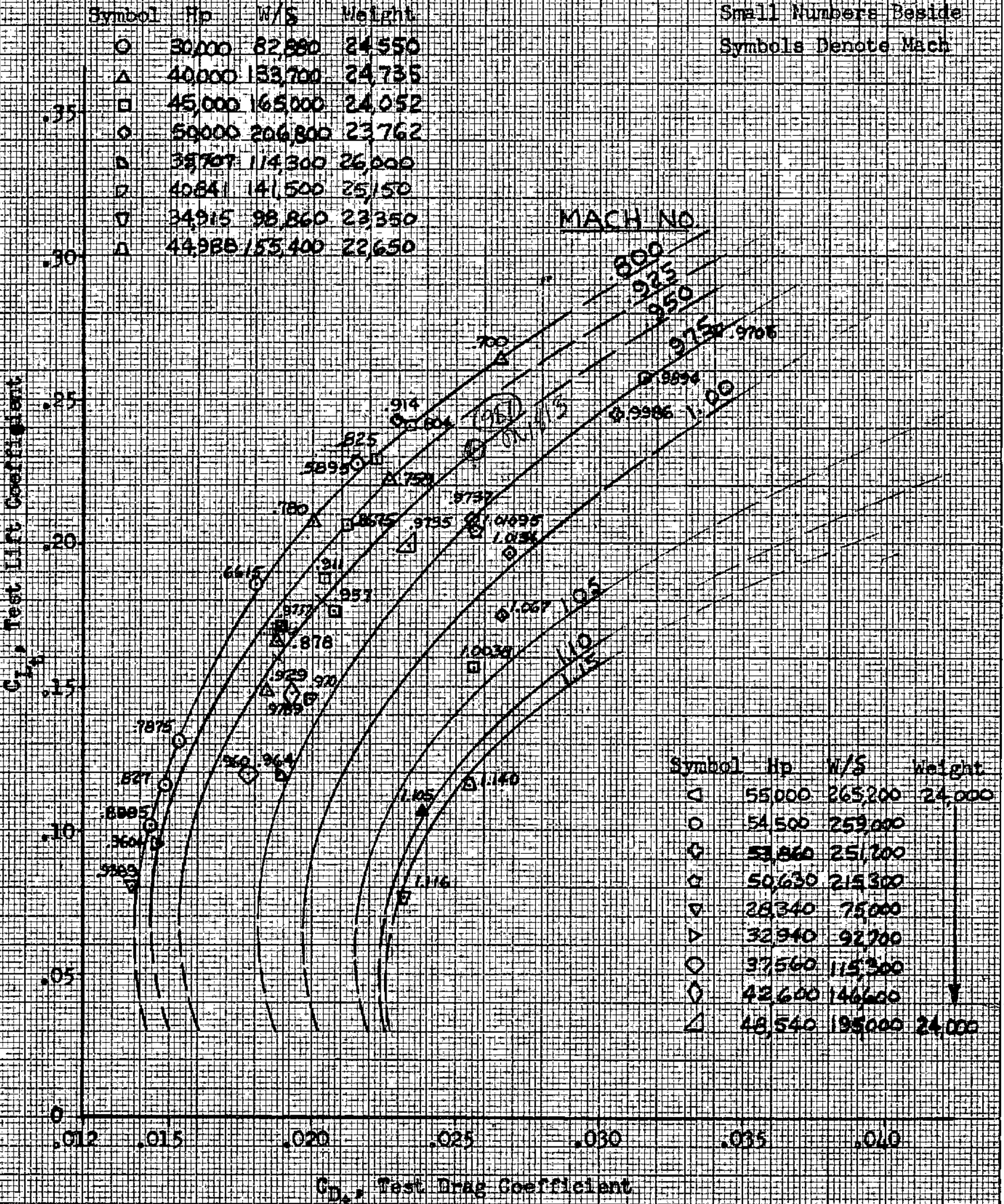
LEVEL FLIGHT PERFORMANCE
Drag vs. V_e
F-102A USAF S/N 56-1317
Engine: J57-P23 S/N P607268
Afterburner: S/N 6170324

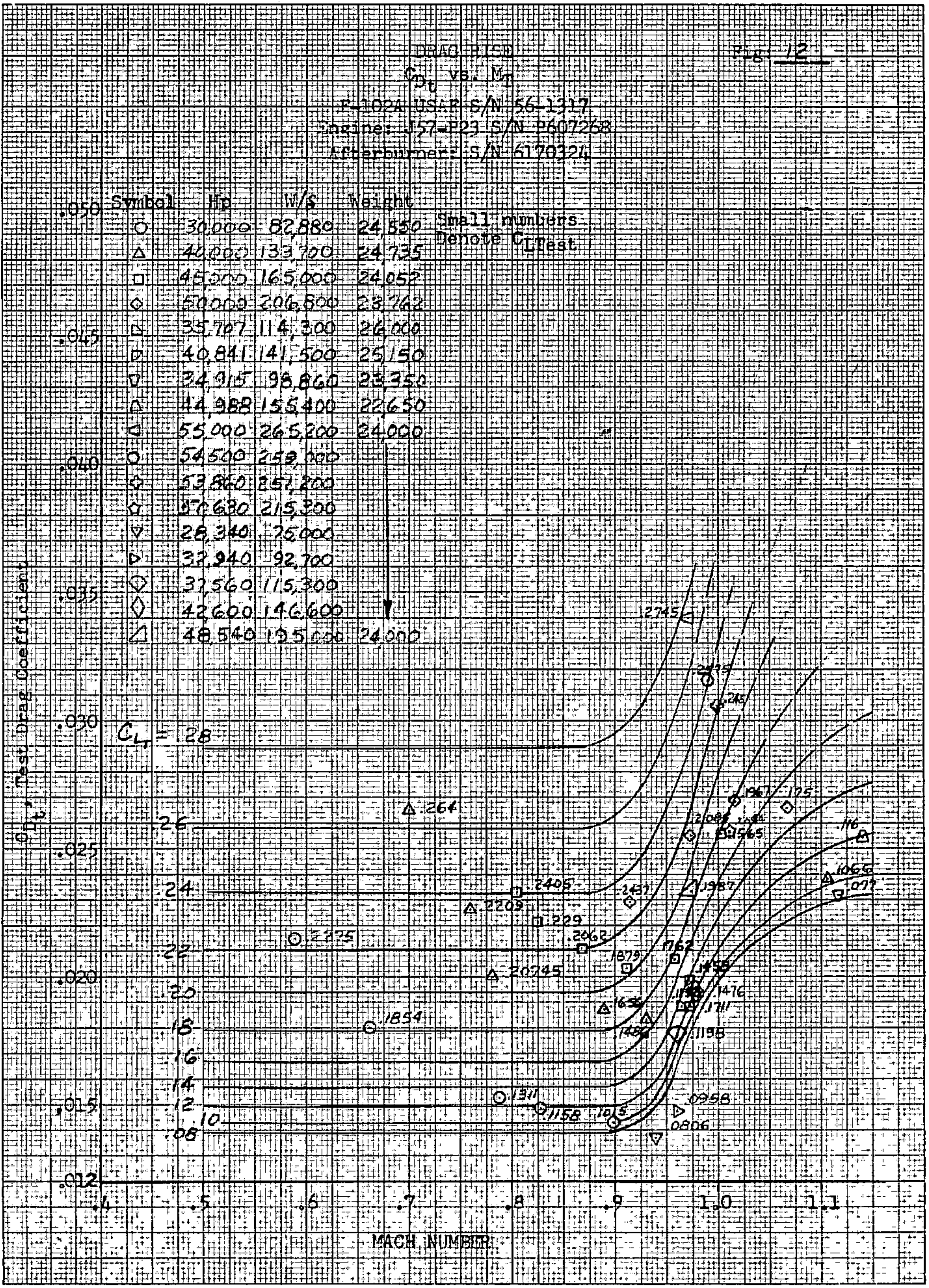
DRAG POLARS

Fig. 11

C_{L_t} vs. C_{D_t}

F-102A USAF S/N 56-1317
Engine: J57-P23 S/N P607268
Afterburner: S/N 6170324





DRAG RISE

C_{D_t} vs. M_T

F-102A USAF S/N 56-1317

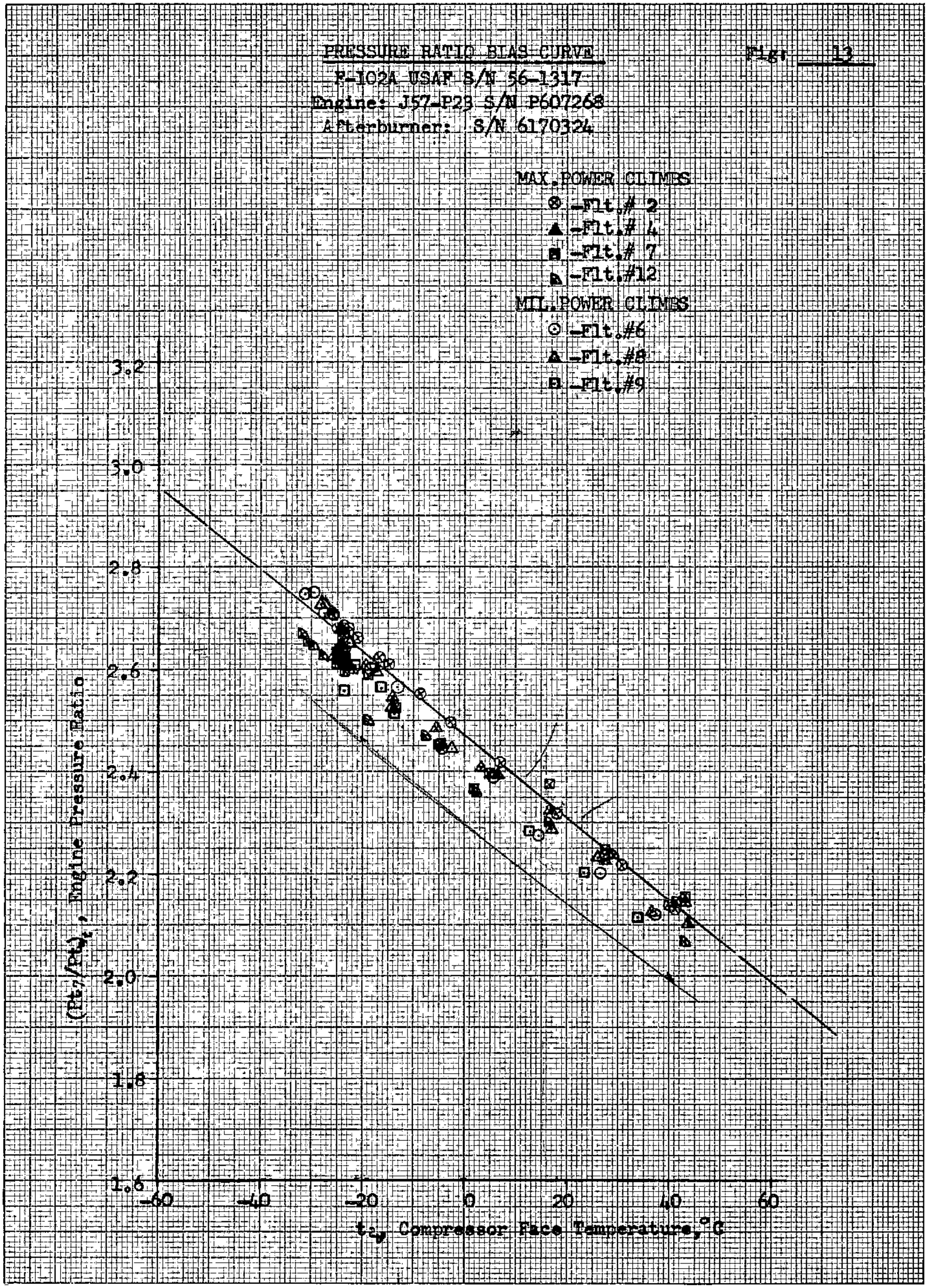
Engine: J57-P23 S/N P607268

Afterburner: S/N 6170324

Fig. 12

Symbol	Hp	W/δ	Weight
○	30,000	82,880	24,550
△	40,000	133,700	24,735
□	45,000	165,000	24,052
◇	50,000	206,800	23,762
◺	35,707	114,300	26,000
◹	40,841	141,500	25,150
▽	34,915	98,860	23,350
⌂	44,988	155,400	22,650
◁	55,000	265,200	24,000
○	54,500	259,000	
✧	53,860	251,200	
☆	47,680	215,300	
▽	28,340	75,000	
▷	32,940	92,700	
⬠	37,560	115,300	
◇	42,600	146,600	↓
◿	48,540	195,000	24,000

Small numbers Denote C_{L_T} Test



PRESSURE RATIO BIAS CURVE

Fig. 13

F-102A USAF S/N 56-1317

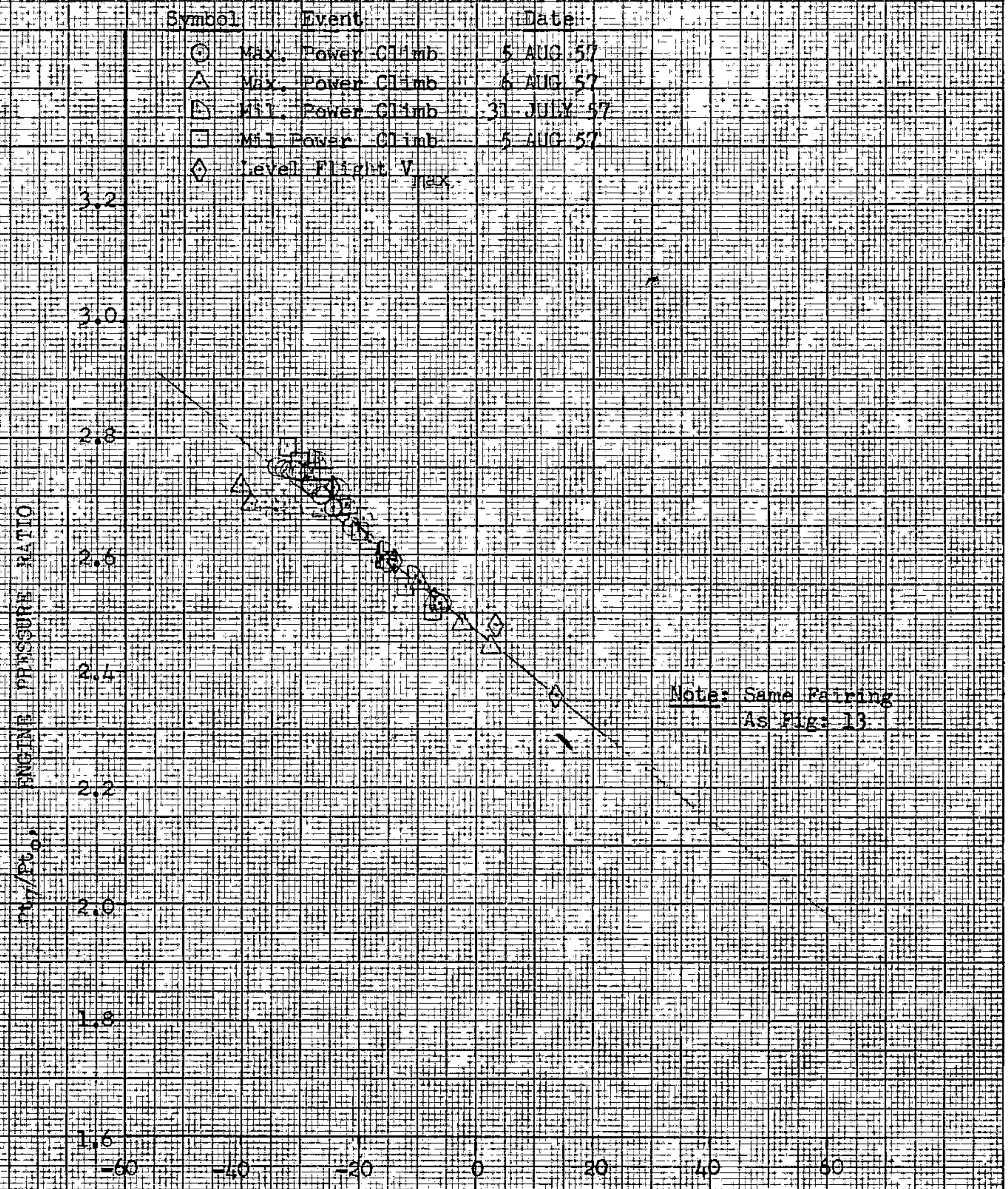
Engine: J57-P23 S/N P607268

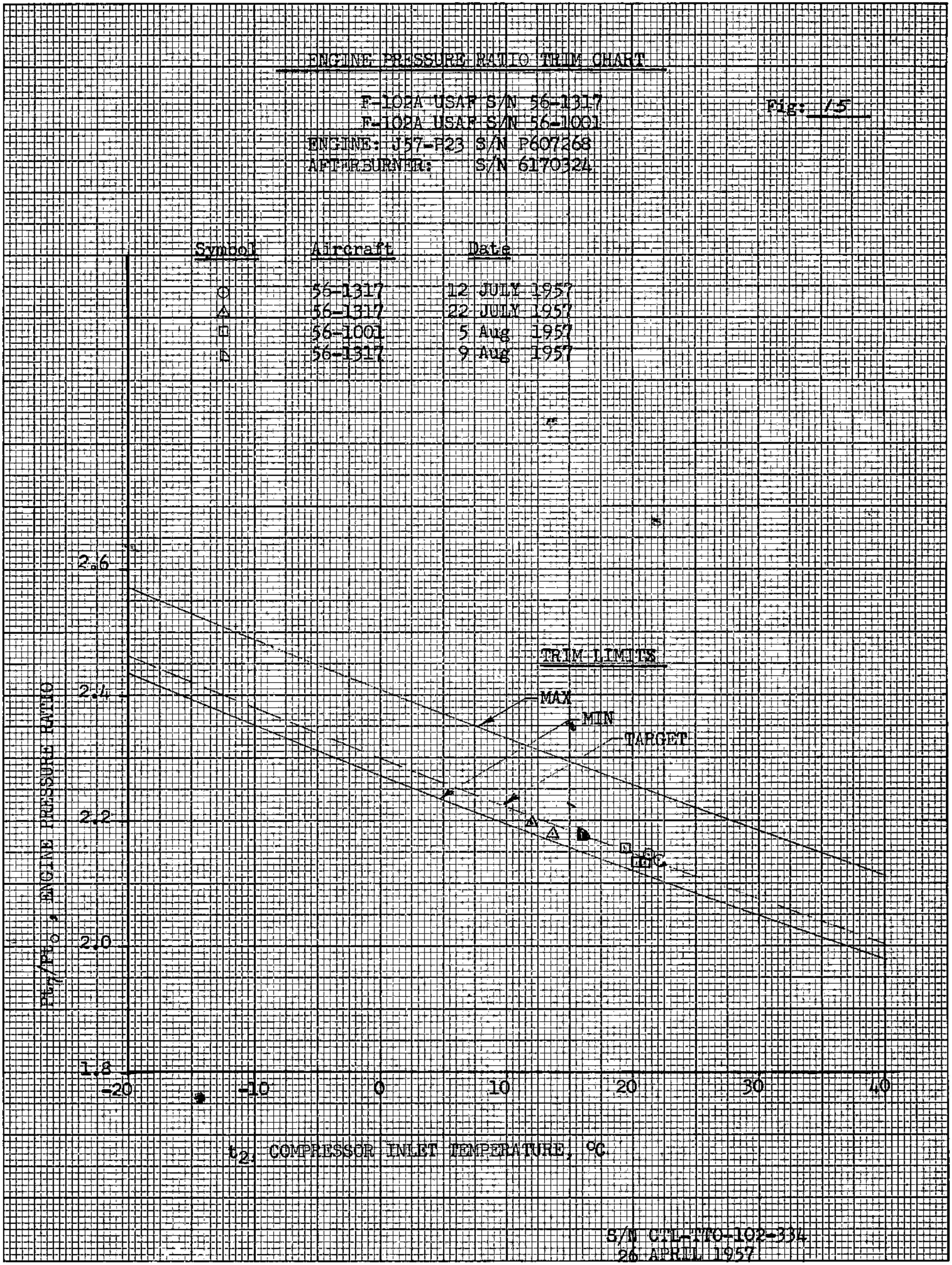
Afterburner: S/N 6170324

ENGINE PRESSURE RATIO BIAS CURVE

FIG: 14

F-102A USAF S/N 56-1001
ENGINE: J57-P23 S/N P607268
AFTERBURNER: S/N 6170324





ENGINE PRESSURE RATIO TRIM CHART

F-102A USAF S/N 56-1317
F-102A USAF S/N 56-1001
ENGINE: J57-P23 S/N P607268
AFTERBURNER: S/N 6170324

Fig: 15

Symbol	Aircraft	Date
○	56-1317	12 JULY 1957
△	56-1317	22 JULY 1957
□	56-1001	5 Aug 1957
◭	56-1317	9 Aug 1957

S/N CTL-TTO-102-334
26 APRIL 1957

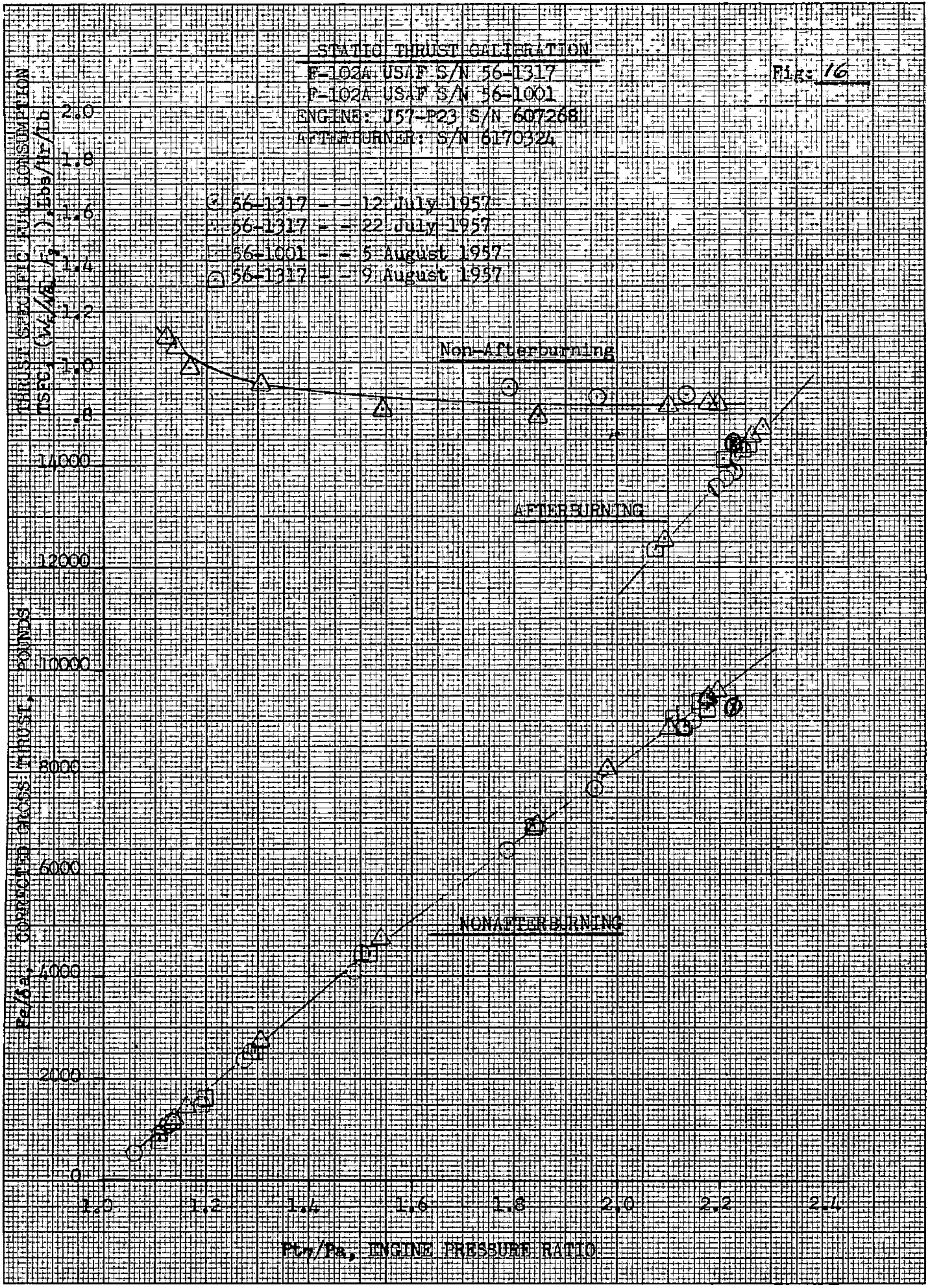


Fig: 16

STATIC THRUST CALIBRATION
F-102A USAF S/N 56-1317
F-102A USAF S/N 56-1001
ENGINE: J57-P23 S/N 607268
AFTERBURNER: S/N 6170324
⊙ 56-1317 - - 12 July 1957
⊡ 56-1317 - - 22 July 1957
△ 56-1001 - - 5 August 1957
⊡ 56-1317 - - 9 August 1957
Non-Afterburning
AFTERBURNING
NONAFTERBURNING
THRUST SPECIFIC FUEL CONSUMPTION TSFC, (Wf/√θa / Fg/δa), Lbs/hr/Lb
2.0
1.8
1.6
1.4
1.2
1.0
.8
Fg/δa, CORRECTED GROSS THRUST, POUNDS
14000
12000
10000
8000
6000
4000
2000
0
1.0
1.2
1.4
1.6
1.8
2.0
2.2
2.4
Pt7/Pa, ENGINE PRESSURE RATIO

AIRSPEED CALIBRATION

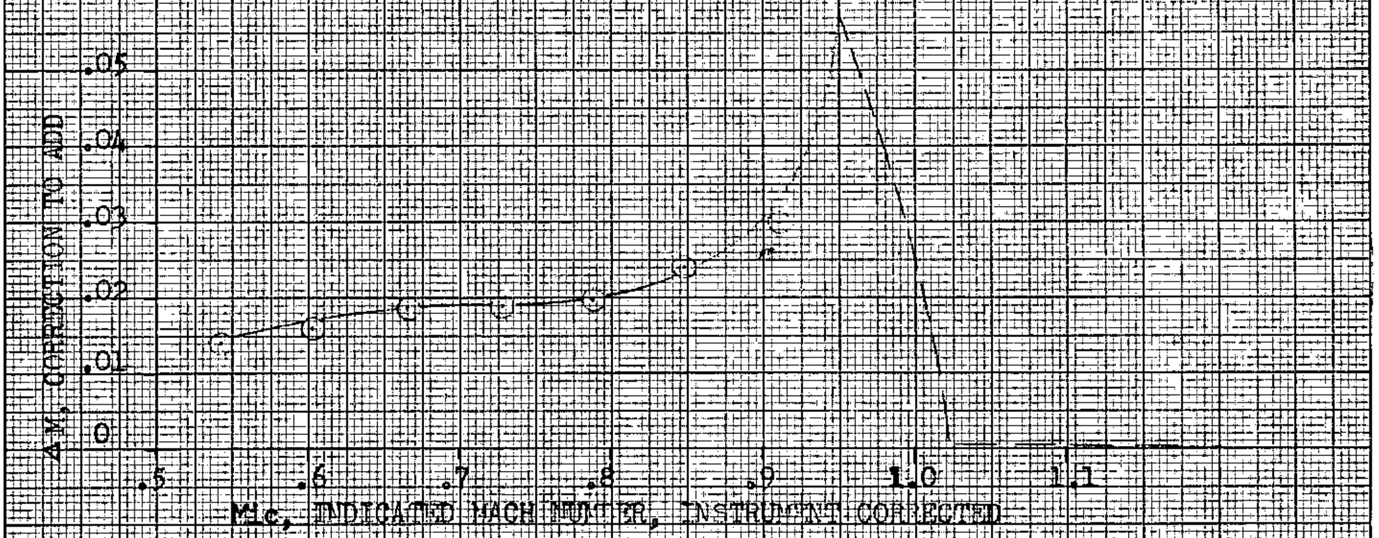
Standard System

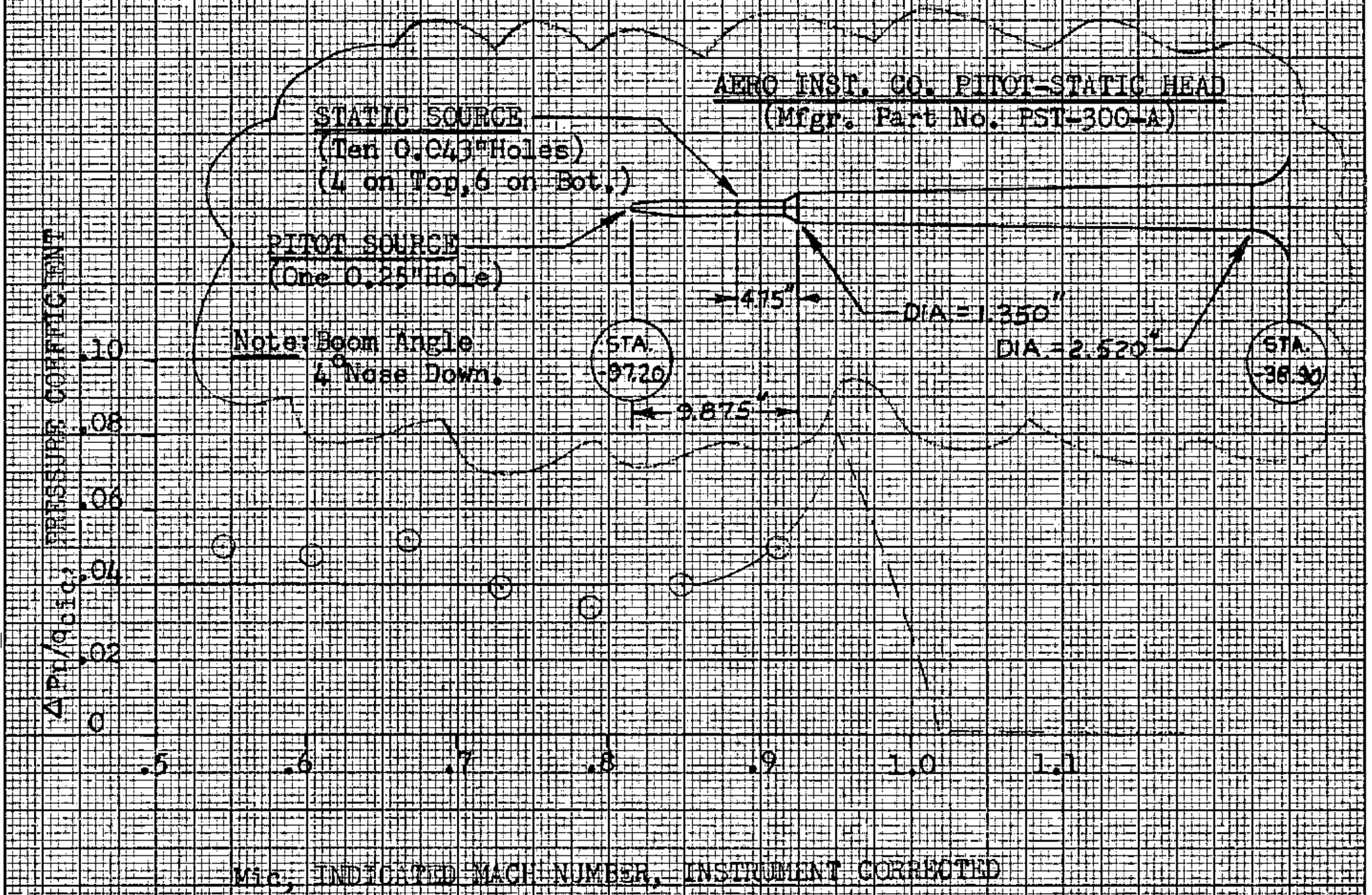
F-102A USAF S/N 56-1317

Fig. 17

Symbol Altitude Method Date

30000 Ft. Pacer 11 July 57





APPENDIX II

aircraft dimensions

	Case XX Wing	Case X Wing
WING:		
Area	695.05 ft.^{2}	695.05 ft.^{2}
Span	38 ft 1.6 in.	38 ft 1.6 in.
MAC	23 ft 9.1 in.	23 ft 9.1 in.
Aspect ratio	2.189	2.189
Incidence	0°	0°
Dihedral	0°	0°
Leading edge sweepback	60°	60°
Trailing edge sweepback	5°	6°
Camber	15% local semispan	6.34% local semispan
ELEVONS:		
Area	70.06 ft.^{2}	67.15 ft.^{2}
Span	14 ft.	12 ft 10.7 in.
Deflection		
symmetrical	+25°, −8°	+25°, −8°
asymmetrical	±7°	±7°
VERTICAL TAIL:		
Area	95.14 ft.^{2}	95.14 ft.^{2}
Length	15 ft 9.2 in.	15 ft 9.2 in.
Height	8 ft 5 in.	8 ft 5 in.
MAC	126.49 in.	126.49 in.
Aspect ratio	1.38	1.38
Airfoil	NACA 0004-65 (modified)	NACA 0004-65 (modified)
Leading edge sweepback	60°	60°
Trailing edge sweepback	0°	0°
RUDDER:		
Area	9.70 ft.^{2}	9.70 ft.^{2}
Deflection	±25°	±25°
FUSELAGE:		
Length	68 ft 3.3 in.	68 ft 3.3 in.

flight limitations

AIRSPEED LIMITATION: 1.5 Mach number or 655 knots IAS

ROLL MENEUVERS:

Altitude—ft.	Airspeed—kts.	Aileron Deflection
Above 20,000	0 to maximum	No restriction
Sea level to 20,000	0 to 450	No restriction
Seal level to 20,000	Above 450	One-half throw

ACCELERATION LIMITATIONS:

Symmetrical	+5.6g to −2.4g +5.6g to −1.4g above 480 knots IAS below 12,000 feet
Asymmetrical	4.0g to 0g
Lateral	144 pounds rapidly applied pedal force

LANDING GEAR EXTENSION SPEED: Maximum allowable speed with landing gear extended is 240 knots IAS

DRAG CHUTE DEPLOYMENT SPEED: 160 knots IAS

RAM AIR TURBINE LIMIT SPEED:

345 knots IAS maximum

125 knots IAS minimum (to maintain adequate emergency hydraulic pressure)

TRIM SERVO: Restricted below 5000 feet above terrain

PITCH DAMPER:

Prohibited at speeds in excess of .95 Mach number below 5000 feet

Do not operate armament gear or bay doors in flight.

ENGINE OPERATION LIMITS:

	EGT — °C Sea Level to 35,000 Feet	EGT — °C Above 35,000 Feet	Time Limit — min. Ground Operation	Time Limit — min. Flight Operation
Maximum	640	670	1	15
Military	630	660	5	30

power plant

The power plant installed in 56-1317 and used also in 56-1001 is a Pratt and Whitney J57-P-23 turbojet engine (S/N P607268) with a flap-type nozzle, and a tandem cylinder, forged flap afterburner. The engine is rated at 16,000 pounds sea level static thrust with afterburner and 9,800 pounds in military power.

weight and balance

In order to simulate combat weight and balance conditions both aircraft were ballasted prior to the program. The gross weight at engine start of 56-1317 (Case XX wing) was 28,110 pounds with a center of gravity location at 29.9 percent MAC. The aircraft weight with fuel expended was 21,112 pounds with a center of gravity location at 23.9 percent MAC. The gross weight at engine start of 56-1001 (Case X wing) was 28,163 pounds with a center of gravity location at 29.2 percent MAC. The weight of this aircraft with fuel expended was 21,104 pounds with a center of gravity of 23.9 percent MAC.

test instrumentation

Data was obtained from the Case XX wing aircraft by means of a photo-panel, telemetering, and precision instruments in the cockpit for pilot call-out as back up. The photo-panel recorded the following parameters required for the flight tests:

Airspeed, altitude, low speed rotor rpm (N_1), high speed rotor rpm (N_2), fuel remaining, correlation counter, time of day, and incremental time.

Engine fuel flow and afterburner fuel flow were telemetered for ground recording.

In addition to these, a sensitive airspeed indicator, sensitive altimeter, and sensitive roll rate indicator were installed in the cockpit for pilot call out for hand recording.

Data was obtained from the Case X wing aircraft by installation of the following calibrated, sensitive instruments into an auxiliary panel in the cockpit: airspeed indicator, altimeter, exhaust gas temperature gage, and pressure ratio gage. In addition, a sensitive normal accelerometer was installed in the standard instrument panel, and a Lewis temperature potentiometer was installed on the right instrument console. All of the above values were read over the radio by the pilot, and hand-recorded on the ground.

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The incorporation of the Case XX wing on an F-102A in place of the Case X wing has produced an increase in maximum power combat ceiling of 4,200 feet, and an increase of .06 Mach number level flight maximum speed at 50,000 feet. Nautical air miles per pound of fuel are decreased 5 percent at low and medium altitudes, but the Case XX wing provides the same range parameters as the Case X wing if a higher cruising altitude is used. The low speed handling characteristics are improved.

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Model YF-102 Serial No. 53-1779

CONVAIR FLIGHT TEST PROGRAM
(Revised)

DYNAMIC STABILITY

Submitted Under
Contract No. AF33(600)-5942

CONVAIR

10

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
SAN DIEGO DIVISION . . SAN DIEGO, CALIFORNIA

DIVISION San Diego
MODEL 8

REPORT ZC-8-108
DATE 6 April 1954
Revised 15 June 1954

TITLE

Model YF-102 Serial No. 53-1779

CONVAIR FLIGHT TEST PROGRAM (Revised)

DYNAMIC STABILITY

Submitted Under
Contract No. AF33(600)-5942

PREPARED BY C. W. Pieper
GROUP EXPERIMENTAL FLIGHT

REVISED BY R. Andrews
REFERENCE ZC-8-001

CHECKED BY
APPROVED BY
for E. D. Shannon
Chief of
Experimental Flight

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INTRODUCTION

This flight test program describes the testing to be conducted by Convair on YF-102 airplane Serial Number 53-1779.

The flight test program will be conducted by Convair, at the Air Force Flight Test Center, Edwards Air Force Base, Edwards, California.

A total of thirty-two (32) flight test hours was originally allotted to the program. However, some of the flight test tasks were transferred to airplane Serial Number 53-1780 in order to insure delivery of 53-1779 to Hughes Aircraft Company by 31 August 1954.

Development testing of the engineering pilot assist system has been transferred in its entirety to 53-1780. Operation of the pitch and yaw damper system will be checked by using nominal value parameters. Full development of the system and determination of optimum parameter values will be accomplished on 53-1780.

Approximately ten (10) flight test hours will be required for this program. The operational test schedule on pages 27 and 28 describes in more detail the utilization of the flight time, and presents a time schedule of events.

This revised program supersedes the original program, Convair Report Number ZC-8-108, dated 6 April 1954. It is authorized by U. S. Air Force Contract AF33(600)-5942, Supplemental Agreement No. 3.

DESCRIPTION AND CONFIGURATIONS

Airplane General

The F-102 airplane is a single place, land based, high altitude, all weather interceptor. The primary mission of the ultimate production airplane is the interception and destruction of attacking enemy aircraft and airborne missiles which fall within the performance capabilities of the aircraft, its armament and associated electronics equipment under all conditions of weather and visibility. Its offensive armament will include combinations of 2.75 inch folding fin aircraft rockets and a maximum of six Falcon GAR type missiles. Guidance and release of this armament will be controlled by the Hughes MX-1179 fire control system in the ultimate production models. The ultimate production version with MX-1179 will be capable of ground controlled interception (GCI) type operations.

The distinguishing physical characteristics of the F-102 airplane are its 60° delta wing platform, half delta vertical fin without horizontal tail surfaces, and the fuselage cheek type engine air inlet ducts.

The irreversible fully hydrallic powered control system has only three movable control surfaces. These consist of an elevon on the trailing edge of each wing and a rudder on the trailing edge of the vertical fin. No wing flaps are used.

DESCRIPTION & CONFIGURATION
(Continued)

YF-102 Airplane S/N 53-1779

The YF-102 airplane S/N 53-1779 for which this flight test program has been compiled, is the third YF-102 type airplane to be constructed. It is a Convair 8-82 version airplane, and is powered by a Pratt & Whitney J57-P-11 turbojet engine with afterburner. Take-off gross weight is approximately 29,000 pounds.

A more complete description of the airplane including operational procedures and limitations is presented in Reference (A).

The test configuration of the subject airplane differs from the contractual airplane specification due to certain deletions and modifications necessary to provide for installation of flight test instrumentation. Major items of difference are as follows:

1. The MG-3 fire control system is not installed.
2. The plastic radome is replaced by a metal nose cap. However, the plastic radome will be installed prior to delivery to HAC.
3. The production nose boom is replaced by a special test nose boom. The production nose boom will be installed and calibrated prior to delivery to HAC.
4. The missile launching equipment will not be installed and the missile launching system will be inoperative.
5. The pilot's armament control panel in the cockpit will not be installed. This will be replaced by a pilot's flight safety panel. A special cockpit switch is provided for opening and closing the armament bay doors.

DATA MEASUREMENTS

A complete list of all data measurements to be recorded during the accomplishment of each flight test item is presented on pages 29 thru 32.

A complete description of the instrumentation system and associated special equipment necessary to record the data is presented in Convair Report Number ZC-8-053, Test Instrumentation Report, Airplane Serial Number 53-1779, dated 15 February 1954.

TEST OBJECTIVES

The flight test task of the subject airplane is given by Exhibit "A" of Convair CCN 32, MSN94352, dated 17 November 1953. The assignement is stated in Exhibit "A" as follows:

"Airplane Serial No. 53-1779

To be utilized for development of the prototype MG-3 fire control system. This will consist of manual operation of MG-3 computer radar sub-systems. System and aircraft tie-in."

Hughes Aircraft Company has been designated as the primary test agency for this airplane. Convair will flight test the airplane during the four months period prior to the start of the HAC program.

The basic objectives of the Convair program are as follows:

1. To establish the basic flying qualities of the airplane and to determine the dynamic response of the airplane to specified control surface deflections.
2. To determine the dynamic response of the flight control system to specified control surface deflections while using predetermined nominal parameter values for the electronic pitch and yaw damping system.
3. To perform required flight calibrations of instrumentation equipment.

The flight test items which will be accomplished during this program are presented on pages 8 thru 26. These tests are in the nature of Phase I tests in accordance with AFR 80-14 of 11 September 1951. Actual flights will be scheduled to accomplish as many items simultaneously as possible for any given configuration and flight condition.

FLIGHT RESTRICTIONS

Temporary operating restrictions will be imposed on this airplane until deleted by results of tests, design studies or agreement. These restrictions will be imposed by Convair Engineering or the USAF. The temporary restrictions will be furnished to the operating agencies by Convair Engineering.

The current operating restrictions for this airplane in effect on the date of publication of this report are given below:

The airplane is not to be operated in excess of current operating limits or restrictions at any time during the flight test program.

Temporary Operating Restrictions for YF-102 Airplane S/N 53-1779

1. Maximum Speeds:
   - a. Clean configuration - 810 MPH IAS or a stagnation temperature of 230° F. However, the subject airplane shall not be flown at speeds in excess of those flown by YF-102 S/N 52-7995.
   - b. Gear Down - 276 MPH IAS.
   - c. Drogue Chute Deployment - 150 MPH IAS (ground operation only).
2. Maximum Load Factors:
   - a. Normal, Symmetrical: +3.55, -2.1 g at a gross weight of 26,000 lbs. Use inverse weight ratio for gross weight corrections.
   - b. Normal, Asymmetrical: 2.7g, -1.0g at a gross weight of 26,000 lbs.
   - c. Lateral: 0.95g

FLIGHT RESTRICTIONS (Continued)

3. Roll Maneuvers:

a. Yaw Damper Operation with Optimum Gain Factors:

(1) Aileron angle is restricted to ±6° for all speeds above M = 0.7

(2) Aileron angle is restricted to ±4° for all speeds above 720 MPH IAS at pressure altitudes less than 12,000 feet.

b. Yaw Damper Not Operative or Yaw Damper Operative with Other than Optimum Gain Factors:

(1) Aileron angle is restricted to ±4° for continuous roll maneuvers above M = 0.7

(2) Aileron angle is restricted to ±6° for changes in angle of bank of 60° or less for speeds above M = 0.7 at pressure altitudes above 12,000 feet.

(3) Aileron angle is restricted to ±4° for speeds above 720 MPH IAS at pressure altitudes less than 12,000 feet.

FLIGHT TEST ITEMS

Various flight tests have been planned to fulfill the objectives of the program. Flight Test Items 1-9, which are listed below, state the type of test, purpose, requirements, and procedure for the various tests to be accomplished.

These flight test items are numbered in the approximate order in which they will be accomplished during the program. The flight test items are not flight plans for individual flights.

The data measurements required to evaluate these flight test items are tabulated on pages 29 thru 32.

Flight Test Item	Description	Page No.
1	Airspeed Calibration - Subsonic Speed Range	10, 11
2	Airspeed Calibration - Transonic & Supersonic Speed Range	12 - 15
3	Angle of Attack & Angle of Yaw Pickup Calibration	16, 17
4	Dynamic Lateral - Directional Stability Without Damping	18
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7	Dynamic Longitudinal Stability With Damping	22
8	Dynamic Response to Speed Brake Operation	23 - 25
9	Dynamic Response to Armament Bay Door Operation	26

SYMBOLS AND CONFIGURATIONS

The following symbols and configurations are used throughout the remainder of this report to specify configurations or flight conditions.

Configuration:

P	Clean - gear up, speed brakes in unless otherwise specified, power required for level flight at the specified trim speeds.
PA	Gear Down - Power required for level flight at the specified trim speed.

Symbols:

V_{max}	Level flight speed at the specified altitude with maximum available thrust (full afterburner operation).
MPH (IAS)	Indicated airspeed in miles per hour, corrected for instrument and position error.
M	Mach Number
PA	Pressure Altitude

FLIGHT TEST ITEM No. 1

Test: Airspeed Calibration - Subsonic Speed Range CVAC Index No. 2.1

Purpose:

To determine static source position error for each of the two nose boom configurations.

Requirements:

Determine static source position error for each of the following nose boom configurations:

1. Test nose boom used with the metal nose cap.
2. Production type nose boom used with the plastic radome.

Requirements according to specification MIL-I-5072, Paragraph 4.3.3.4.3 are applicable to the production nose boom airspeed calibration in the speed range below M = 0.85. This requirement is stated as follows:

> "Airspeed and Altitude Indication Errors
>
> The airspeed and altitude indication errors due to the location of the pitot-static tube, corrected to standard sea level conditions, shall not exceed the limits specified in Table I when tested in flight."

The following configurations listed in Table I are applicable in the speed range below M = 0.85. Tolerances listed are corrected to standard sea level conditions 15°C, 29.92" Hg.

Configuration	Speed Range	Gross Weight	TOLERANCES A.S. Ind.	 Altimeter
Approach (gear down)	135-188 MPH (IAS)	Landing 23,000#	4.5 MPH	25'/115 MPH (IAS)
Approach (gear down)	143-200 MPH (IAS)	Normal 26,000#	4.5 MPH	25'/115 MPH (IAS)
Clean	143 MPH (IAS) to M 0.85	Normal 26,000#	4.5 MPH	25'/115 MPH (IAS)

FLIGHT TEST ITEM No. 1
(Continued)

Procedure

The test nose boom will be used during the Convair flight test program, and will contain angle of attack and angle of yaw instrumentation in addition to airspeed instrumentation. The airspeed calibration for the test nose boom will be conducted during the early stages of the Convair flight test program.

The production type nose boom and plastic radome will be installed on the airplane prior to delivery to Hughes Aircraft Company. This nose boom contains airspeed instrumentation only, and will require a position error calibration. This calibration will be conducted during the latter stages of the Convair flight test program.

1. Conduct a pacer method calibration using a calibrated AFFTC F-86 pacer airplane.
2. Conduct tests at 15,000' P.A. for the following configurations and speed ranges.

Configuration	Speed Range (IAS)	Remarks
P	150 - 400 MPH (IAS)	20 MPH increments
PA	150 - Placard speed for gear extension	10 MPH increments

FLIGHT TEST ITEM No. 2

CVAC Index No. 2.1

Test: Airspeed Calibration - Transonic and Supersonic Flight Range

Purpose

To determine static source position error for each of the two nose boom configurations.

Requirements

Determine static source position error for the following nose boom configurations:

1. Test nose boom used with the metal nose cap.
    a. Calibrate for airspeed range to be flown during the Convair flight test program.
2. Production type nose boom used with the plastic radome.
    a. Calibrate for airspeed range required by HAC flight test program.

Procedure

The test nose boom will be used during the CVAC flight test program and the following calibration tests will be conducted concurrently with other test items.

The production type nose boom will be installed prior to Hughes delivery and will be calibrated during the latter part of the CVAC flight test program.

The method which will be used for calibration of the airspeed system in the transonic and supersonic flight regimes is a statictical method which provides for obtaining the value of the airspeed system error. The data from several flights may be combined into a single solution which is obtained by the theory of least squares.

The airplane's tapeline altitude must be accurately determined with radar to obtain data for this method. Since data are desired from a wide range of altitudes,

FLIGHT TEST ITEM No. 2
(Continued)

Procedure

airspeed calibration data may be obtained during the climb to altitude when the purpose of the flight is to perform tests of a different character at high altitudes.

Physical data required for this method are as follows:

1. An accurate survey of the static pressure and temperature versus altitude of the atmosphere in which the test flights are to be made. These data will be obtained by radiosonde atmospheric survey and should be obtained over several successive days during the test period.

2. Indicated static and dynamic pressure of the test airplane's airspeed system. These data must be obtained during a subsonic, atmospheric surveying climb to altitude and a subsequent high speed dive to low altitude. These flights must be accomplished on at least three different days to obtain ample data for atmospheric survey.

   The tapeline altitude of the test airplane must be accurately determined by radar tracking during these flights. The actual flight test procedures will be as follows:

   (1) Airplane configuration - P (clean)

   (2) Request special radiosonde data at approximately the time of the flight.

FLIGHT TEST ITEM No. 2
(Continued)

Procedure

(3) Track the entire flight of the airplane or as much of it as possible with the radar available at EAFB using 2.0 second intervals between data points.

(4) Obtain photo panel data at the rate of one frame every two seconds during the entire flight. Higher frequency photo panel data permitted on any portion of the flight.

(5) Climb at NRT to 37,000 feet P.A. Maintain indicated speed over the range of M = .6 to M = .8, but do not exceed M = 0.85. Maintain flight pattern during climb as shown in sketch, page 15 Stay within 15 miles distance of radar station during climb, and fly a pattern which keeps the airplane opposite the sun from the radar tracking station as much as possible.

(6) Make high speed runs along highway U.S. 466 as shown in sketch, page 15 .

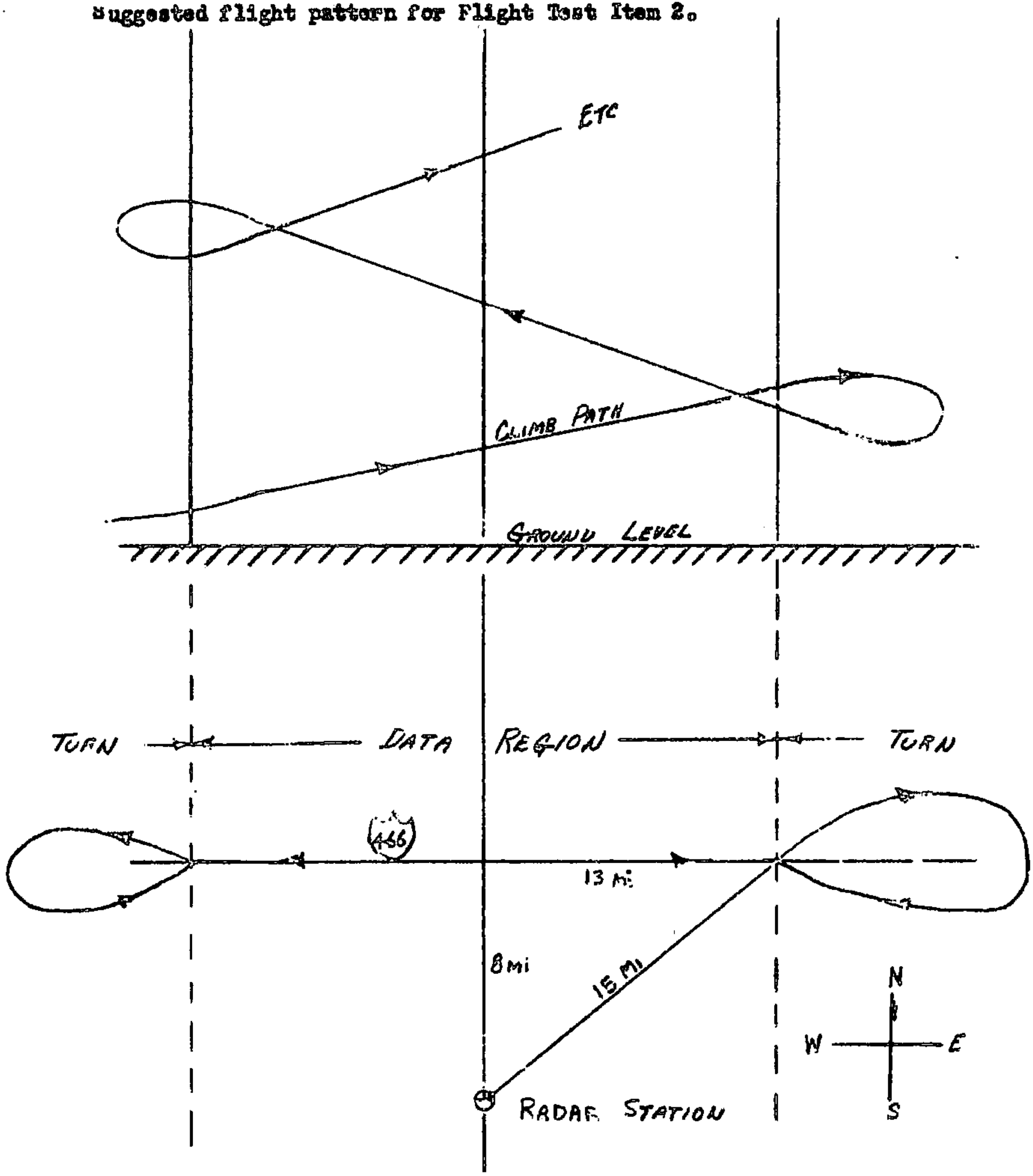
(7) Descent may be made at any speed, but airplane should be kept within a 15 mile radius and out of the sun quadrant as much as possible.

(8) These tests are to be performed on at least three different flights.

(9) The effects of load factor on the airspeed position error will be investigated by accomplishing above procedure under accelerated flight conditions.

FLIGHT TEST ITEM No. 2
(Continued)

Suggested flight pattern for Flight Test Item 2.



FLIGHT TEST ITEM No. 3

CVAC Index No. 1.7

Test: Angle of Attack and Angle of Yaw Pickup Calibration

Purpose

To determine the position error of the test Specialties (ADD) angle of attack and angle of yaw pickups mounted on the fuselage.

The fuselage ADD pick-ups will be necessary for determination of angle of attack and angle of yaw when the plastic nose radome and production nose boom are installed on the airplane.

These pickups are for test purposes only and will be replaced by the production servo mechanisms vane type indicators at a later date when these indicators become available and when the optimum location for them has been determined.

Requirements

Angle of attack measurements are desired to an accuracy of .25 degree.

The accuracy of the angle of yaw measurements is not stated. However, the error should be small for small angles of yaw (0-5°).

Procedure

Fuselage Specialties airstream directional detector (ADD) pickups will be calibrated with vane type angle of attack and angle of yaw indicators mounted on the test nose boom.

The ADD angle of attack pickup is located on the left hand side of the fuselage at Station 89. The ADD angle of yaw pickup is located at the bottom centerline of the fuselage at Station 89.8.

FLIGHT TEST ITEM NO. 3
(Continued)

Correction constants for the nose boom angle of attack indicators will be determined during unaccelerated level flight by means of a recording deck angle inclinometer. This will be accomplished by recording deck angle over the range of test airspeeds while the airplane is in stabilized level flight. These corrections will be applied to the nose boom angle of attack data for all other accelerated, manouvering flight conditions during ADD calibration.

The ADD yaw indicator data will be compared directly to the nose boom yaw vane data for all flight conditions during ADD calibration.

Except for the level flight checks of the nose boom angle of attack indicators, no special flight tests are required for the ADD pickup calibrations. The calibration will be determined by statistical analysis of nose boom data taken for various flight conditions which will be flown during the program. The calibration should cover the speed, altitude, and load factor ranges which will be flown during the HAC flight test program.

FLIGHT TEST ITEM NO. 4

CVAC Index No. 1.2B

Test: Dynamic Lateral-Directional Stability Without Damping.

Purpose:

(1) To determine the airplane's inherent damping characteristics of rudder induced lateral-directional oscillations.

(2) To determine control system response to pulsed rudder input signals while flying without damping functions.

Requirements

(1) Determine airplane response to abrupt rudder deflections without damping functions for the given conditions.

Procedure

(1) The CVAC flight test programmer will be used to produce pulsed rudder input signals to the rudder servo amplifier. This signal will cause a rudder deflection of predetermined rate, amplitude, and duration. The required data will be recorded during the control surface deflection and during the following lateral-directional oscillations.

(2) Trim the airplane in stabilized level flight in Manual Flight Mode for the configurations, speeds, and altitudes specified for runs 1 - 14 tabulated on page 33. The pitch damping system must be inoperative during this test.

(3) Operate the programmer as previously described to produce the desired rudder deflection, and to operate the data recording systems.

FLIGHT TEST ITEM NO. 5

CVAC Index No. 1.2B

Test: Dynamic Lateral - Directional Stability With Damping.

Purpose:

(1) To demonstrate damping of rudder induced lateral-directional oscillations with nominal pitch and yaw damper parameter values.

(2) To demonstrate control system response to pulsed rudder input signals with nominal pitch and yaw damper parameters.

Requirements:

(1) Determine airplane response to abrupt rudder deflections with damping circuits in operation.

Procedure

(1) The CVAC flight test programmer will be used to produce pulsed input signals to the rudder servo amplifier. A predetermined nominal yaw rate gain factor, yaw rate time constant, and yaw rate gyro tilt angle will be fixed into the yaw damper network for the given flight conditions.

(2) Trim airplane in stabilized level flight in manual flight mode for the configurations, speeds, and altitudes specified for runs 29 - 34 tabulated on page 33.

(3) Operate programmer as previously described to produce the desired rudder deflection and to operate the recording system.

FLIGHT TEST ITEM NO. 6

CVAC Index No. 1.1C

Test: Dynamic Longitudinal Stability Without Damping.

Purpose

(1) To determine the airplane's inherent damping characteristics of elevator induced longitudinal oscillations.

(2) To determine control system response to pulsed elevator input signals while flying without damping functions.

Requirements

(1) Determine airplane response to abrupt elevator deflections without damping functions for the given conditions.

Procedure

(1) The CVAC Flight test programmer will be used to produce pulsed elevator input signals to the elevon servo amplifiers. The input signal will cause an elevator deflection of predetermined rate, amplitude and duration.

(2) Trim airplane in stabilized level flight in the Manual Flight Mode for the configurations, speeds, and altitudes specified for Runs 15 - 28 tabulated on page 33.

The pulse and input elevator deflections during initial tests will be limited by mechanical stops in the elevon servo actuators to ±.75 degrees. After a preliminary evaluation of airplane response the stops will be removed and the deflections indicated for the runs above will be possible. All pitch and yaw damper gain factors on the parameter control box will be set to zero value for this test.

FLIGHT TEST ITEM No. 6
(Continued)

(3. Operate the programmer system as previously described to produce the desired elevator deflection and to operate the data recording system.

(4) Retrim airplane as necessary to return to original trim condition.

FLIGHT TEST ITEM NO. 7

CVAC Index No. 1.1C

Test: Dynamic Longitudinal Stability With Damping

Purpose

(1) To demonstrate damping of elevator induced oscillations with nominal pitch damper parameter values.

(2) To demonstrate control system response to pulsed elevator input signals during pitch damper operation with nominal parameter settings.

Requirements

(1) Determine airplane response to abrupt elevator deflections with damping circuits in operation.

Procedure

(1) The CVAC flight test programmer will be used to produce pulsed elevator input signals to the elevon servo amplifiers. This signal will cause an elevator deflection of predetermined rate, amplitude and duration. A predetermined nominal pitch gain factor will be fixed into the pitch damper network for the given flight conditions.

(2) Trim the airplane in stabilized level flight in Manual Flight Mode for runs 35 - 40 tabulated on page 33. The pulse input elevator deflections during initial tests will be limited by mechanical stops in the elevon servo actuators to ±.75 degree. After a preliminary evaluation of airplane response, the stops will be removed and the deflections indicated for the runs above will be possible.

(3) Operate programmer system as previously described to produce the desired elevator deflection and to operate the data recording system.

FLIGHT TEST ITEM NO. 8

CVAC Index No. 1.7

Test: Dynamic Response to Speed Brake Operation.

Purpose

(1) To determine the effect of speed brake extension and retraction while flying with constant power settings.

(2) To determine the static and dynamic stability characteristics of the airplane with speed brakes extended.

Requirements:

(1) Determine the effects of extending and retracting speed brakes while flying in stabilized level flight for the conditions specified below.

(2) Determine the following characteristics when flying in stabilized level flight with speed brakes extended.

- (a) Static longitudinal stability
- (b) Dynamic longitudinal stability
- (c) Static lateral-directional stability
- (d) Dynamic lateral-directional stability

Procedure

Conduct tests in Manual Flight Mode using nominal damping parameters.

(1) Trim airplane in stabilized level flight conditions specified below.

Configuration	Trim Speed	Altitude	Speed Range
P	330 MPH (IAS) (M = .9)	35,000' P.A.	330 - 220 MPH (IAS) (M = .9 to M = .6)
P	285 MPH (IAS) (M = .5)	15,000' P.A.	320 - 185 MPH (IAS) M = .56 to M = .32)

FLIGHT TEST ITEM NO. 8
(Continued)

Procedure

Operate speed brake control and extend brakes full out. Operate programmer during deceleration period to produce pulsed rudder deflections. Maintain constant altitude during deceleration period. When lower limit of speed range is reached, retract speed brakes. Record data during entire operation outlined above.

(2) Trim airplane in stabilized level flight with speed brakes extended with NRT for the flight conditions specified below.

(a) Without re-trimming controls, change power setting and stick force as required to maintain stabilized flight at the specified speed increments. Record data at each trim speed and at each stabilized airspeed increment. Maintain the test altitude to ±2000 feet. If the altitude margin cannot be maintained, it will be necessary to break the test up into several runs so that the test altitude can be held more closely.

(b) Trim airplane as for (2) above at 293 MPH IAS (Mach .8) and 35,000' P.A. Perform steady wings level sideslips to the right and left in increments of 2° up to a maximum of 8° sideslip. Stabilize sideslip at each increment. Record data for the trim speed and at each increment of sideslip angle. Repeat the test at 370 MPH IAS (Mach .64) and 15,000' P.A.

(c) Trim airplane as for (2) above, and determine dynamic longitudinal stability by the procedure outlined in Flight Test Item 6, page 20. Operate programmer system to produce pulsed elevator input signals.

FLIGHT TEST ITEM NO. 8
(Continued)

(d) Trim airplane as for (2) above, and determine dynamic lateral-directional stability by the procedure outlined in Flight Test Item 4, page 18. Operate programmer system to produce pulsed rudder input signals.

Test Conditions

Configuration	Trim Speed	Altitude	Speed Range
P	293 MPH (IAS) (M= .80)	35,000' P.A.	230 to 305 MPH (IAS) (M = .64 to M = .83) 20 MPH Increments
P	370 MPH (IAS) (M = .64)	15,000' P.A.	275 - 460 MPH (IAS) (M = .48 to M = .80) 20 MPH Increments

FLIGHT TEST ITEM NO. 9

CVAC Index No. 1.7

Test: Dynamic Response to Armament Bay Door Operation.

Purpose

(1) To determine the transitory response effects caused by armament bay door operation.

Procedure

(1) Conduct tests in the Manual Flight Mode using nominal damping parameters.

(2) Trim airplane in stabilized level flight for the flight conditions specified below:

Configuration	Trim Speed	Altitude
P	330 MPH (IAS) (M = .9)	35,000' P.A.
P	415 MPH (IAS) (M = .8)	20,000' P.A.

(3) Allow airplane to fly "hands off" controls and operate armament bay door control switch to open doors. Hold doors open five seconds and then close. Record data during stabilized period and during period of door operation and subsequent airplane behavior. If possible, maintain "Hands Off" flight until airplane restabilizes at original conditions.

(4) Repeat above test holding doors open four seconds, three seconds, and two seconds intervals for the same altitudes and trim speeds.

OPERATIONAL TEST PHASES

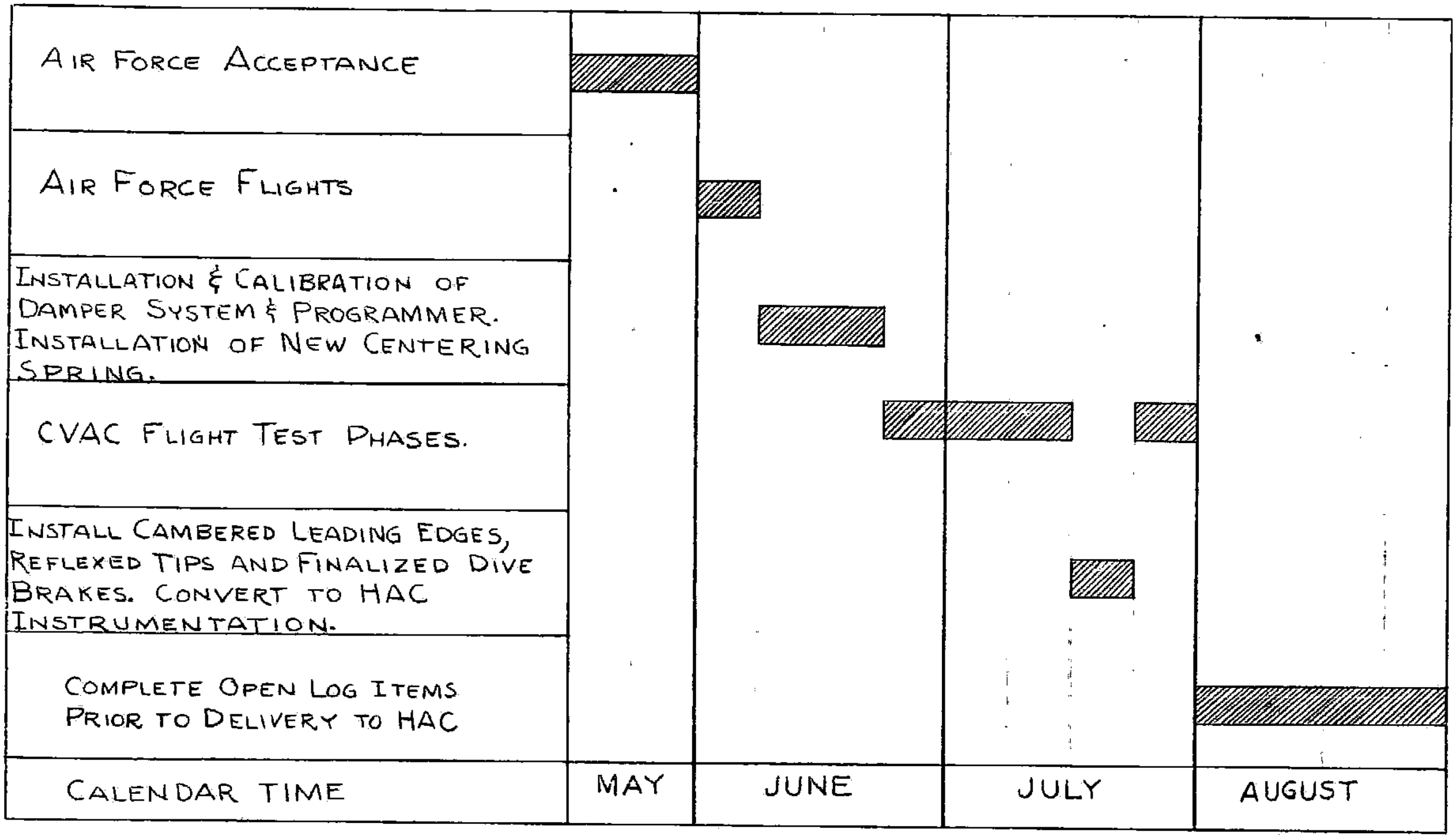
The flight test program will be accomplished in operational test phases listed below. The calendar period during which these phases will be conducted is shown on page 28.

Phase A must be conducted and completed prior to Phase B. Phase C flights will be conducted during convenient periods of the flight test period.

Phase	Flight Test Items	Flight Hours Required
A	4, 5, 6, and 7	4
B	8 and 9	3
C	1, 2, and 3	3
	TOTAL HOURS	10

REVISED 15 JUNE 1954

SCHEDULE OF EVENTS



DATA MEASUREMENTS REQUIRED FOR FLIGHT TEST ITEMS

Item Number	MEASUREMENT	FLIGHT TEST ITEMS								
		Airspeed Calib. Subsonic Range	Airspeed Calib. Trans-Supersonic	Angles Attack-Yaw Pickup Calib.	Dyn. Lat-Direc. Stab. No Damping	Dyn. Lat-Direc. Stab. With Damp.	Dyn. Long. Stab. Without Damping	Dyn. Long. Stab. With Damping	Dyn. Resp. Speed Brake Operation	Dyn. Resp. Bay Door Operation
		1	2	3	4	5	6	7	8	9
	FLIGHT ATTITUDE GROUP									
1	Mach Number	X	X	S	S	S	S	S	S	S
2	Indicated Airspeed	X	X	X	X	X	X	X	X	X
3	Pressure Altitude	X	X	X	X	X	X	X	X	X
4	Angle of Attack - Fwd. Vane (R) On Nose Boom	X	X	X	X		X	X	X	X
5	Angle of Attack - Aft Vane (L) On Nose Boom			X						
6	Angle of Attack - Left Side at Sta. 89			X						
7	Angle of Yaw - Btm Vane on Nose Boom	X	X	X	X	X			X	X
8	Angle of Yaw - Btm CL at Sta. 89.8			X						
9	Pitch Attitude - CG						X	X		
10	Roll Attitude - CG				X	X				
11	Directional Heading - CG				X	X			X	X
12	Pitch Rate - CG					X	X	X	X	X
13	Roll Rate - CG				X	X			X	X
14	Yaw Rate - CG				X	X			X	X
15	Normal Acceleration - Pilot's Panel	S	S	S	S	S	X	X	S	S
16	Normal Acceleration - Nose Wheel Well						X	X		
17	Normal Acceleration - CG				X	X	X	X	X	X
18	Lateral Acceleration - CG				X	X			X	X
19	Longitudinal Acceleration - CG						X	X	X	

S - Suggested for Flight Safety or Supplementary Information
X - Required Measurement

CONFIDENTIAL

DATA MEASUREMENTS REQUIRED FOR FLIGHT TEST ITEMS

Item Number	MEASUREMENT	FLIGHT TEST ITEMS								
		Airspeed Calib. Subsonic Range 1	Airspeed Calib. Trans-Supersonic 2	Angles Attack-Yaw Pickup Calib 3	Dyn. Lat-Direc. Stab. No Damping 4	Dyn. Lat-Direc. Stab. With Damp. 5	Dyn. Long. Stab. Without Damping 6	Dyn. Long. Stab. With Damping 7	Dyn. Resp. Speed Brake Operation 8	Dyn. Resp. Bay Door Operation 9
	FLIGHT CONTROL GROUP									
1	Longitudinal Stick Force						X	X	S	S
2	Lateral Stick Force				X	X			S	S
3	Rudder Pedal Force				X	X			S	S
4	Longitudinal Stick Position						X	X	S	S
5	Lateral Stick Position								S	S
6	Rudder Pedal Position				S	S			S	S
7	Elevon (L) Servo Actuator Position						X	X	X	X
8	Elevon (R) Servo Actuator Position						X	X	X	X
9	Rudder Servo Actuator Position				X	X			X	X
10	Elevon (L) Control Valve Spool Position						X	X	X	X
11	Elevon (R) Control Valve Spool Position						X	X	X	X
12	Rudder Control Valve Spool Position				X	X			X	X
13	Elevon (L) Surface Inboard Hinge Position				X		X	X	X	X
14	Elevon (R) Surface Inboard Hinge Position				X		X	X	X	X
15	Rudder Surface Lower Hinge Position				X	X			X	X
16	Elevon (L) Surface Actuator Position						X	X	X	X
17	Elevon (R) Surface Actuator Position						X	X	X	X
18	Rudder Surface Actuator Position				X	X			X	X
19	Elevator Trim Actuator Position						X	X	X	X

S - Suggested for Flight Safety or Supplementary Information
X - Required Measurement

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DATA MEASUREMENTS REQUIRED FOR FLIGHT TEST ITEMS

Item Number	MEASUREMENT	FLIGHT TEST ITEMS Airspeed Calib. Subsonic Range 1	Airspeed Calib. Trans-Supersonic 2	Angles Attack-Yaw Pickup Calib. 3	Dyn. Lat-Direc. Stab. No Damping 4	Dyn. Lat-Direc. Stab. With Damp. 5	Dyn. Long. Stab. Without Damping 6	Dyn. Long. Stab. With Damping 7	Dyn. Resp. Speed Brake Operation 8	Dyn. Resp. Bay Door Operation 9
20	Aileron Trim Actuator Position								X	X
21	Rudder Trim Actuator Position				X	X			X	X
22	Engine Power Lever Position	S	S	S	X	X	X	X	X	X
23	Lower Speed Brake Position								X	
	FLIGHT CONTROL SUB-SYSTEM GROUP									
1	Rudder Summing Amplifier				X	X				
2	Pitch Rate Sensing Gyro					X	X	X		
3	Roll Rate Sensing Gyro				X	X				
4	Yaw Rate Sensing Gyro				X	X				
5	Elevon (L) Control Valve Actuator Follow-Up Potentiometer						X	X		
6	Elevon (R) Control Valve Actuator Follow-Up Potentiometer						X	X		
7	Rudder Control Valve Actuator Follow-Up Potentiometer				X	X				
8	Elevon (L) Position Potentiometer						X	X		
9	Elevon (R) Position Potentiometer						X	X		
10	Power Supply (+105-volts d-c)				X	X	X	X	X	X
11	Power Supply (-105-volts d-c)				X	X	X	X	X	X
12	Flight Programmer Pitch Signal Output						X	X	X	X
13	Flight Programmer Yaw Signal Output				X	X			X	X

S - Suggested for Flight Safety or Supplementary Information
X - Required Measurement

DATA MEASUREMENTS REQUIRED FOR FLIGHT TEST ITEMS

Item Number	MEASUREMENT	FLIGHT TEST ITEMS								
		Airspeed Calib. Subsonic Range 1	Airspeed Calib. Trans-Supersonic 2	Angles Attack-Yaw Pickup Calib. 3	Dyn. Lat-Direc. Stab. No. Damping 4	Dyn. Lat-Direc. Stab. With Damping 5	Dyn. Long. Stab. Without Damping 6	Dyn. Long. Stab. With Damping 7	Dyn. Resp. Speed Brake Operation 8	Dyn. Resp. Bay Door Operation 9
	MISCELLANEOUS									
1	Time of Day	X	X	X	X	X	X	X	X	X
2	Time Correlation	X	X	X	X	X	X	X	X	X
3	OAT - Stagnation Probe on Metal Nose	X	X	X	X	X	X	X	X	X
4	OAT - Stagnation Probe in Engine Inlet Duct	X	X							
5	OAT - Flush Probe in Engine Inlet Duct	X	X							
6	ATM Alternator Frequency	S	S	S	S	S	S	S	S	S
7	Ratiometer (P_{t7}/P_{t2})	S	S	S	X	X	X	X	X	X
8	Fuel Quantity (Total)	X	X	X	X	X	X	X	X	X
9	Missile Bay Doors Opening Indication (sw - Left side bay doors opened 2 in.)									X
10	Engine RPM, N_2	S	S	S	X	X	X	X	X	X
11	Pilot's Recorded Comments	X	X	X	X	X	X	X	X	X
	DATA FROM INDICATOR LIGHTS									
1	Afterburner Nozzle Position - Open/Closed	X	X	X	X	X	X	X	X	X
2	Low Speed Compressor Bleed Valves (2) Position - Closed	X	X	X	X	X	X	X	X	X
3	Missile Bay Door Position - Open									X

S - Suggested for Flight Safety or Supplementary Information
X - Required Measurement

CONF

FLIGHT

PIT

FLIGHT CONDITIONS & CONFIGURATIONS

Flight Test Item	Run No.	Config.	Press. Altitude	Trim Speed IAS MPH	Pulse inputs Elevator or Rudder		Pitch Gain K_4	Pitch Network	Yaw Gain K_2	Yaw Time Constant τ_1	Yaw Gyro Tilt Angles	K_1 & K_3
					Deg.	Sec.						
4	1	PA	15,000	230	+2R	1.0	0	In	0	1.0	0	0
↓	2	PA	↓	(M=.4)	-2R	↓	↓	↓	↓	↓	↓	↓
↓	3	P	↓	↓	+2R	↓	↓	↓	↓	↓	↓	↓
↓	4	↓	↓	↓	-2R	↓	↓	↓	↓	↓	↓	↓
↓	5	↓	↓	345	+2R	↓	↓	↓	↓	↓	↓	↓
↓	6	↓	↓	(M = .6)	-2R	↓	↓	↓	↓	↓	↓	↓
↓	7	↓	↓	515	+2R	↓	↓	↓	↓	↓	↓	↓
↓	8	↓	↓	(M = .9)	-2R	↓	↓	↓	↓	↓	↓	↓
↓	9	↓	35,000	255	+2R	↓	↓	↓	↓	↓	↓	↓
↓	10	↓	↓	(M = .7)	-2R	↓	↓	↓	↓	↓	↓	↓
↓	11	↓	↓	V_{max}	+2R	↓	↓	↓	↓	↓	↓	↓
↓	12	↓	↓	(Level)	-2R	↓	↓	↓	↓	↓	↓	↓
↓	13	↓	45,000	260	+2R	↓	↓	↓	↓	↓	↓	↓
↓	14	↓	↓	(M = .9)	-2R	↓	↓	↓	↓	↓	↓	↓
6	15	PA	15,000	230	+2E	.75	↓	↓	↓	↓	↓	↓
↓	16	PA	↓	(M = .4)	-2E	↓	↓	↓	↓	↓	↓	↓
↓	17	P	↓	↓	+2E	↓	↓	↓	↓	↓	↓	↓
↓	18	↓	↓	↓	-2E	↓	↓	↓	↓	↓	↓	↓
↓	19	↓	↓	345	+1E	↓	↓	↓	↓	↓	↓	↓
↓	20	↓	↓	(M = .6)	-1E	↓	↓	↓	↓	↓	↓	↓

NDITIONS AND CONFIGURATI
AND YAW DAMPING TESTS

	21			515	+.5E							
	22		↓	(M = .9)	-.5E							
	23		35,000	255	+2E							
	24			(M = .7)	-2E	↓	↓					
	25			V_{max}	+1E	.3	0					
	26		↓	↓	-1E	↓						
	27		45,000	260	+2E	.75						
↓	28		↓	(M = .9)	-2E	↓			↓	↓		
5	29		15,000	230	+2R	1.0			Nom.*	Nom.*		
	30		↓	(M = .4)	-2R							
	31		35,000	255	+2R							
	32			(M = .7)	-2R							
	33			V_{max}	+2R							
↓	34		↓	↓	-2R	↓	↓		↓	↓		
7	35		15,000	230	+2E	.75	Nom.*		0	1.0		
	36		↓	(M = .4)	-2E							
	37		35,000	255	+2E							
	38			(M = .7)	-2E	↓						
	39			V_{max}	+1E	.3						
↓	40	↓	↓	↓	-1E	↓	↓	↓	↓	↓	↓	↓

NOTES:

Nom.* = Nominal values derived from items 4 & 6
+2R = Left rudder deflection of 2°
-2R = Right rudder deflection of 2°
+2E = Down elevator deflection of 2°
-2E = Up elevator deflection of 2°
V_{max} = Max. level flt speed with Milita-(Afterburner) Power
PA = Landing Configuration
P = Clean configuration

FLIGHT CONTROL SYSTEM

General

The YF-102 flight control surfaces are actuated by irreversible, dual, full hydraulic power control systems, which include artificial feel, rate-gyro stabilized pitch and yaw damping systems, trimming devices, and various safety features. Full hydraulic power is used because of the large control surface hinge moments required to deflect the surfaces at high dynamic pressures and because the control system must be capable of responding to electronic signals from an automatic fire control system.

The airplane can be flown at the pilot's option in 3 modes of operation. These are Direct Manual Flight Mode, Manual Flight Mode, and the Pilot Assist Mode.

In the Direct Manual Mode, the rate-gyro stabilized damper systems have no effect on the basic control system; flight control surface response results solely from pilot movement of stick and rudder pedals which operate the full hydraulic power control systems. This mode is always available at the pilot's option or automatically in case of a malfunction sensed by the damping system monitoring circuits, loss of hydraulic pressure in the secondary system or loss of electrical power to the control valve servo actuators.

The control stick and rudder pedals are mechanically connected to the main hydraulic control valves by three separate cable, rod, and bellcrank systems; one to operate the elevons as elevators, one to operate elevons as ailerons, and one for the rudder. When operating in Direct Manual mode, the damping system is disengaged and pilot control motion moves the spools in the hydraulic control valves which, in turn, admit hydraulic pressure to the surface actuating cylinders. The cylinders move the surfaces only as long as the pilot moves his

FLIGHT CONTROL SYSTEM (Continued)

General

controls. When pilot control motion stops, a mechanical follow-up mechanism shuts the control valve off. The surfaces are then hydraulically locked in the position last selected by pilot control, and will remain locked in that position until repositioned by pilot initiated control motion.

Trim is accomplished by changing the zero feel force position by means of electrical trim actuators installed between the feel mechanism and the mechanical control mechanism.

In the Manual Flight Mode, the pilot has control of the airplane as previously described for the Direct Manual Mode, but in addition, the airplane has dynamic stability augmentation superimposed by the pitch and yaw damping system. The elevons and rudder operate with limited authority with respect to pitch and yaw damping, but the pilot has full authority through his own controls. Elevon and rudder deflections, which are superimposed from the pitch and yaw damping system are not felt at pilot's controls.

Pitc aand Yaw Damping System

A functional block diagram of the pitch and yaw damping system is shown on page 47. The system has been designed to give the airplane a high degree of dynamic stability over its operating range and thereby provide desirable flying qualities as well as a highly stable armament launching platform.

FLIGHT CONTROL SYSTEM
(Continued)

Pitch and Yaw Damping System

The principal components of the pitch damping system are:

1. Pitch rate gyro
2. Pitch rate network
3. Left and right elevon servo amplifiers
4. Left and right elevon control-valve servo actuators
5. Power supply common to both pitch and yaw damping systems.

In the Manual Flight Mode, the pitch damping system will tend to maintain the change in pitch rate of the airplane at zero value. The pitch rate gyro senses any pitch rate (oscillation or rotation of the airplane about its lateral axis), and sends a corrective voltage signal through the pitch rate network to the elevon servo amplifiers. After the signal is amplified, it is fed to the left and right elevon control valve servo actuators which are connected to the main hydraulic control valves in series with the pilot's linkage. The servo actuators are hydraulic in operation, and are capable of responding to the electronic signals of the servo amplifiers by means of an electric torque motor which is an integral part of the actuator. These actuators move the main elevon control valves which produce an elevator deflection tending to reduce the airplane's pitch rate to zero. Follow-up potentiometers mounted on the actuators cause these actuators to return to neutral or closed position when the corrective deflection called for by the pitch rate gyro has been accomplished. The process is repeated as long as a change in pitch rate is sensed by the pitch rate gyro.

FLIGHT CONTROL SYSTEM
(Continued)

Pitch and Yaw Damping System

In the Manual Flight Mode, the pitch damping system is capable of producing elevator deflections of 2.7 degrees. During initial flight tests the pitch damping authority will be limited to elevator deflections of .75 degrees.

The principle components of the yaw damping system are:

1. Yaw rate gyro
2. Roll rate gyro
3. Airspeed compensator
4. Yaw rate network
5. Roll rate network
6. Aileron position potentiometer
7. Rudder summing amplifier
8. Rudder servo amplifier

In the Manual Flight Mode, the yaw damping system tends to maintain the change in yaw rate of the airplane at zero value. The yaw damping system will also coordinate sideslip up to the limit of its authority. The yaw rate and roll rate gyros function to sense any yaw or roll rates of the airplane and send out corrective voltage signals. The output of the yaw rate gyro is biased by the aileron position potentiometer and the yaw rate network. The output of the roll rate gyro is modified by the airspeed compensator and the roll rate network. These signals are summed together by the rudder summing amplifier and fed to the rudder servo amplifier. After amplification, the signal is fed to the rudder control valve servo actuators. This actuator moves the main rudder control valve which causes a rudder deflection tending to reduce lateral-directional

FLIGHT CONTROL SYSTEM (Continued)

Pitch and Yaw Damping System

oscillations or to reduce sideslip during banked turns. The rudder control valve servo actuator is equipped with a follow-up potentiometer which causes the actuator to return to neutral or closed position when the corrective deflection called for has been accomplished. The process is repeated as long as yaw and/or roll rates are sensed by the yaw and roll rate gyros.

In the Manual Flight Mode, the yaw damping system is capable of producing rudder deflections of ±3-1/2 degrees.

Pilot Assist System

The Pilot Assist Mode utilizes the pilot assist system. The pilot assist system is being developed for use on F-102 type airplanes that have an interim type fire control system without CSTI. A functional block diagram of the pilot assist system is shown on page 48.

The pilot assist system may be engaged when the airplane is being flown in the Manual Flight Mode only. The function of the pilot assist system is to maintain the airplane continuously in a reference attitude taken as that attitude which exists at the instant the system was engaged.

In the Pilot Assist Mode, the pilot's control of the elevon system through the stick control is locked out by the elevon manual control lockout valves. Mechanical feedback from the elevon to the main control valve is locked out, and an electrical feedback system is used. Full authority control of the elevon system is accomplished by the pilot assist system. The pilot's stick control moves with the elevon surfaces, but the pilot cannot produce elevon deflections. However, by exerting the necessary stick forces for "break-out" the pilot may "breakout" of

FLIGHT CONTROL SYSTEM (Continued)

Pilot Assist System

the Pilot Assist Mode. This action will return the control system to Direct Manual Flight Mode.

Control of the elevon system in Pilot Assist Mode is accomplished by the control stick trim button, which becomes the pilot assist pitch and roll "beep" control button when the pilot assist is engaged. Operation of the pilot assist "beep" control serves to change the attitude gyro reference voltage of the pilot assist system, and thereby establishes a new reference attitude for the system. If a large change in reference attitude is desired, the pilot assist system is disengaged and the airplane flown in the Manual Mode to the desired attitude and pilot assist engaged again.

SPECIAL TEST EQUIPMENT

Flight Test Programmer System

A flight test programmer has been installed in this airplane to minimize the time required to conduct certain tests and to provide a means of accurately repeating the test independent variables. The programmer will be used to produce pulsed, control surface deflections of predetermined rate, amplitude and duration.

A parameter control box is used in conjunction with the programmer system to vary the characteristics of the pitch rate, yaw rate, and roll rate network circuits in the pitch and yaw damping system, and also the various parameters in the pilot assist system. The variation of these parameters provides a means of changing the characteristics of either system to determine the necessary parameters for optimum operation.

The flight test programmer consists of the following components:

	Location
1. Programmer control panel	Pilot's R/H console
2. Input hand control	Pilot's L/H console
3. Pulse counter	Pilot's R/H console
4. Programmer	Fwd. center missile bay
5. Programmer relay box	Fwd. center missile bay

A detailed description of these components is presented in Reference (B). The flight test programmer system is a special piece of flight test equipment, and is not a part of the damping or pilot assist system.

The programmer system produces pulsed control surface deflections by supplying a voltage signal to the servo amplifiers of the pitch and yaw damping system in the case of limited authority damping tests (see flight test

SPECIAL TEST EQUIPMENT (Con't.)

Flight Test Programmer System

items 4-7, 8 and 9 or to the summing amplifiers of the pilot assist system in the case of full authority elevon tests.

The programmer system may be operated in two different ways at the pilot's option. These are the "manual" or "programmer", and are selected on the programmer control panel. The airplane must be flown in the Manual Flight Mode to utilize the programmer system in either of the above operations.

In "programmer" all the details of each individual test to be conducted during any given flight are automatically sequenced by the programmer. The cockpit pulse counter indicates to the pilot the number of programmed pulses which have been accomplished during the flight.

The programmer itself is the primary component of the system, and is utilized for "programmer" of operation. It is an electro-mechanical sequencing device installed in the forward center missile bay. It operates on the principle of using a moving strip of film to keep two banks of contact points separated. The contact points connect various programmer and parameter change control relay circuits. Perforations in the tape permit the points to close in single sets or groups at any given time, and for any chosen interval. The tape moves at a constant predetermined rate.

The programmer uses a rolled 40 foot strip of 70mm film tape which advances at a rate of one inch/second during programmer operation. This provides for 20 to 30 average length test pulse periods.

SPECIAL TEST EQUIPMENT (Con't)

Flight Test Programmer System

To accomplish a flight test or group of tests in "programmer", a program tape must be prepared on the ground prior to flight. This tape is then installed and checked out in the airplane before take-off. The necessary settings of the parameter change control panel are also made before take-off. The airplane is flown to the flight condition indicated by the flight plan, and trimmed in stabilized level flight in the Manual Flight Mode for the given conditions. The programmer control panel controls must be set in the appropriate manner for the type test to be conducted. The pilot then flys "hands-off" the stick and rudder pedal controls and operates the input hand control. The programmer operates the recording system and applies the input signal to the control system. The input signal may be stopped by the pilot by releasing the input hand control.

The actual control system pulse which is supplied by the programmer may or may not be felt by the pilot, but will be recorded by the pulse counter when the signal is applied. After the pulse is applied, the programmer causes the data recording system to operate for at least 10 seconds. The programmer then causes the resetting of the various parameters in the parameter control box for the next programmed test. The various test runs which will be accomplished using the flight test programmer system are tabulated on page 33.

When the necessary parameter changes have been accomplished for the next pulse, the programmer will stop and a signal light will go out on the programmer control panel. This is an indication to the pilot that the programmer is ready for the next pulse. No manual positioning of the parameter change control by the

SPECIAL TEST EQUIPMENT (Cont.)

pilot is required during the entire programmed flight. The airplane may or may not require to be retrimmed between the signal pulses supplied by the programmer.

In the "manual" operation of the programmer system, no tape is required as the programmer unit is not used. The pilot is required to manually position all programmer control panel and parameter change panel controls before each input signal pulse is applied. Any series of operations produced automatically by the "programmer" tape method can be repeated by manual operation if so desired.

Manual operation of the programmer system may be accomplished during a tape programmed flight providing the correct operations are performed. If a manual operation is performed during a tape programmed flight, the parameter change controls must be reset manually by the pilot to the last automatic settings which were set by the programmer before manual operation was begun. If these controls are not reset correctly, all subsequent "programmed" tests will be incorrect.

Parameter Change System

The parameter change system consists of a parameter relay box and control panel located in the cockpit R/H console. The parameter control panel contains eleven rotary switches which control various resistors and capacitors in the panel. These resistors and capacitors determine the various parameters in the

SPECIAL TEST EQUIPMENT (Con't.)

Parameter Change System

damping and the pilot assist systems. The various damper system and pilot assist system parameters which are controlled by the parameter change system are listed on page 45.

The damper parameter change control relay box, located under the R/H console of the cockpit, is operative only when the programmer is used in "programmer". It responds to tape programmed signals that determine which of the damper parameter change controls will be automatically repositioned next by the subsequent tape perforations.

The parameter change system has two methods of operation, manual and automatic, corresponding to the two methods of operation of the programmer system. When the programmer system is to be operated in "manual", the parameter change control panel selector knobs must be positioned manually by the pilot before each operation to select the various parameters as required by the test. When the programmer system is used in "programmer", the parameter change control system is controlled by the program punched in the programmer tape after a preliminary manual setting of the parameter change controls.

A program being run automatically in "programmer" may be interrupted to repeat a pulse or to substitute a different set of parameters than those programmed, but this may be done only if all of the selector knobs on the parameter change panel are returned to the positions required for the next expected programmed pulse. That is, they must be returned to the position they were in at the end of the previous automatic operation.

SPECIAL TEST EQUIPMENT (Con't.)

Parameter Change System

When all parameter change control selector knobs are set to the positions for zero parameter values, the programmer system may be used to supply input signals to the control system, but the damping system will not superimpose any damping action on the resulting airplane response. In effect, this simulates the Direct Manual Flight Mode while the airplane is being flown in the Manual Flight Mode. This technique will be used to determine the inherent dynamic damping characteristics of the airplane to programmed, pulsed, control surface deflections, (see flight test items 4 and 6, pages 18 and 20).

Parameter Control Box Variables

The following variables are controlled by the rotary "Ledex" switches on the parameter control panel. The symbols correspond to those used in the pitch and yaw damping and pilot assist system diagrams, pages 47 and 48, and in the Flight Condition and Configuration Table on page 33.

K_2	Yaw rate into rudder gain
K_3	Roll rate into rudder gain
τ_1	Time constant, roll rate, yaw rate
K_1	Aileron into rudder gain
K_4	Pitch rate gain
Pitch Rate Network	
K_5	Pilot Assist pitch gain
K_6	Pilot Assist up elevator gain

SPECIAL TEST EQUIPMENT (Con't.)

Parameter Control Box Variables

K_7	Pilot assist roll gain
K_8	Pilot assist roll rate gain
K_9	Pilot assist heading gain

LEGEND

L. E.	Left Elevon
R. E.	Right Elevon
F. C.	Fire Control
C. V.	Control Valve
δ_r	Rudder Deflection
δ_{e_l}	Left Elevon Deflection
δ_{e_r}	Right Elevon Deflection
δ_{e_p}, δ_{e_p}, δ_{a_p}	Pilot Control Motions
δ_{a_D}	Desired Aileron Deflection
δ_{e_D}	Desired Elevator Deflection
TP	Test Points, Leads to HAC Plug
HAC	Hughes Aircraft Co.
M	Mach Number
q	Dynamic Pressure
P_{1E}	Programmer System input signals for elevator deflections with CST1 switch open (Manual Flight Mode).
P_{1R}	Programmer system input signals for rudder deflections with CST1 switch open (Manual Flight Mode).
P_{2A}	Programmer system input signals for aileron deflections in full authority elevon control without pilot assist. (CST1 switch closed).
P_{2E}	Programmer system input signals for elevator deflections in full authority elevon control without pilot assist. (CST1 switch closed).

DAMPER SYSTEM PARAMETERS

K_1	Aileron into rudder gain factor
K_2	Yaw Rate into Rudder Gain Factor
K_3	Roll Rate into Rudder Gain Factor
K_4	Pitch Rate Gain Factor
τ_1	Yaw & Roll Rate Time Constant
Pitch Network	

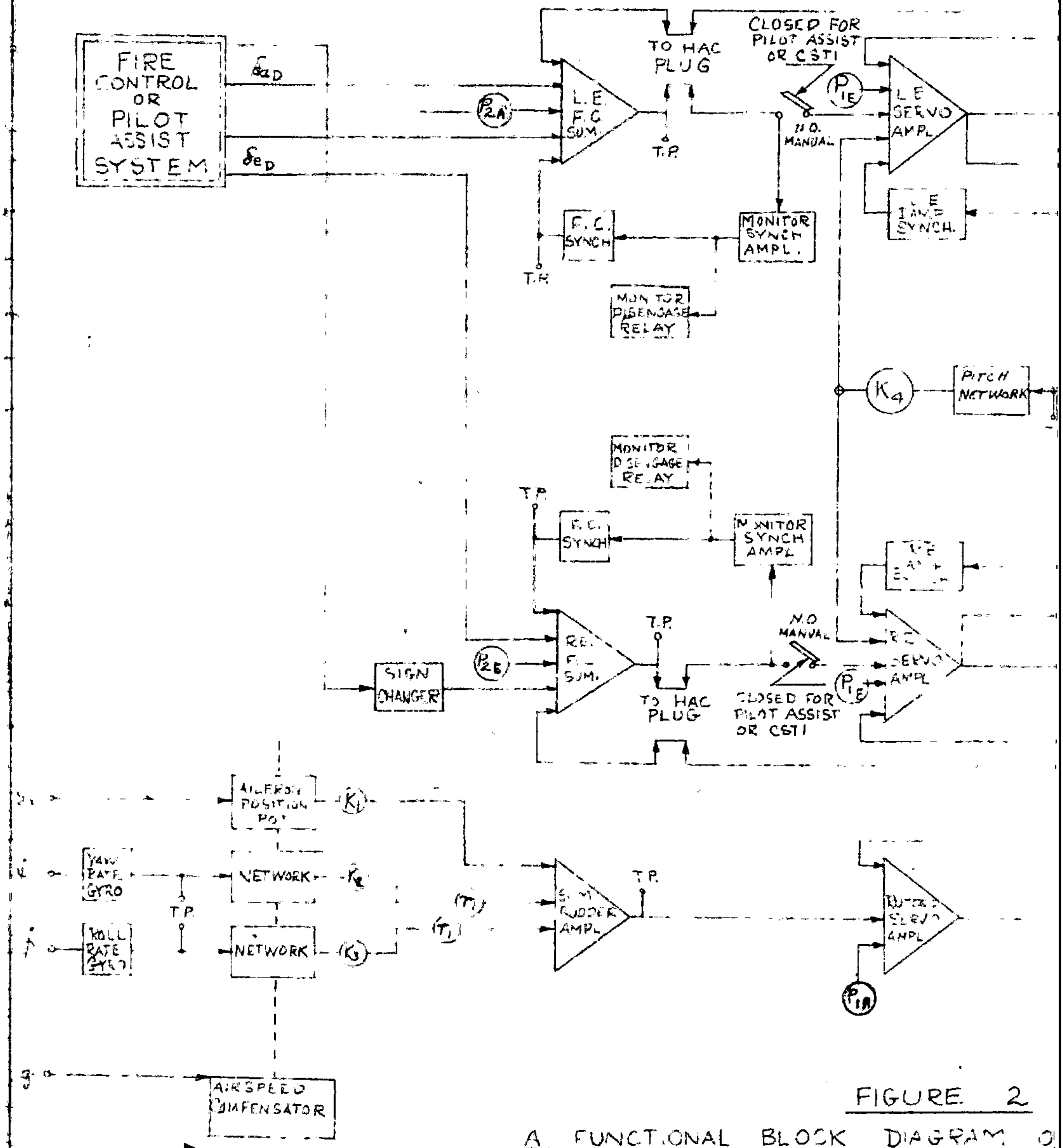
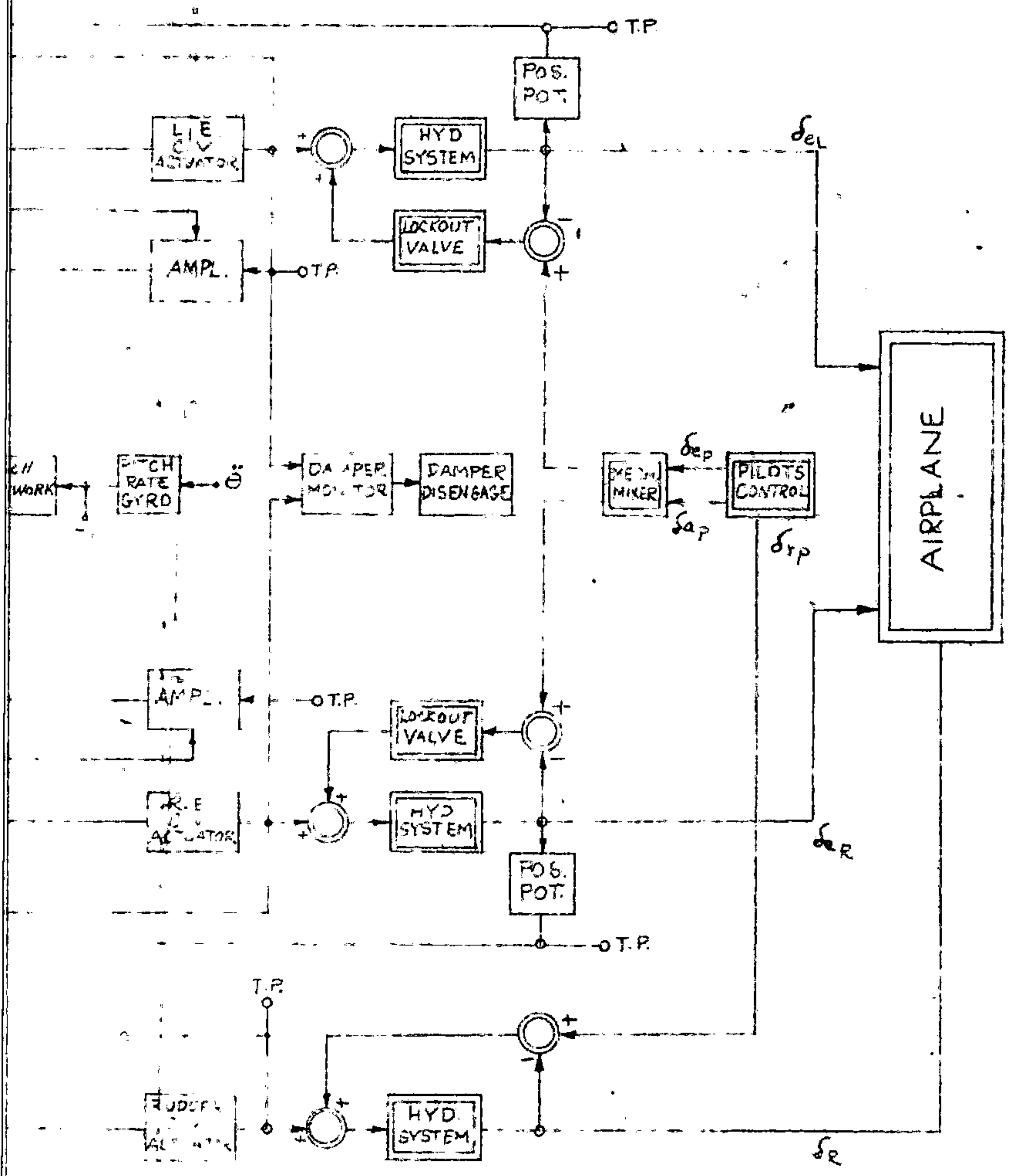


FIGURE 2

A FUNCTIONAL BLOCK DIAGRAM O

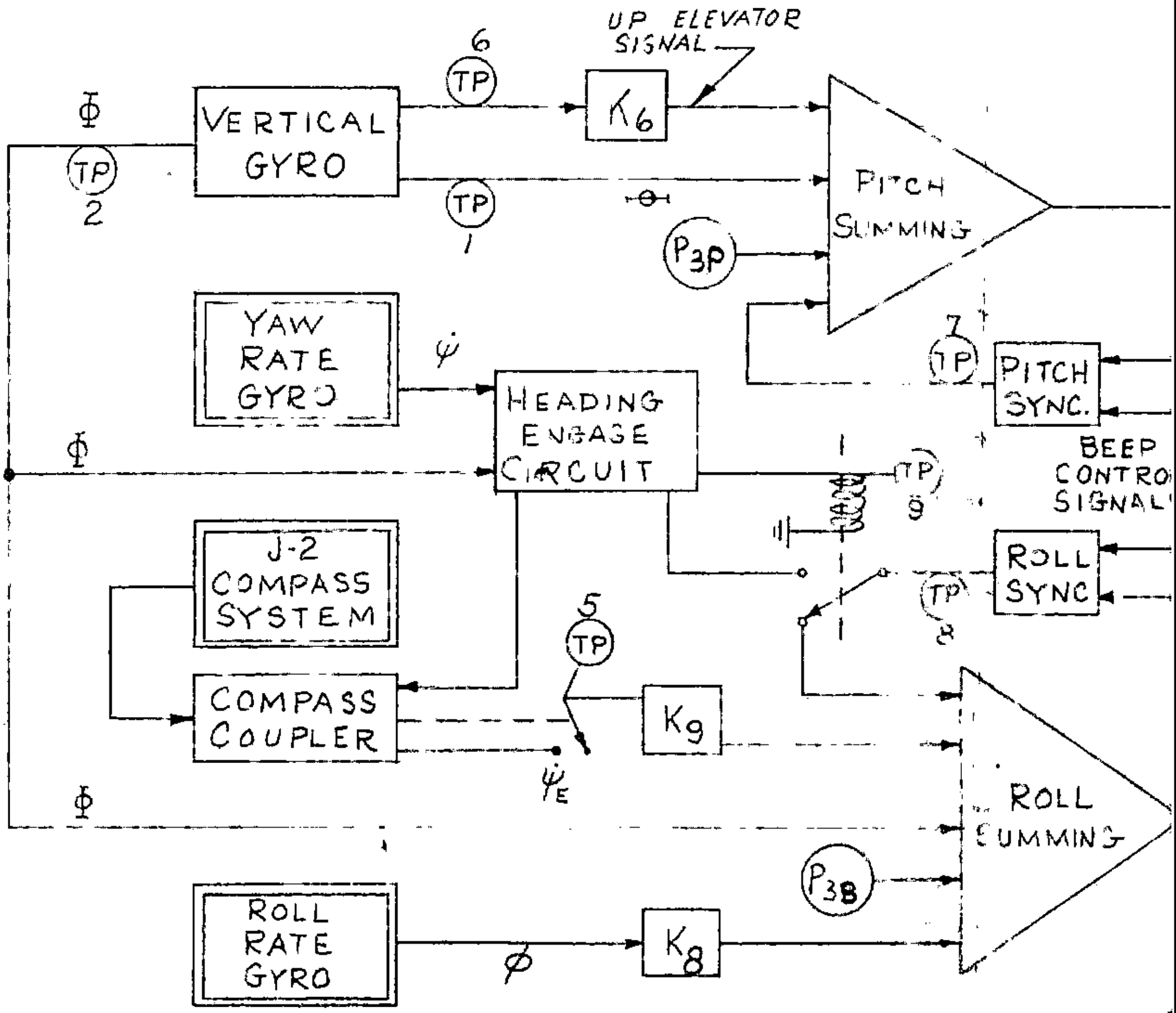
AND YAW DAMPER SYSTEM FO



CONFIDENTIAL
SECURITY INFORMATION

2

M. OF THE PITCH DAMPER
FOR THE F-102 AIRPLANE



K5, Pil

K6, Pil

K7, Pil

K8, Pil

K9, Pil

*Th
pa

LEGEND

(TP) - - - Test Points

Φ - - - Roll Attitude

ϕ - - - Roll Rate

$\dot{\psi}$ - - - Yaw Rate

$-\theta-$ - - - Pitch Attitude

R.E., L.E.- Right Elevon Left Elevon

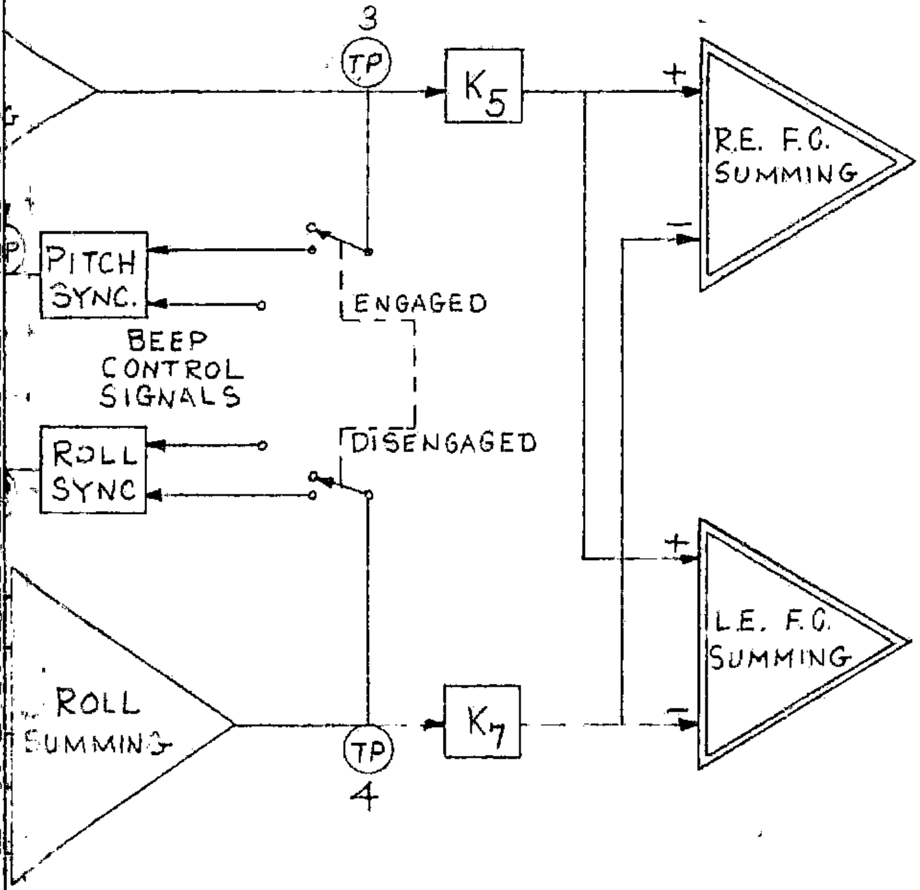
F.C. - - - Fire Control

P_{3P} Programmer System Input Signals fo
Pilot Assist Mode, (Pitch Attitude

P_{3B} Programmer System Input Signals Fo
Pilot Assist Mode (Roll Attitude (

FIGURE 1

F-102 PILOT ASSIST SYSTEM BLO



PILOT ASSIST SYSTEM PARAMETERS*

K_5, Pilot Assist Pitch Gain Factor

K_6, Pilot Assist Up Elevator Gain Factor

K_7, Pilot Assist Roll Gain Factor

K_8, Pilot Assist Roll Rate Gain

K_9, Pilot Assist Heading Gain Factor

*These parameters are controlled by the parameter change control panel.

nput Signals for Change in Pitch Attitude in
(Pitch Attitude Command Signals)

nput Signals For Change in Bank Attitude in
oll Attitude Command Signals)

URE 1

TEM BLOCK DIAGRAM

REFERENCES

(A) Convair ZM-8-095, Utility Flight Handbook, 53-1779, 15 April 1954

(B) Convair ZC-8-053, Test Instrumentation Report, 53-1779, 15 Feb 1954

(C) Convair ZC-8-001, Model Spec. for Convair Model 8, USAF Model YF-102 and F-102A, 1 March 1952, Revised 1 Dec. 1952

(D) Convair ZA-8-004, Predicted Performance Characteristics of the YF-102 Airplane (J57-P-11 Engine) 15 October 1953

(E) Convair ZA-232, A Statistical Method for Calibrating The Airspeed System of an Airplane

(F) Convair ZA-8-003, Predicted Flight Characteristics of the YF-102 Airplane, 1 October 1953

(G) Convair ZC-8-107, YF-102 and F-102A Accelerated Development Test Program, 4 April 1954

(H) Convair ZC-8-111, Flight Test Program for YF-102, S/N 53-1780, dated 2 June 1954.

[redacted]

R.L. Runnels ②
Eng. Flight

PROGRESS REPORT 9475-8-1

1115

UNCLASSIFIED

Date: 24 May 1955

To: W. W. Withoo - Chief of Experimental Flight Test Operations

Subject: Progress Report on Duct and Cockpit Buzz and Rumble Survey During High Speed Flight Test. Airplane YF 102A - No. 53-1787, Flights 47 and 48.

References:
(a) Test Request 9308
(b) Progress Report 9308-8-1
(c) Progress Report 9308-8-2
(d) Test Request 9475 (a continuation of Test Request 9308)

A detailed harmonic analysis has been made of acoustic levels present in the cockpit, and in the engine inlet duct. A detailed harmonic analysis has been made of vibration levels present in the modified engine air inlet duct splitter, the nose wheel door, and the nose cone. These data are presented in the accompanying graphs, figure 33 through figure 242.

The instrumentation used for this flight is as follows: A microphone was located in the cockpit at Station 183, at the pilots left shoulder. The microphone output was amplified and telemetered. This installation was the same as the microphone used in reference (b) and (c)

A microphone was located in the engine inlet duct at Station 430, W.L.O., in the Convair "Test Doughnut". The microphone output was amplified and telemetered. This microphone installation was the same as the installation used in reference (b) and (c).

An accelerometer was mounted in the trailing edge of the modified duct splitter, it's output being amplified and telemetered. This pickup recorded lateral vibrations of the duct splitter.

Two velocity pickups were mounted in the foreward end of the nose cone at approximately Station 39. These pickups were aligned to transmit lateral and vertical vibration signals. The output of the two pickups was amplified and telemetered.

All data was reduced from the telemetering tape recordings.

Figure 33 to figure 97 are the complete surveys of the first dive on flight 47.

Figure 98 to figure 188 are the complete surveys of the second dive on flight 47.

Figure 189 to figure 242 are the complete surveys of the dive on flight 48.

Figure 33 through 41 are a detailed harmonic analysis of the acoustic levels present in the cockpit during the dive to Mach 1.30. These data were reduced from discrete two second sections of the recorded data taken at selected Mach Numbers. Nine surveys were analyzed.

GR. 4
DOWNGRADED AT 3 YEAR INTERVALS; DECLASSIFIED AFTER
12 YEARS.
DOD DIR 5200.10

CLASSIFIED [redacted]

Figure 42 shows a plot of the significant frequencies 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 33 through 41.

Figure 43 is a plot of overall sound level of the cockpit as a function of Mach Number. These data were obtained from the survey points used for the harmonic analysis. This is an overall for the frequencies of 5 to 1000 cps.

Figure 44 through 53 are a detailed harmonic analysis of the acoustic levels present in the engine air inlet duct during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 54 shows a plot of the significant frequencies 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 44 through 53.

Figures 56 through 64 are a detailed harmonic analysis of the vibration levels present in the lateral plane of the duct splitter. These data were taken coincident with the cockpit survey.

Figure 65 shows a plot of the significant frequency 170 cps as a function of Mach Number taken from figures 56 through 64.

Figures 66 through 75 are a detailed harmonic analysis of the vibration levels present in the vertical plane in the nose cone during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figures 76 is a plot of the significant frequencies 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 66 through 75.

Figures 77 through 86 are a detailed harmonic analysis of the vibration levels present in the lateral plane in the nose cone during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 87 is a plot of the significant components 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 77 through 86

Figures 88 through 96 are a detailed harmonic analysis of the vibration levels present in the vertical plane of the nose wheel door. These data were taken coincidental with the cockpit survey.

Figure 97 is a plot of the significant components 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 88 through 96.

Figures 98 through 110 are a detailed harmonic analysis of the acoustic levels present in the cockpit during the second dive to Mach 1.30. These data were taken at discrete intervals of 2 seconds out of approximately every 4 seconds during the dive time. Fourteen (14) discrete surveys were made. Survey data taken from figure 110 proved to be below usable levels, therefore figure 110 has been deleted.

Figure 111 is a plot of the significant components 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 98 through 110.

Figure 112 is a plot of overall sound level of the cockpit as a function of Mach Number. These data were obtained from the increments used for the harmonic analysis. This is an overall for the frequencies of 5 to 1000 cps.

Figures 113 through 126 are a detailed harmonic analysis of the acoustic levels present in engine air inlet duct during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 127 is a plot of the significant frequencies 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 113 through 126.

Figures 129 through 142 are a detailed harmonic analysis of the vibration levels in the lateral plane of the duct splitter during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 143 is a plot of the significant frequency component 170 cps as a function of Mach Number taken from figures 129 through 142.

Figures 144 through 156 are a detailed harmonic analysis of the vibration level in the vertical plane of the nose cone during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 157 is a plot of the significant frequency components as a function of Mach Numbers taken from figures 144 through 156.

Figure 158 through 171 are a detailed harmonic analysis of the vibration levels present in the lateral plane of the nose cone during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 172 is a plot of the significant frequency components as a function of Mach Number taken from figures 158 through 171.

Figure 173 through 186 are a detailed harmonic analysis of the vibration components present in the vertical plane of the nose wheel door during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 187 is a plot of the significant frequency components as a function of Mach Number taken from figure 173 through 186.

Figure 188 is a time history of Mach Numbers through the two dives of Flight 47.

FLIGHT NUMBER 48

Figures 189 through 195 are a detailed harmonic analysis of the acoustic levels present in the cockpit during the dive to Mach 1.30. This data was reduced from discrete two second sections of the recorded data taken at selected Mach Numbers. Eight (8) discrete surveys were made.

Figure 196 shows a plot of the significant frequencies 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 189 through 195.

Figure 197 is a plot of overall sound level of the cockpit as a function of Mach Number. These data were obtained from the increments used for the harmonic analysis. This is an overall for the frequencies of 5 to 1000 cps.

Figures 198 through 205 are a detailed harmonic analysis of the acoustic levels present in the inlet ducts during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 206 is a plot of the significant frequencies 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken figures 198 through 205.

Figures 208 through 215 are a detailed harmonic analysis of the vibration levels present in the engine duct splitter. These data were taken coincident with the cockpit survey.

Figure 216 is a plot of the significant frequencies 170, cps plotted as a function of Mach Number taken from figures 208 through 215.

Figures 217 through 223 are a detailed harmonic analysis of the vibration levels present in the lateral plane in the nose cone during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 224 shows a plot of the significant frequencies 24, 40-50, 80-100, 170, 250-300, and 500 cps as a function of Mach Number taken from figures 217 through 223.

Figure 225 through 231 are a detailed harmonic analysis of the vibration levels present in the vertical plane in the nose cone during the dive to Mach 1.30. These data were taken coincident with the cockpit surveys.

Figure 232 shows a plot of the significant frequencies 24, 40-50, 80-100, 170, 250-300 and 500 cps as a function of Mach Number taken from figures 225 through 231.

Figures 233 through 240 are a detailed harmonic analysis of the vibration levels present in the nose wheel door in the vertical plane, during the dive to Mach 1.30. These data were taken coincident with the cockpit survey.

Figure 241 shows a plot of the significant frequencies 100, 170, 250-300 and 400 cps as a function of Mach Number, taken from figures 233 through 240.

Figure 242 is a time history of Mach Numbers vs. Counter Numbers for the portion of Flight 48 that contains the dive to Mach 1.30.

PREPARED BY W G Messer
W. G. Messer

CHECKED BY M C Brady
M. C. Brady
Dynamics Test Group Engineer

APPROVED BY E F Strong
E. F. Strong
Chief of Engineering Test Laboratory

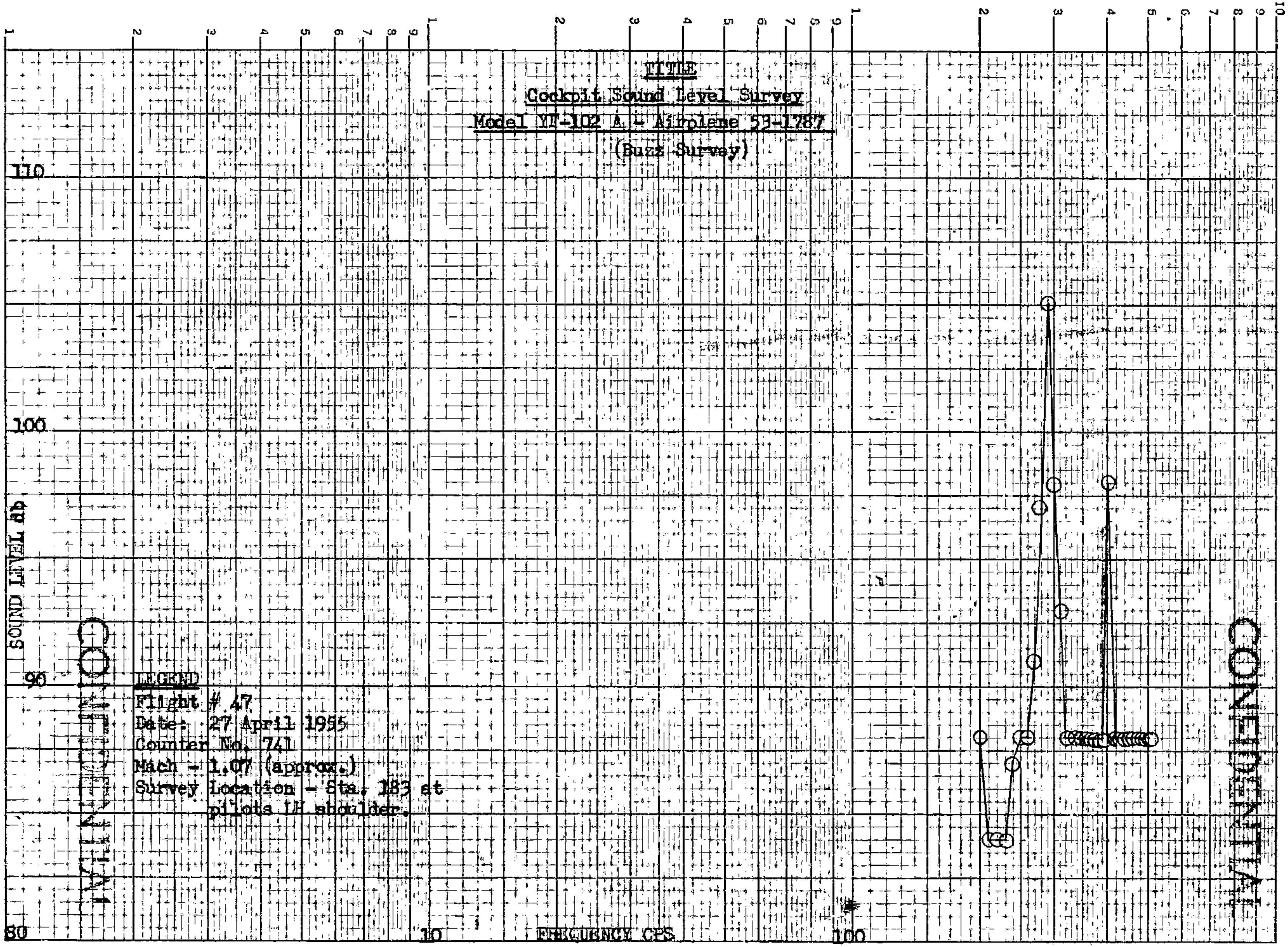
EFSLWGM:MCB:acc

cc:

Addressee	(2)
Model 8 P/O	(2)
Ken Ward	(2)
L. Clements	
M.Doublin/G.Getline	(2)
V.L.Allwardt/F.L.Cook	(3)
R.L.Runnels	(2)
T.L.Maloy	(2)
W.G.Messer	
E.F.Strong	
ETL Files	
Dyn. Test Files	

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Cockpit Sound Level Survey
Model YF-102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 741
Mach - 1.07 (approx.)
Survey Location - Sta. 183 at
pilots LH shoulder.

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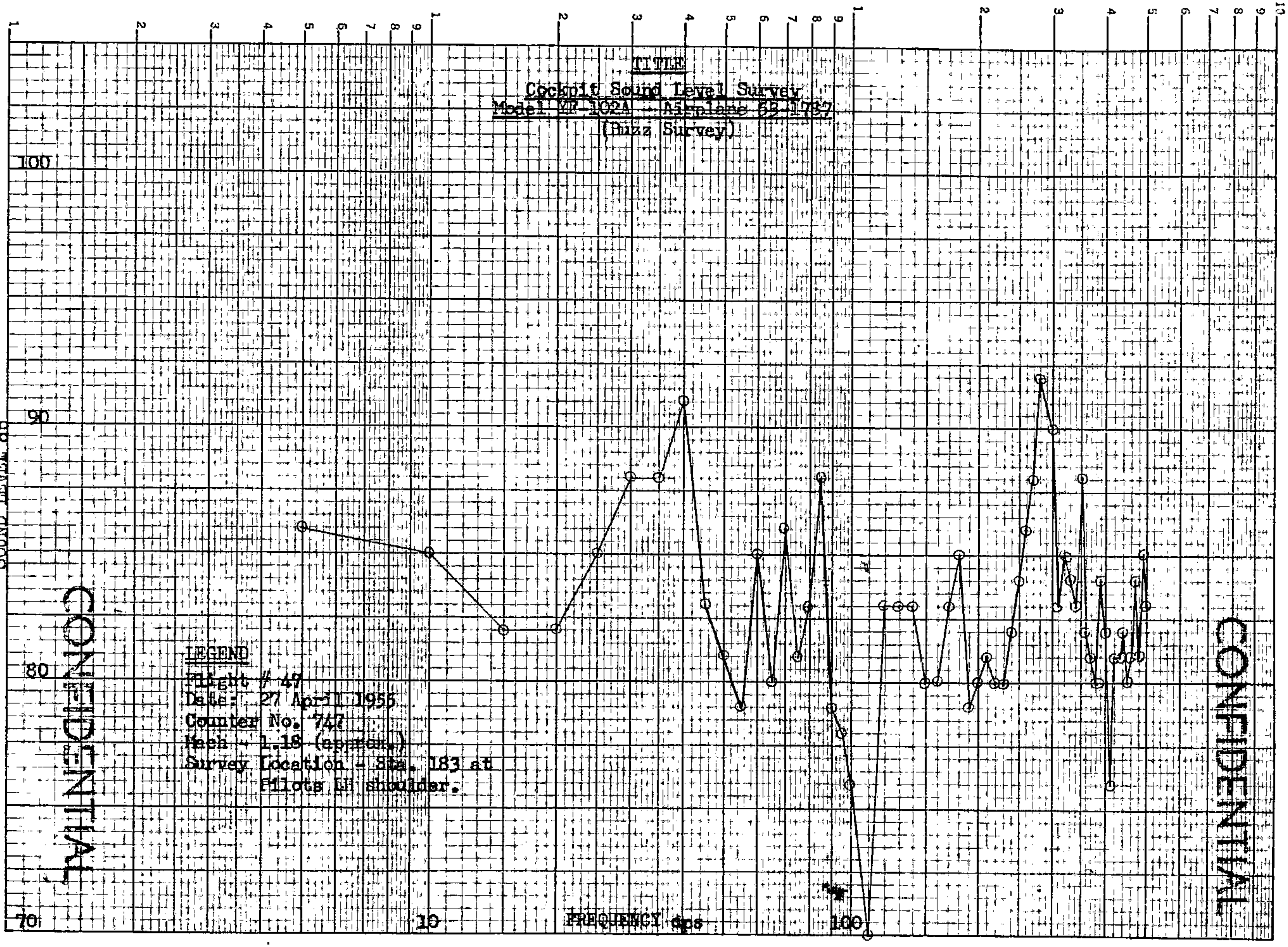
Fig. 33

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

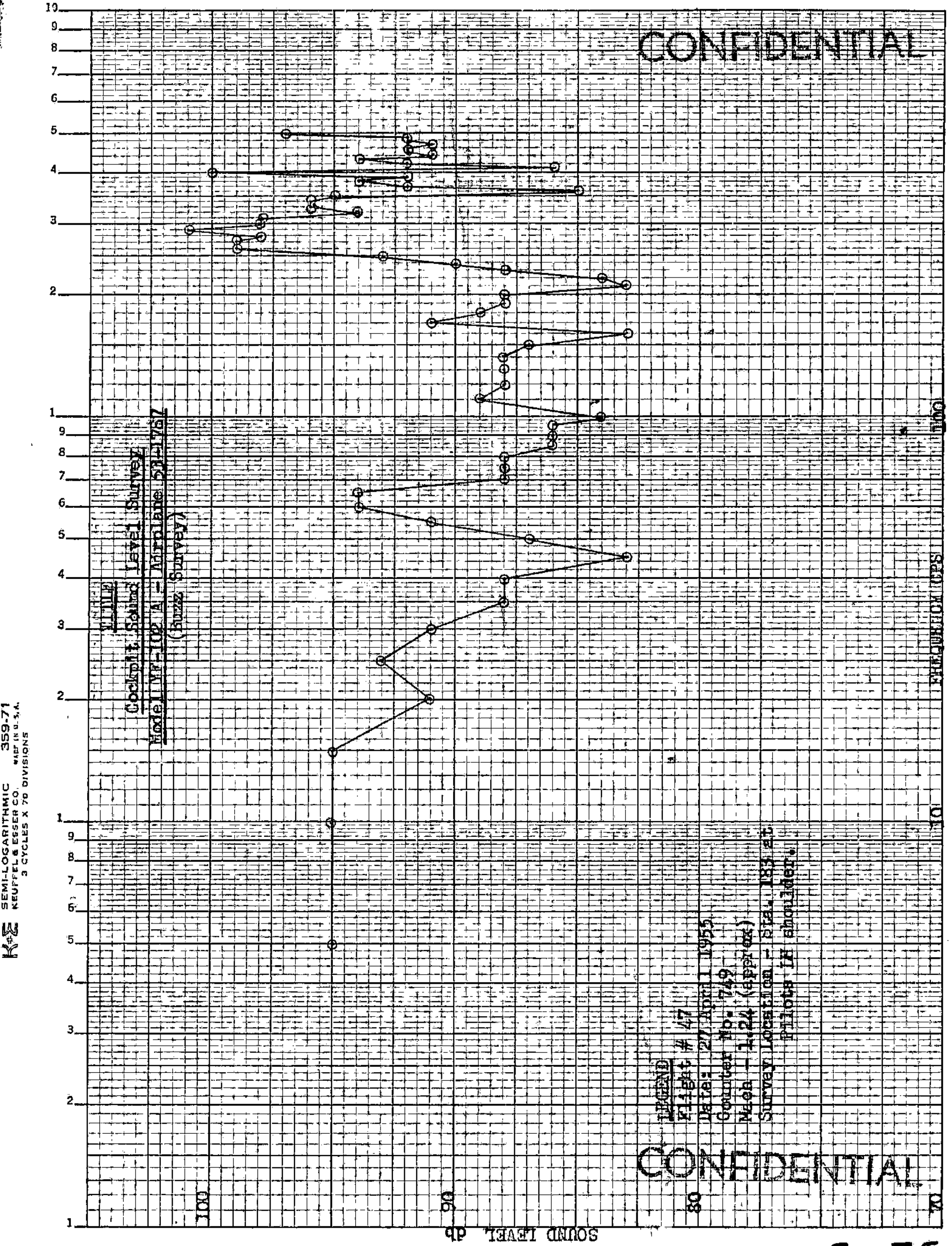
CONFIDENTIAL



CONFIDENTIAL

Fig. 34

CONFIDENTIAL



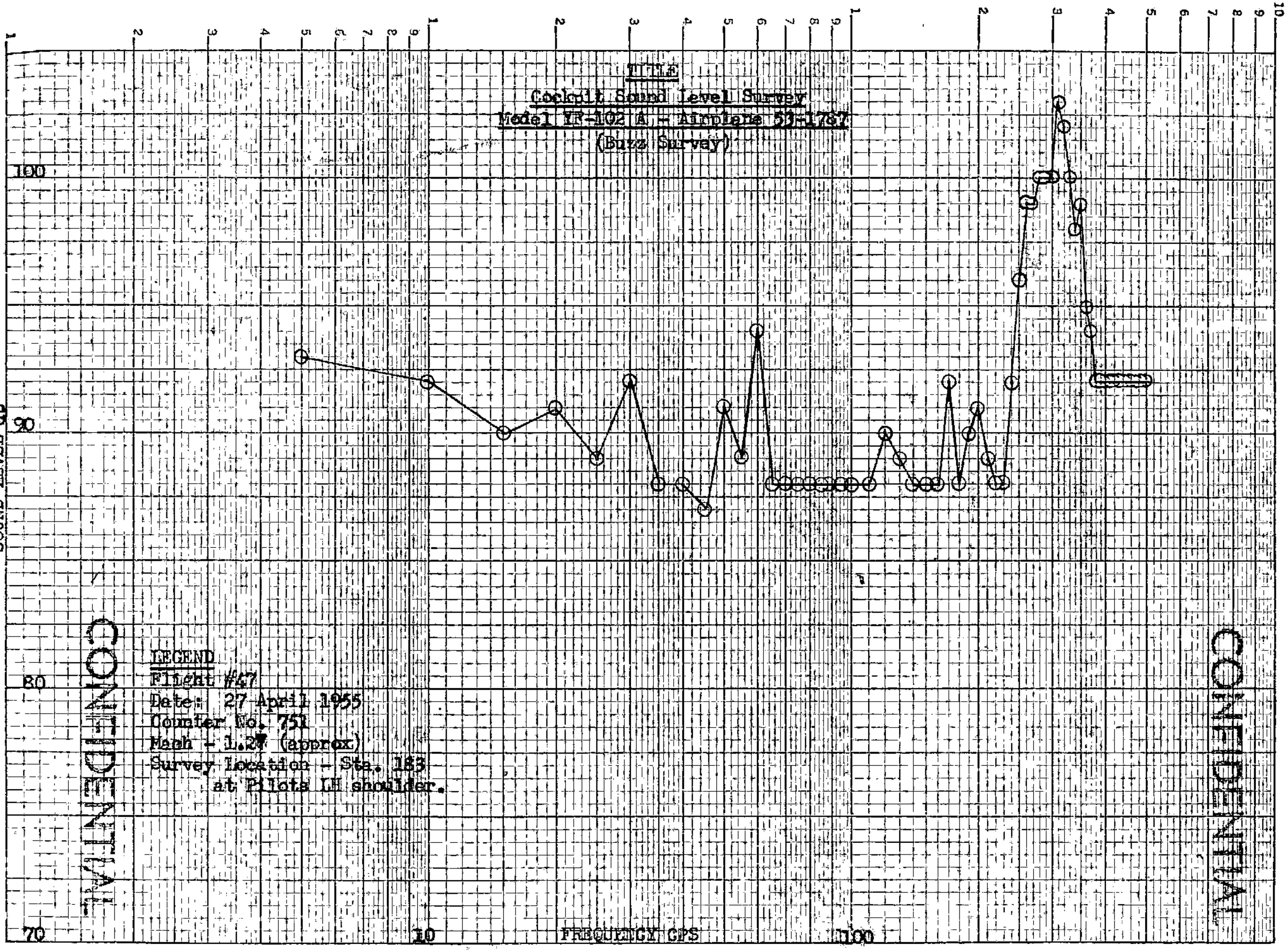
TITLE
Cockpit Sound Level Survey
Model YF-102 A - Airplane 53-1767
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 749
Mach - 1.24 (approx)
Survey Location - Sta. 183 at
Pilots LH shoulder.

CONFIDENTIAL

Fig. 35

CONFIDENTIAL



TITLE
Cockpit Sound Level Survey
Model YF-102 A - Airplane 53-1787
(Buzz Survey)

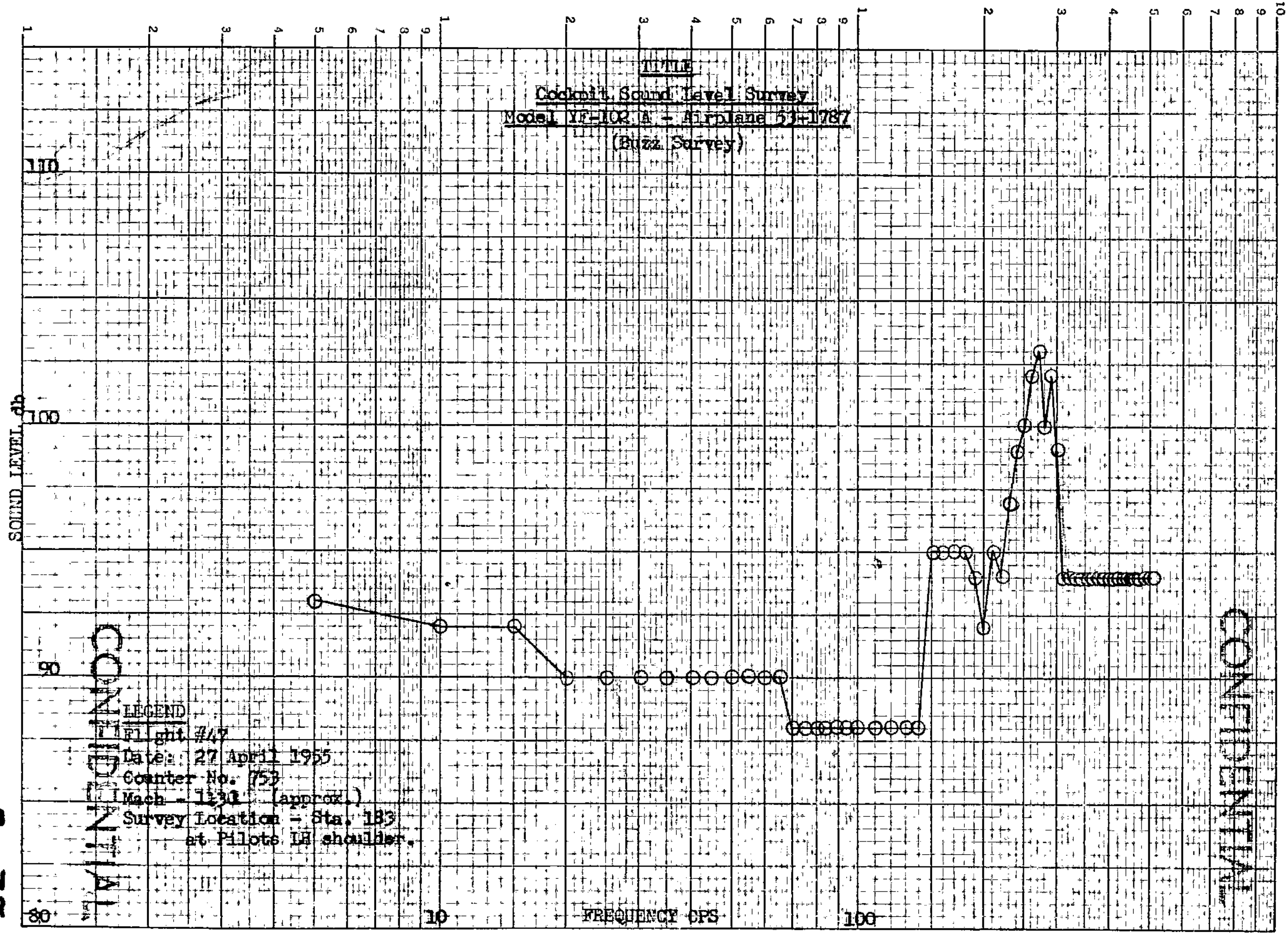
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 751
Mach - 1.27 (approx)
Survey Location - Sta. 183
at Pilots LH shoulder.

CONFIDENTIAL

Fig. 36

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Cockpit Sound Level Survey

Model YF-102 A - Airplane 53-1787

(Buzz Survey)

LEGEND

Flight #47

Date: 27 April 1955

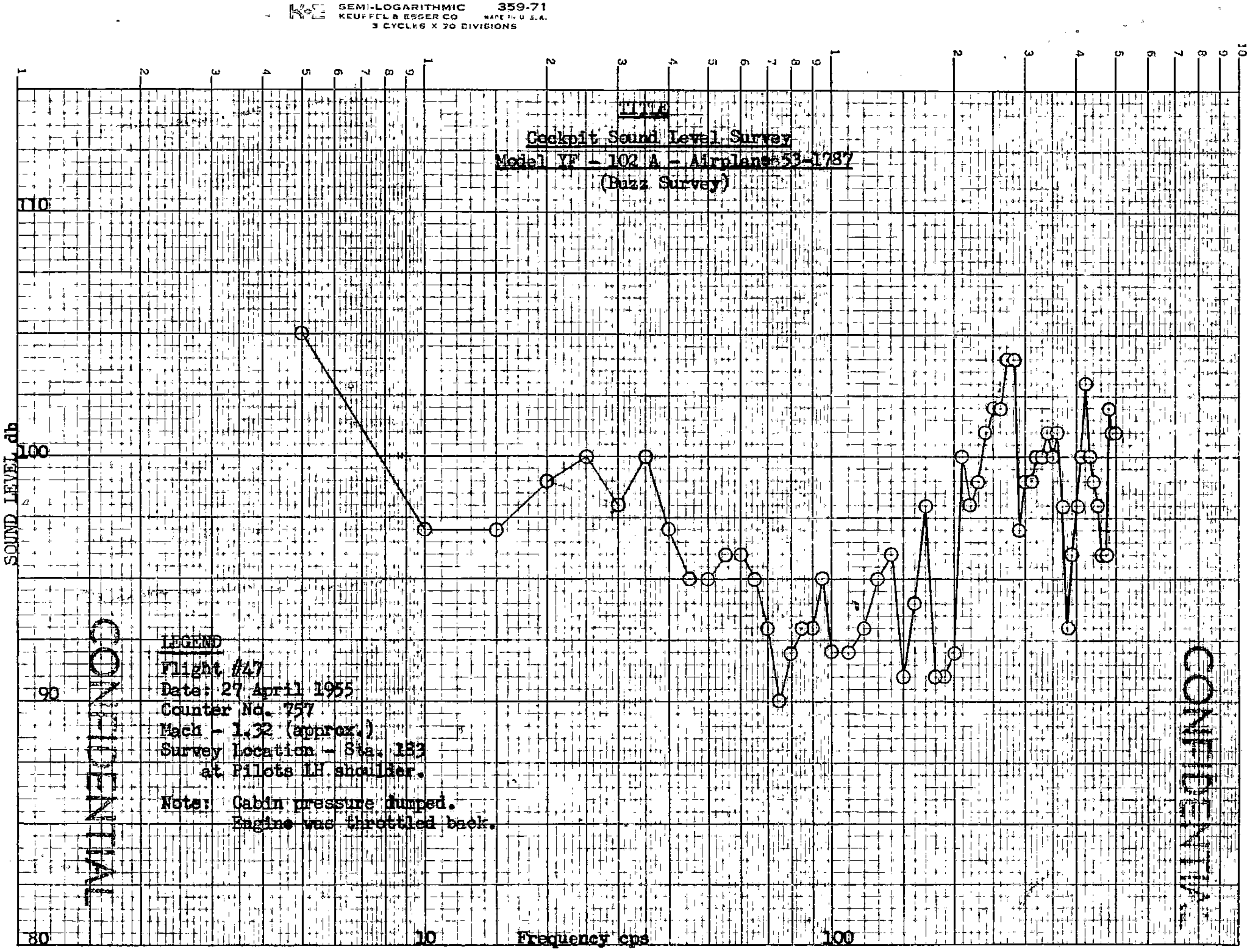
Counter No. 753

Mach - 1.31 (approx.)

Survey Location - Sta. 183
at Pilots LH shoulder.

CONFIDENTIAL

Fig 37



TITLE

Cockpit Sound Level Survey

Model YF - 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND

Flight #47
Date: 27 April 1955
Counter No. 757
Mach - 1.32 (approx.)
Survey Location - Sta. 183
at Pilots LH shoulder.

Note: Cabin pressure dumped.
Engine was throttled back.

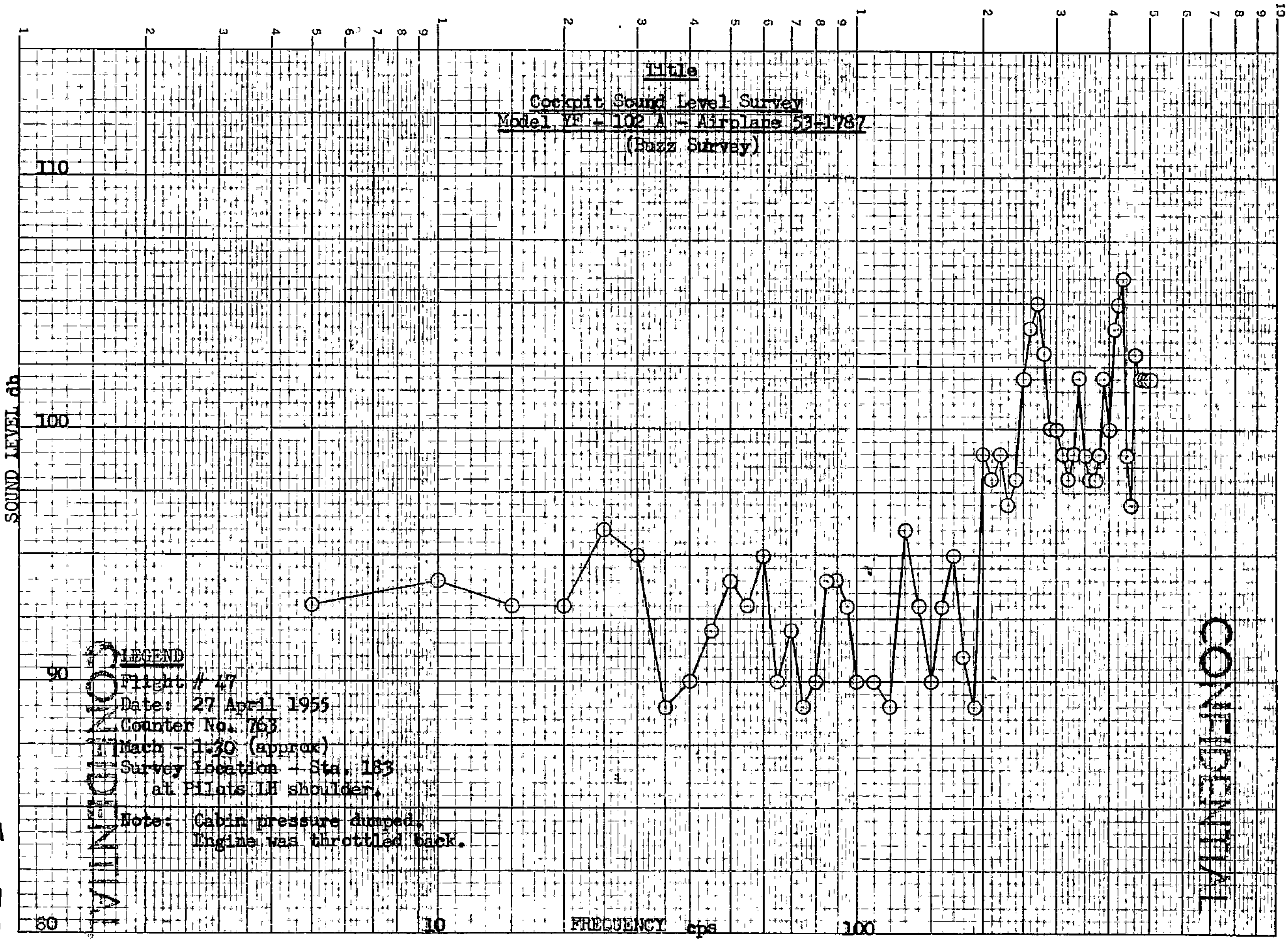
CONFIDENTIAL

CONFIDENTIAL

Fig. 38

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Title

Cockpit Sound Level Survey
Model YF - 102 A - Airplane 53-1787
(Buzz Survey)

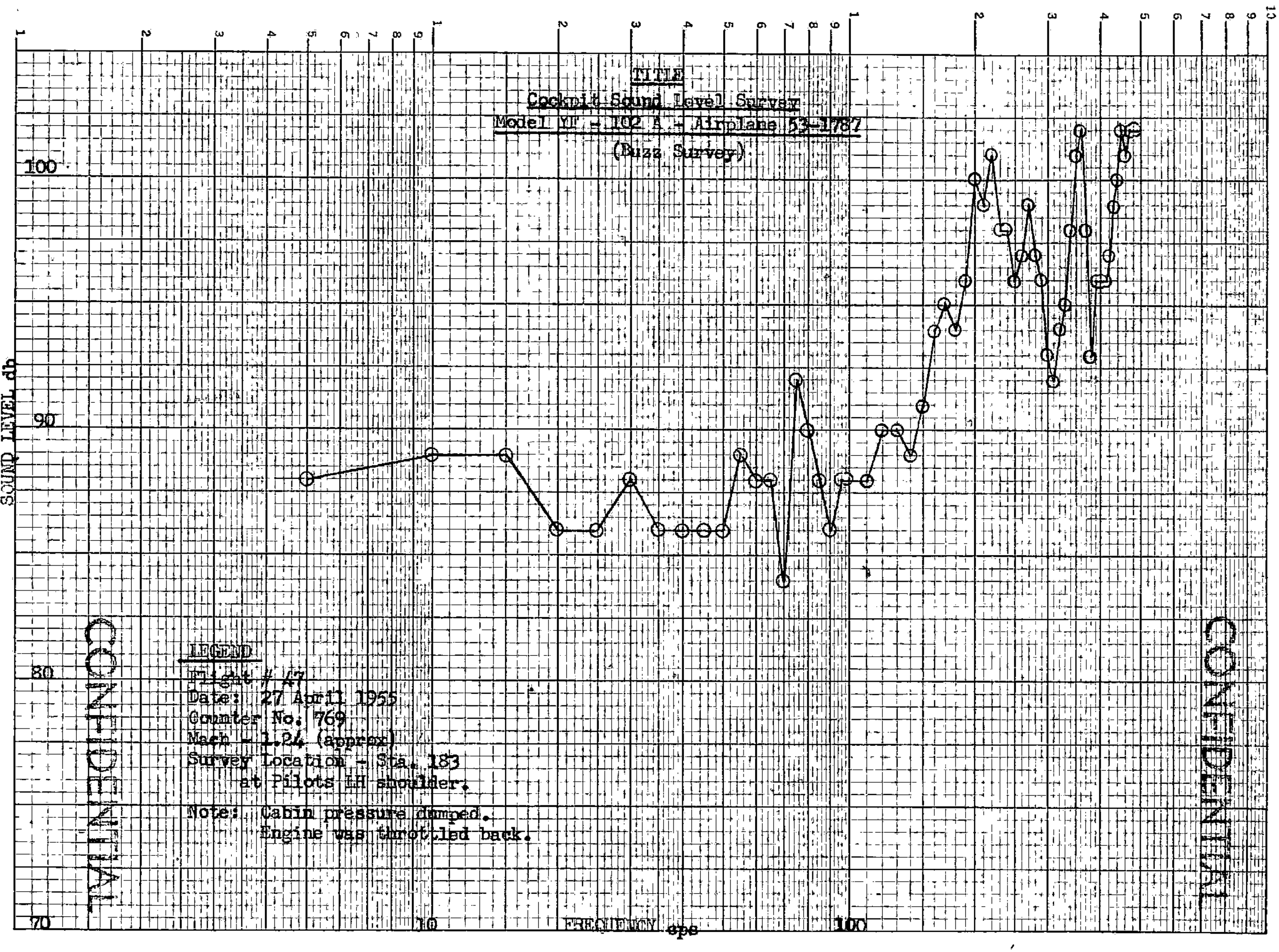
LEGEND
○ Flight # 47
Date: 27 April 1955
Counter No. 763
Mach - 1.30 (approx)
Survey Location - Sta. 183
at Pilots LH shoulder.
Note: Cabin pressure dumped.
Engine was throttled back.

CONFIDENTIAL

Fig.39

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Cockpit Sound Level Survey

Model YF - 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND

Flight # 47
Date: 27 April 1955
Counter No. 769
Mach - 1.24 (approx)
Survey Location - Sta. 183
at Pilots LH shoulder.

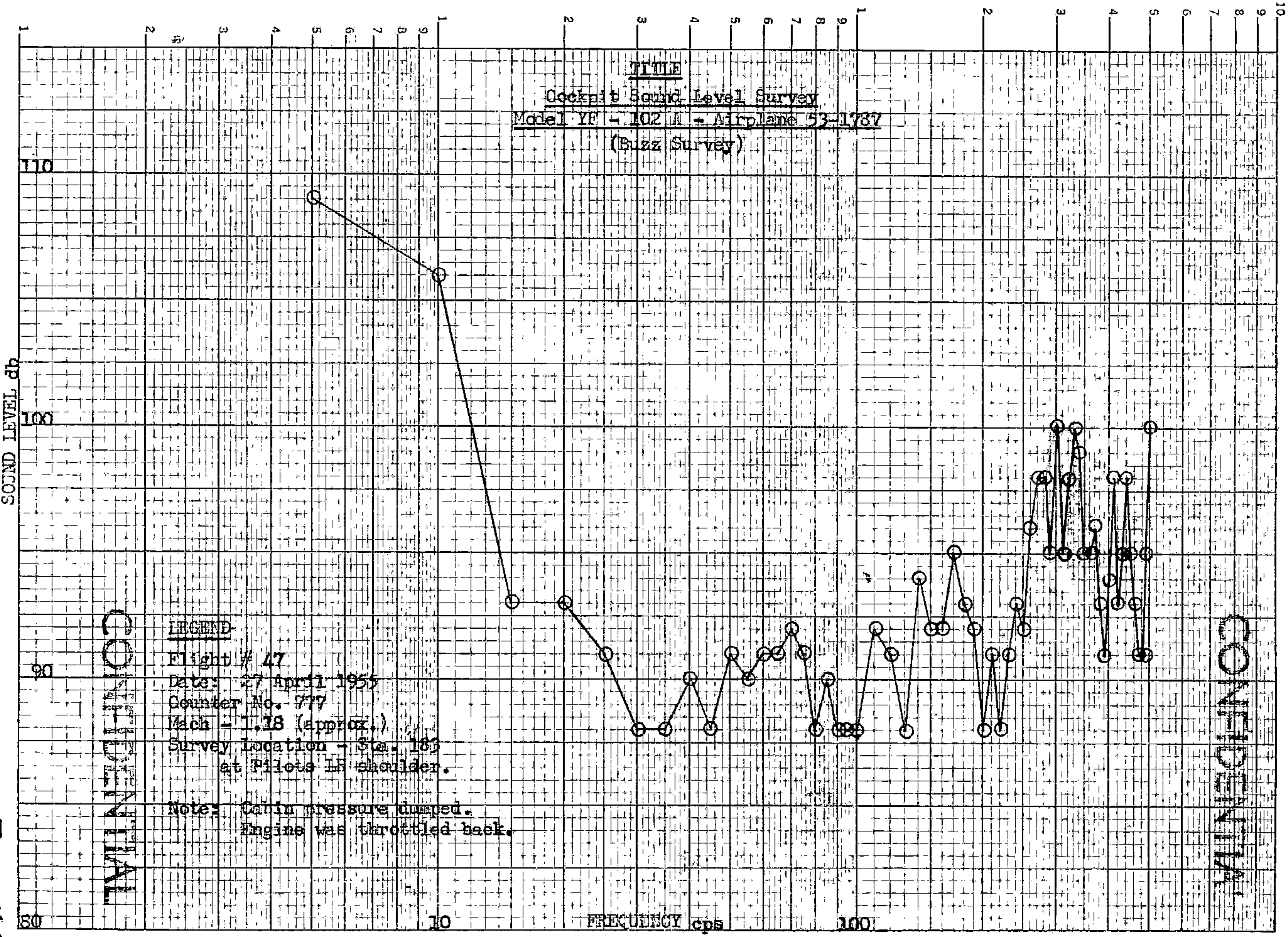
Note: Cabin pressure dumped.
Engine was throttled back.

CONFIDENTIAL

Fig. 40

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Cockpit Sound Level Survey
Model YF - 102 A - Airplane 53-1787
(Buzz Survey)

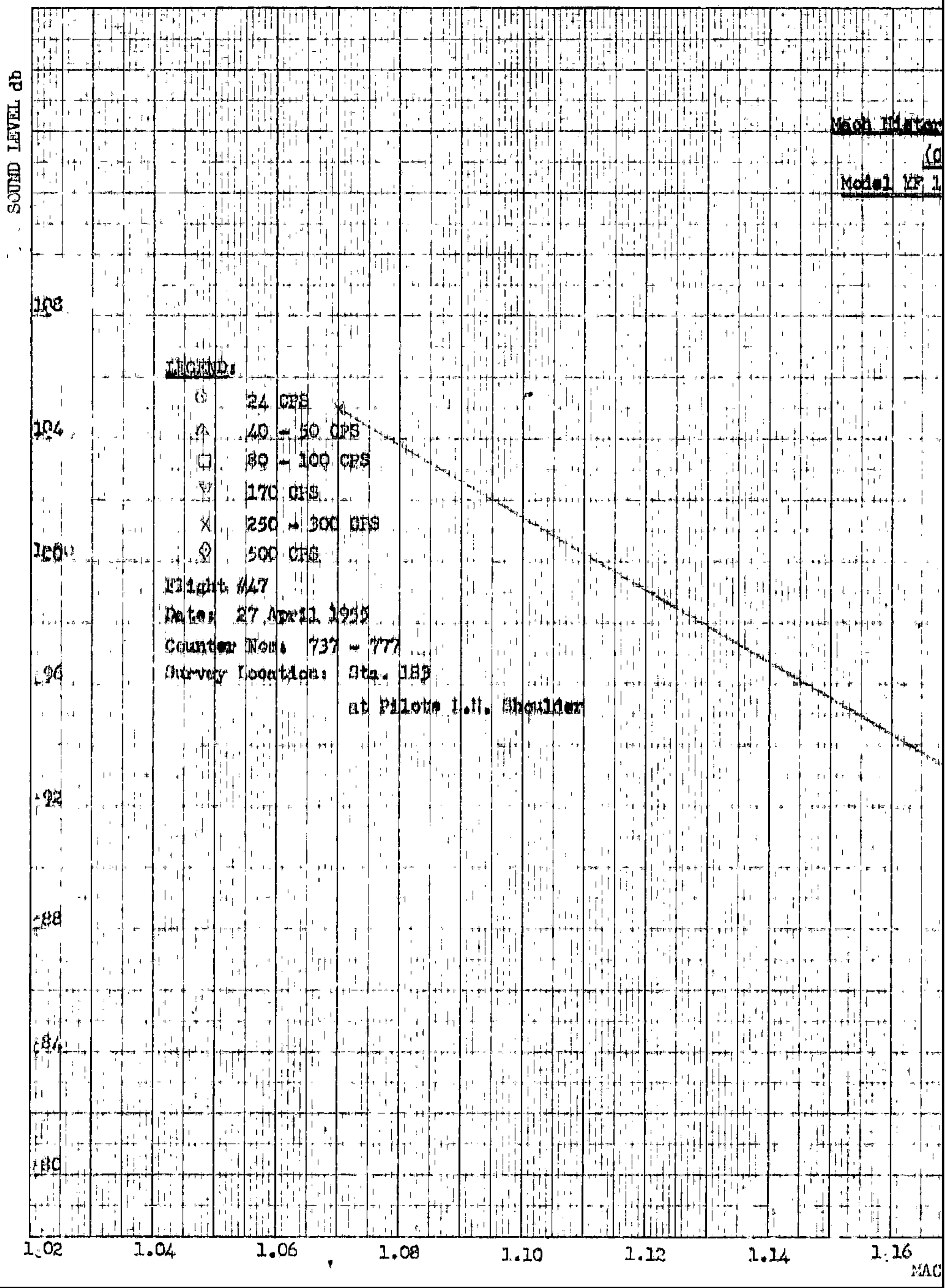
LEGEND

Flight # 47
Date: 27 April 1955
Counter No. 777
Mach - 1.18 (approx.)
Survey Location - Sta. 183
at Pilots LH shoulder.

Note: Cabin pressure dumped.
Engine was throttled back.

CONFIDENTIAL

Fig. 41

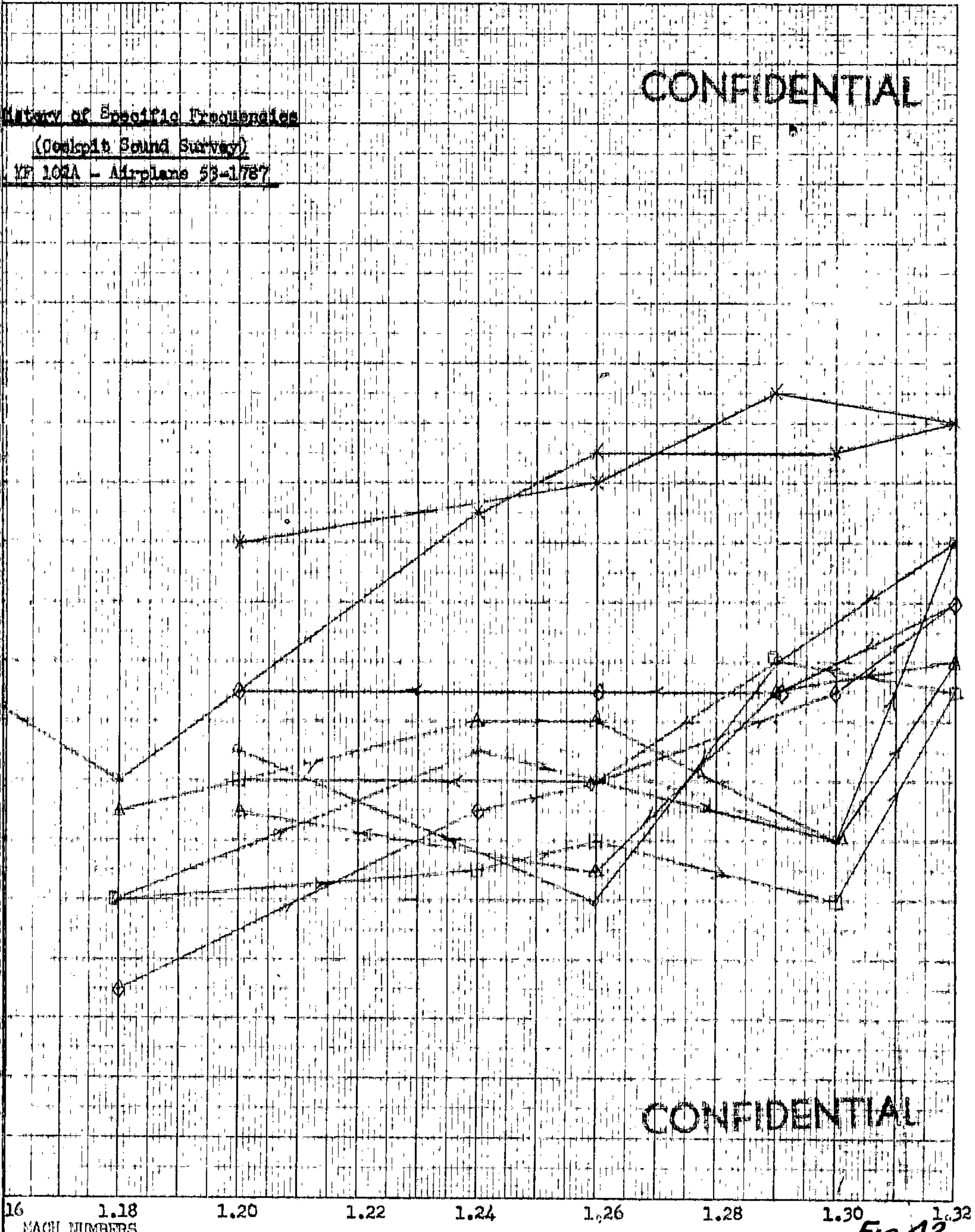

Mach Histor
(c
Model XF 1
SOUND LEVEL db
108
104
100
96
92
88
84
80
1.02
1.04
1.06
1.08
1.10
1.12
1.14
1.16
MAC
LEGEND:
24 CPS
40 - 50 CPS
80 - 100 CPS
170 CPS
250 - 300 CPS
500 CPS
Flight #47
Date: 27 April 1955
Counter Nos. 737 - 777
Survey Location: Sta. 185
at Pilots L.H. Shoulder

CONFIDENTIAL

History of Specific Frequencies

(Cockpit Sound Survey)

YF 102A - Airplane 53-1787

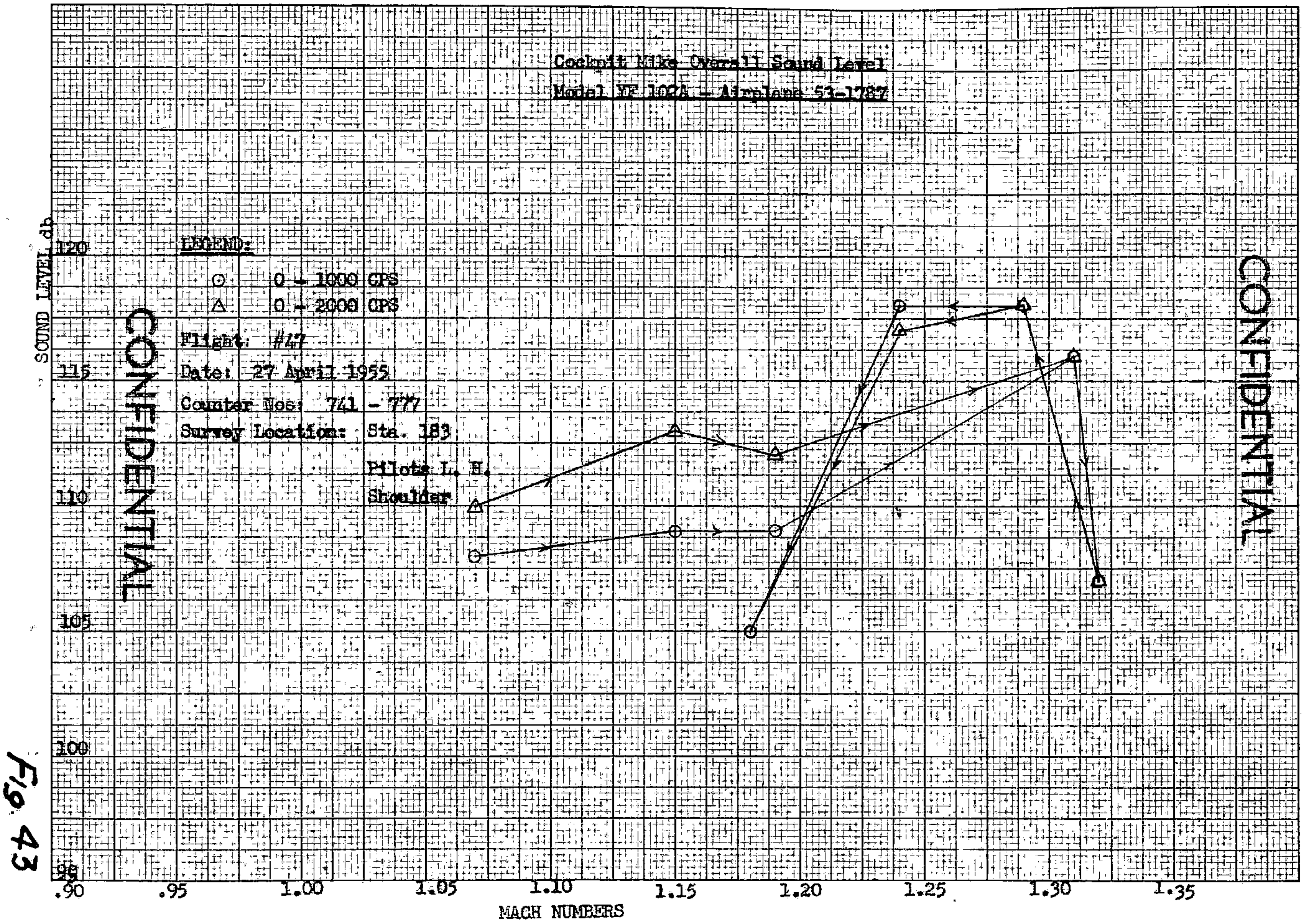


CONFIDENTIAL

Fig. 42

K+E 10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO. MADE IN U.S.A.

CONFIDENTIAL

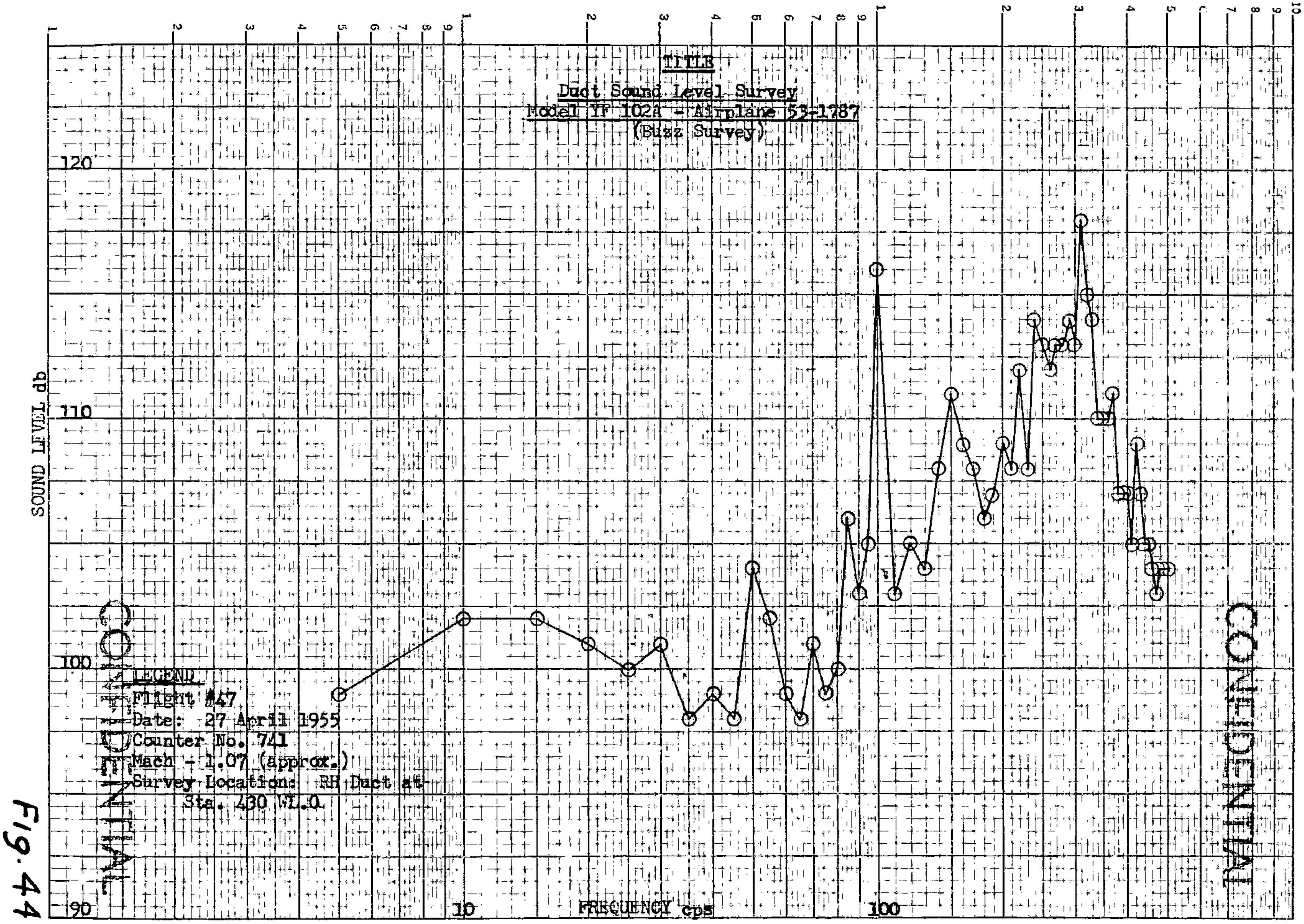


CONFIDENTIAL

Fig. 43

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL

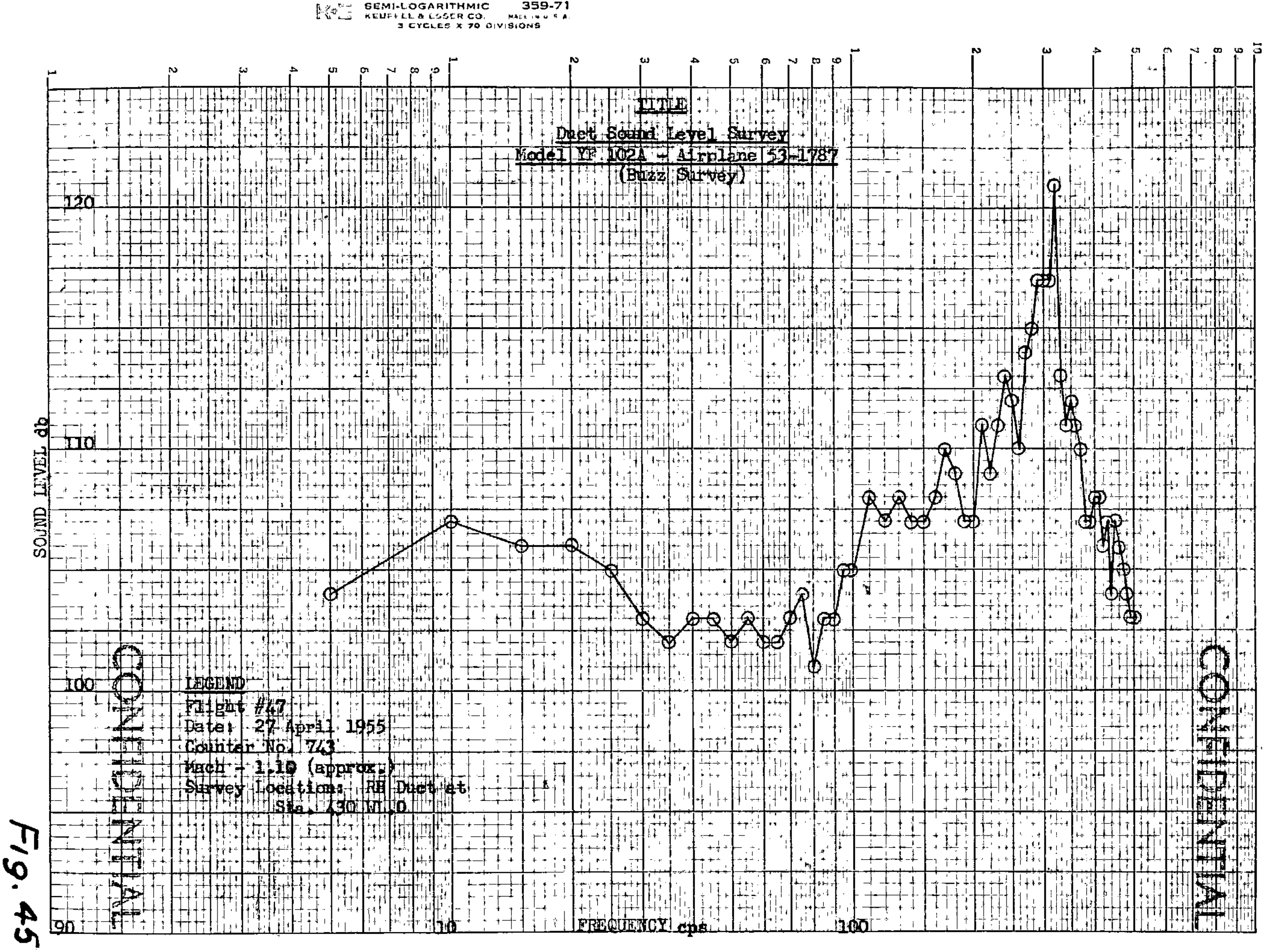


TITLE
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 741
Mach - 1.07 (approx.)
Survey Location: RH Duct at
Sta. 430 WL.0

Fig. 44

CONFIDENTIAL

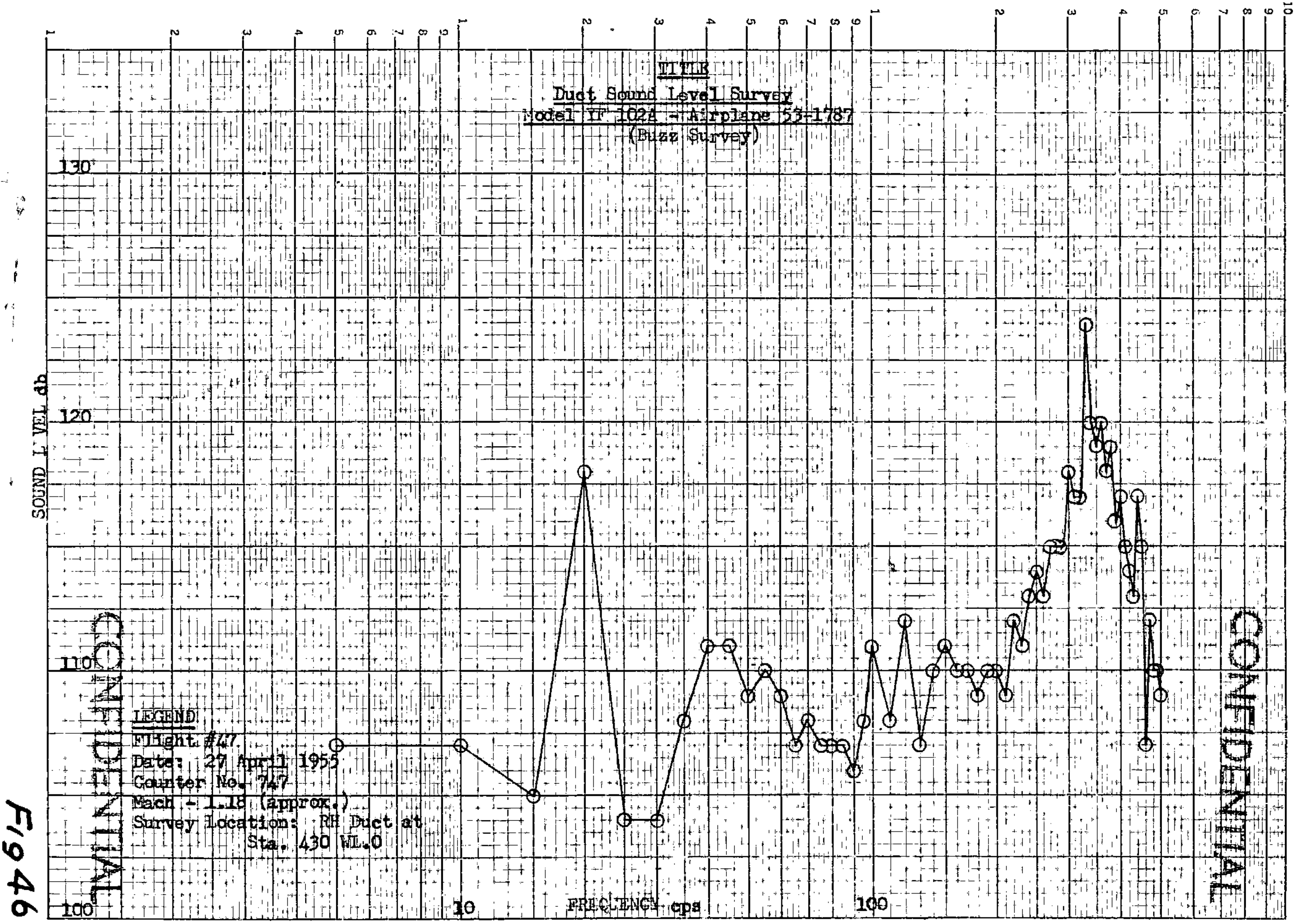


CONFIDENTIAL

Fig. 45

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

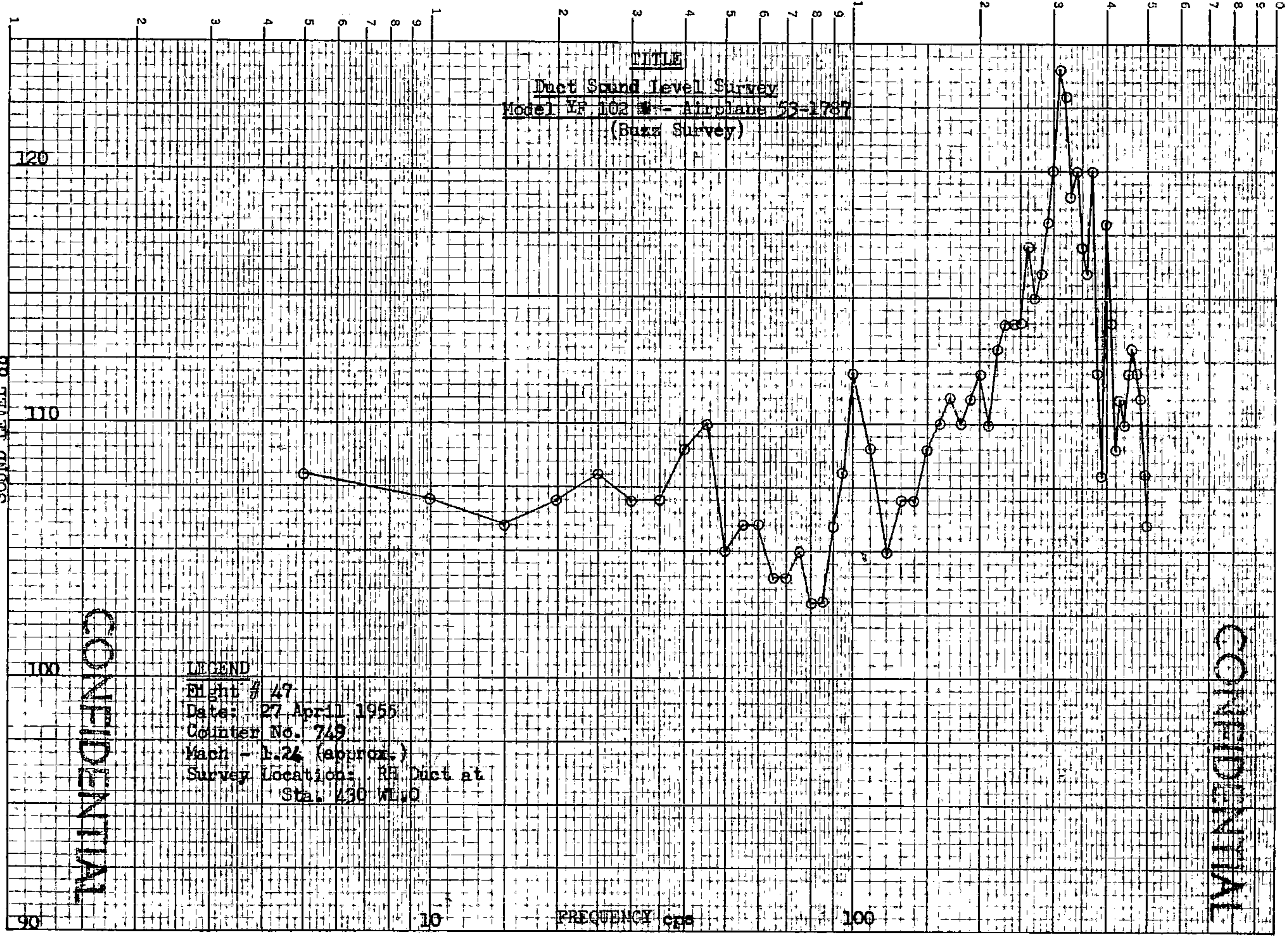
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 747
Mach - 1.18 (approx.)
Survey Location: RH Duct at
Sta. 430 WL.0

CONFIDENTIAL

Fig 46

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

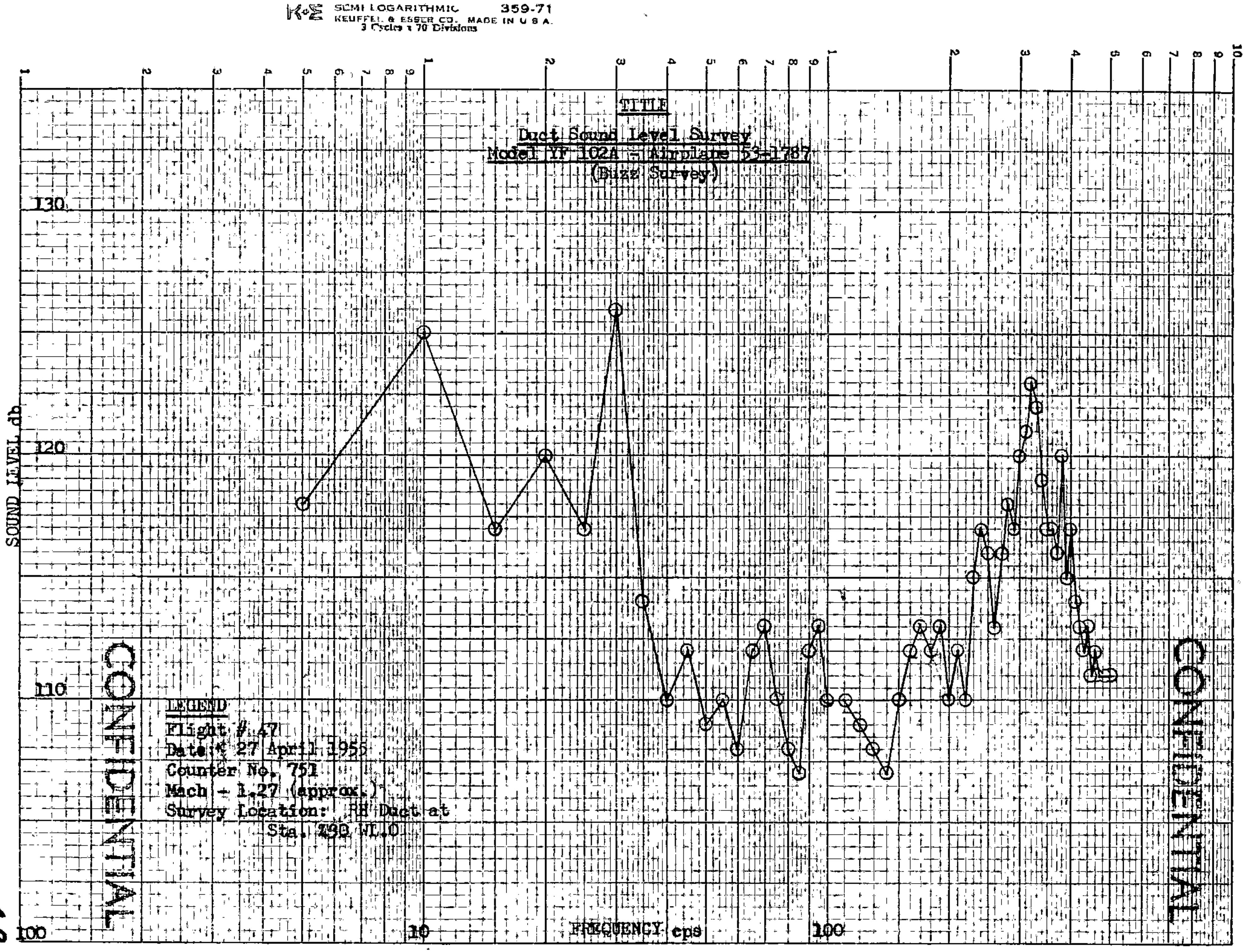
Duct Sound Level Survey
Model YF 102 [illegible] - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 749
Mach - 1.24 (approx.)
Survey Location: RH Duct at
Sta. 430 WL.0

CONFIDENTIAL

Fig. 47

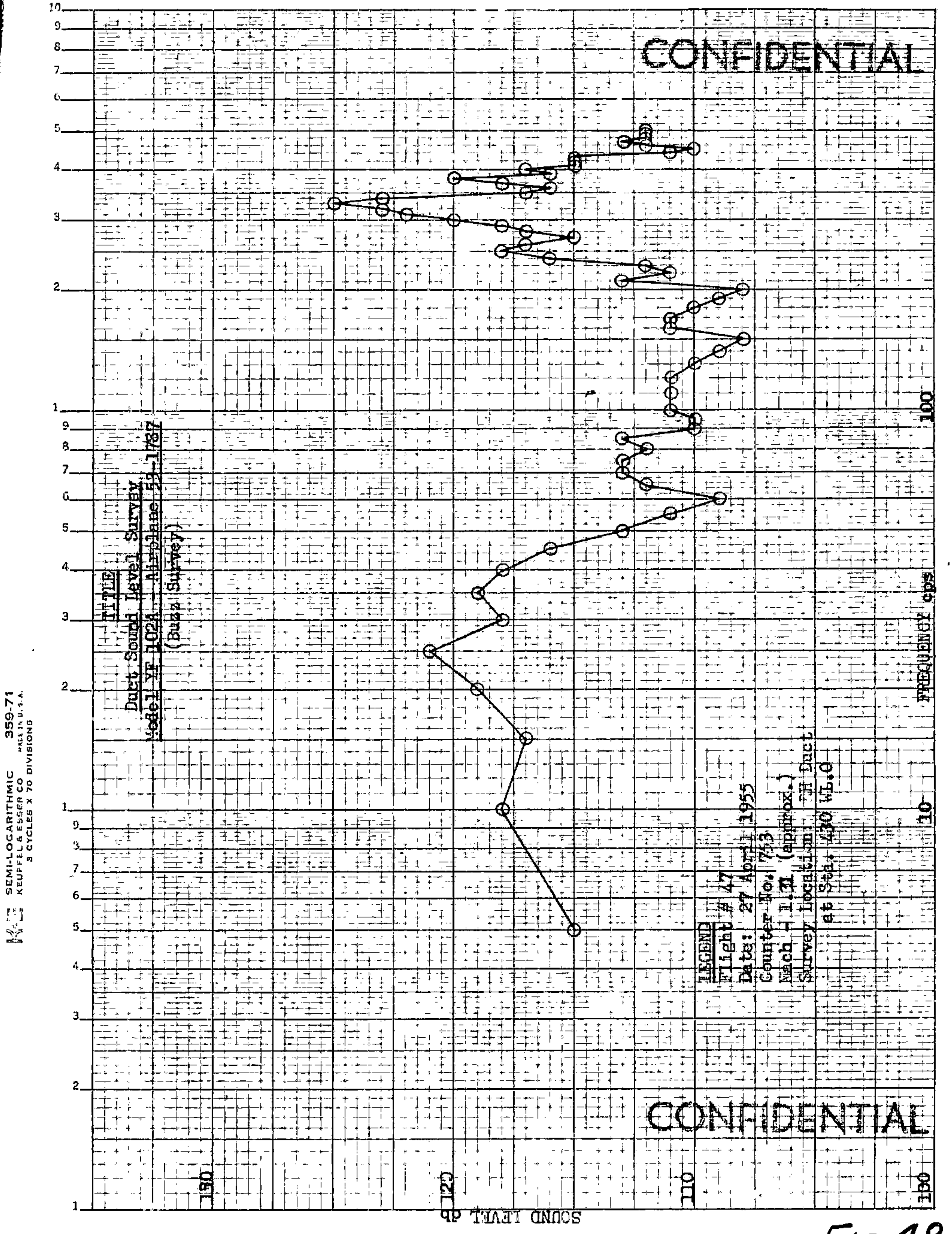
CONFIDENTIAL



CONFIDENTIAL

Fig. 48

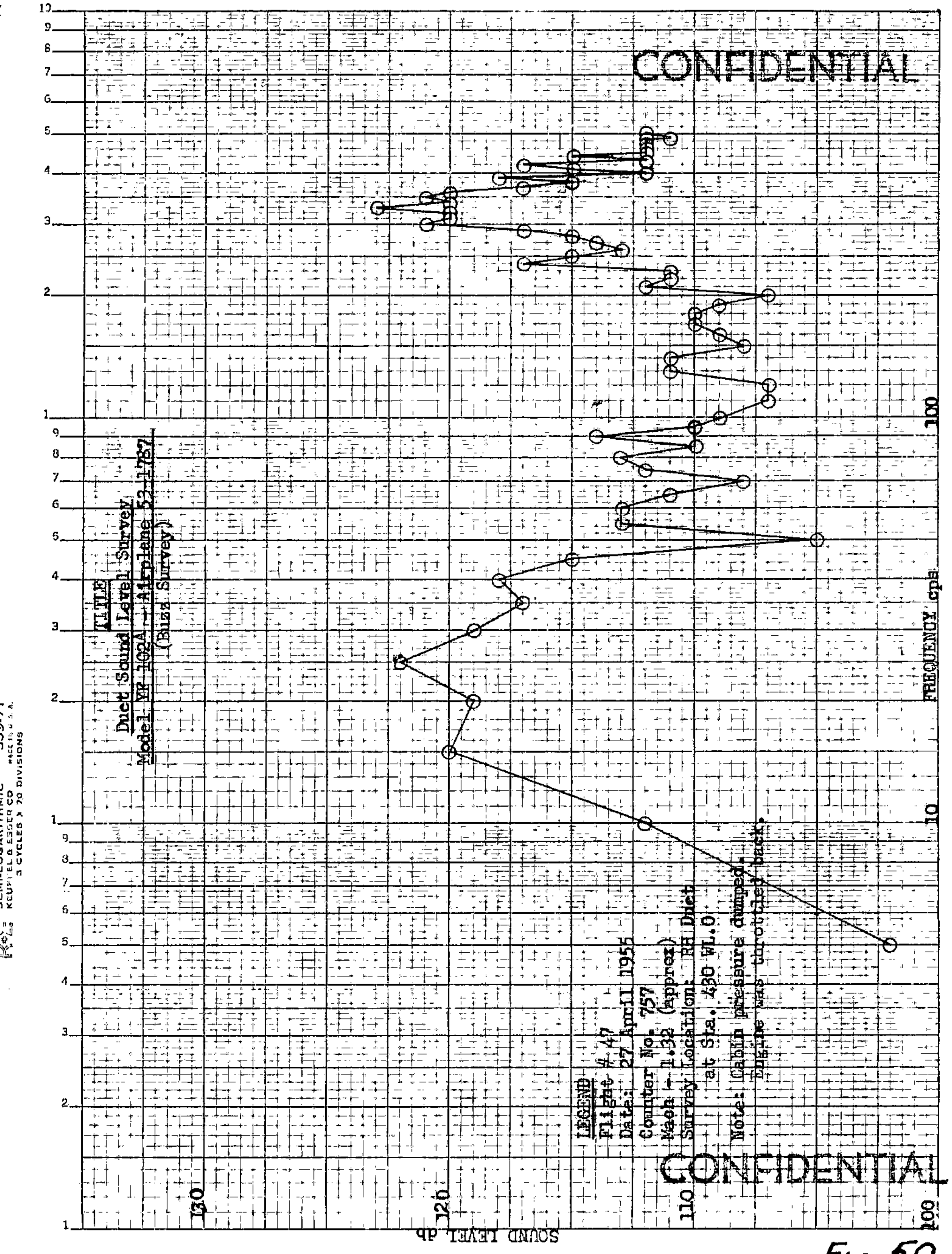
CONFIDENTIAL



CONFIDENTIAL

Fig. 49

CONFIDENTIAL



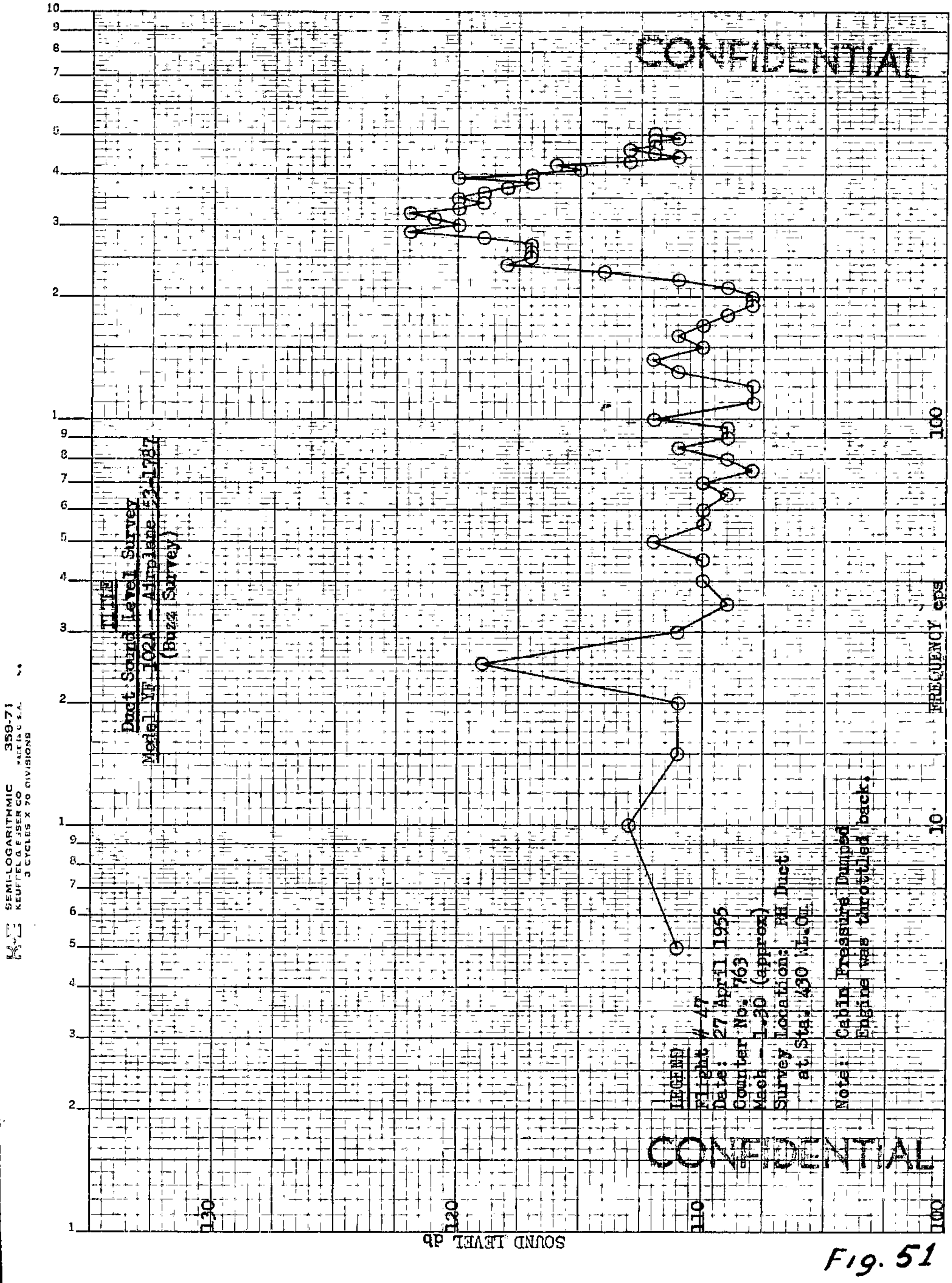
TITLE
Duct Sound Level Survey
Model YF 102A – Airplane 52-1757
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 757
Mach – 1.32 (approx)
Survey Location: RH Duct
at Sta. 430 WL.0

Note: Cabin pressure dumped.
Engine was throttled back.

CONFIDENTIAL

Fig. 50



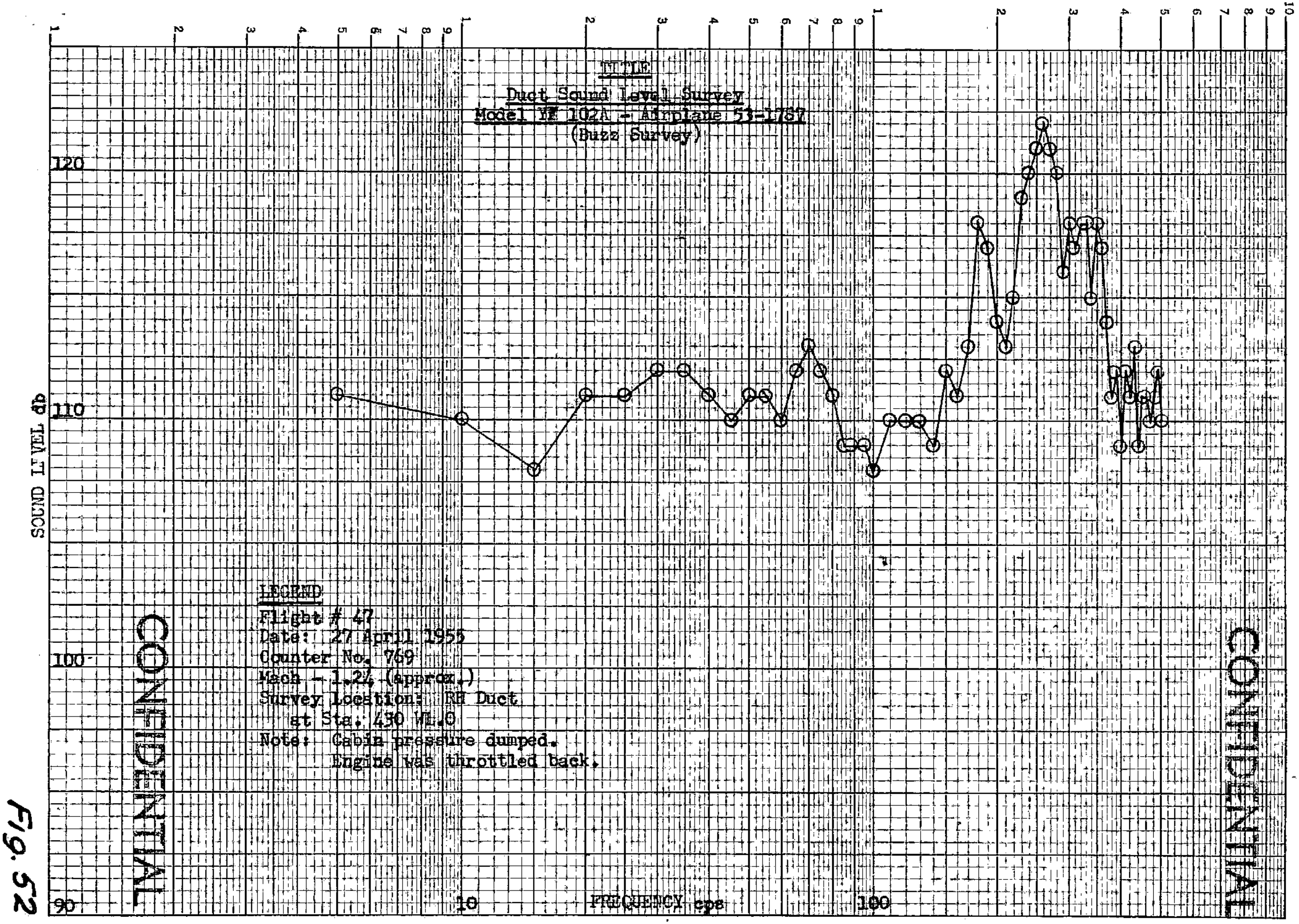
TITLE
Duct Sound Level Survey
Model YF 102A — Airplane 53-1787
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 763
Mach — 1.30 (approx)
Survey Location: RH Duct
at Sta. 430 WL.On.

Note: Cabin Pressure Dumped
Engine was throttled back.

Fig. 51

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

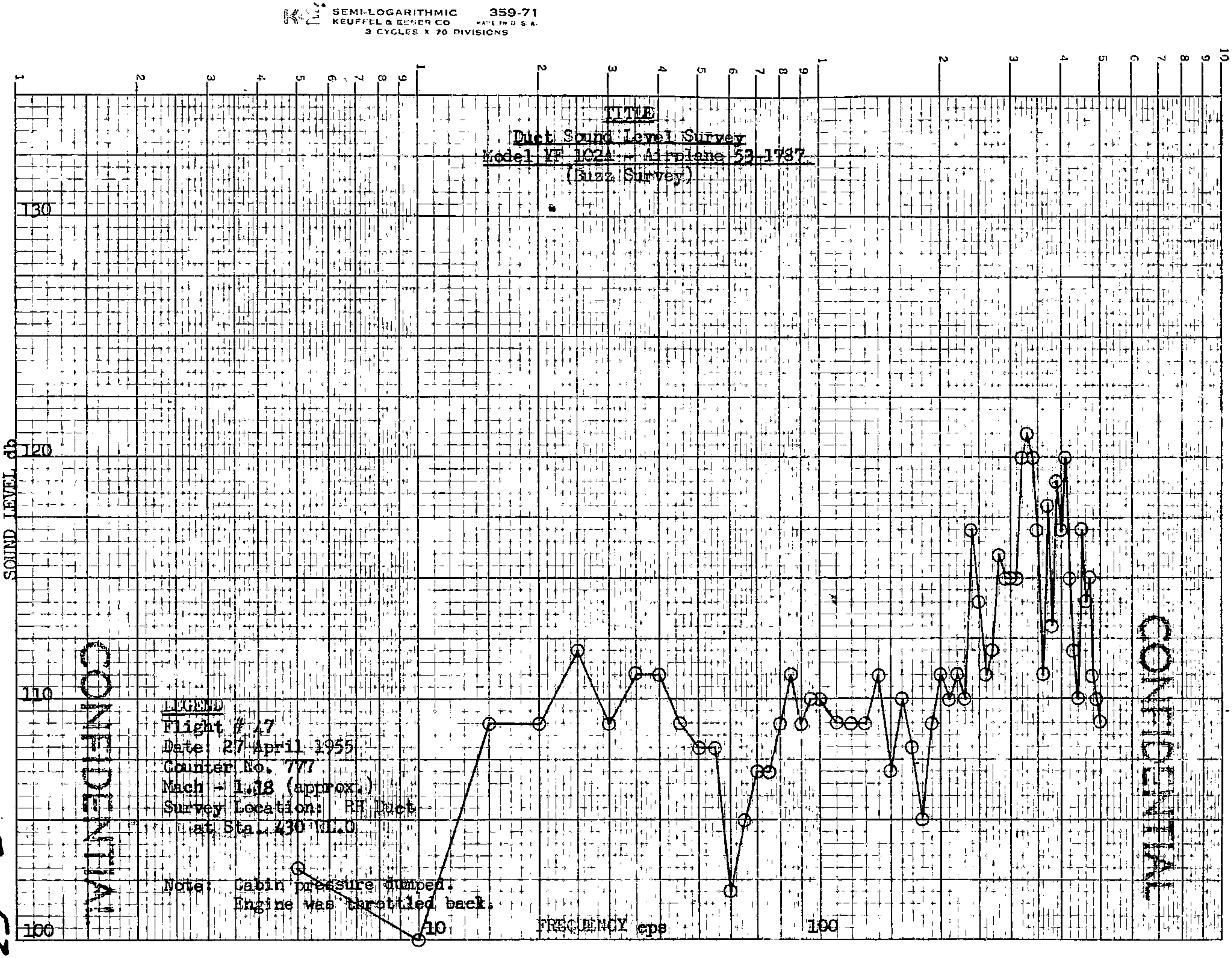


TITLE

Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 769
Mach - 1.24 (approx.)
Survey Location: RH Duct
at Sta. 430 WL.0
Note: Cabin pressure dumped.
Engine was throttled back.

Fig. 52



TITLE

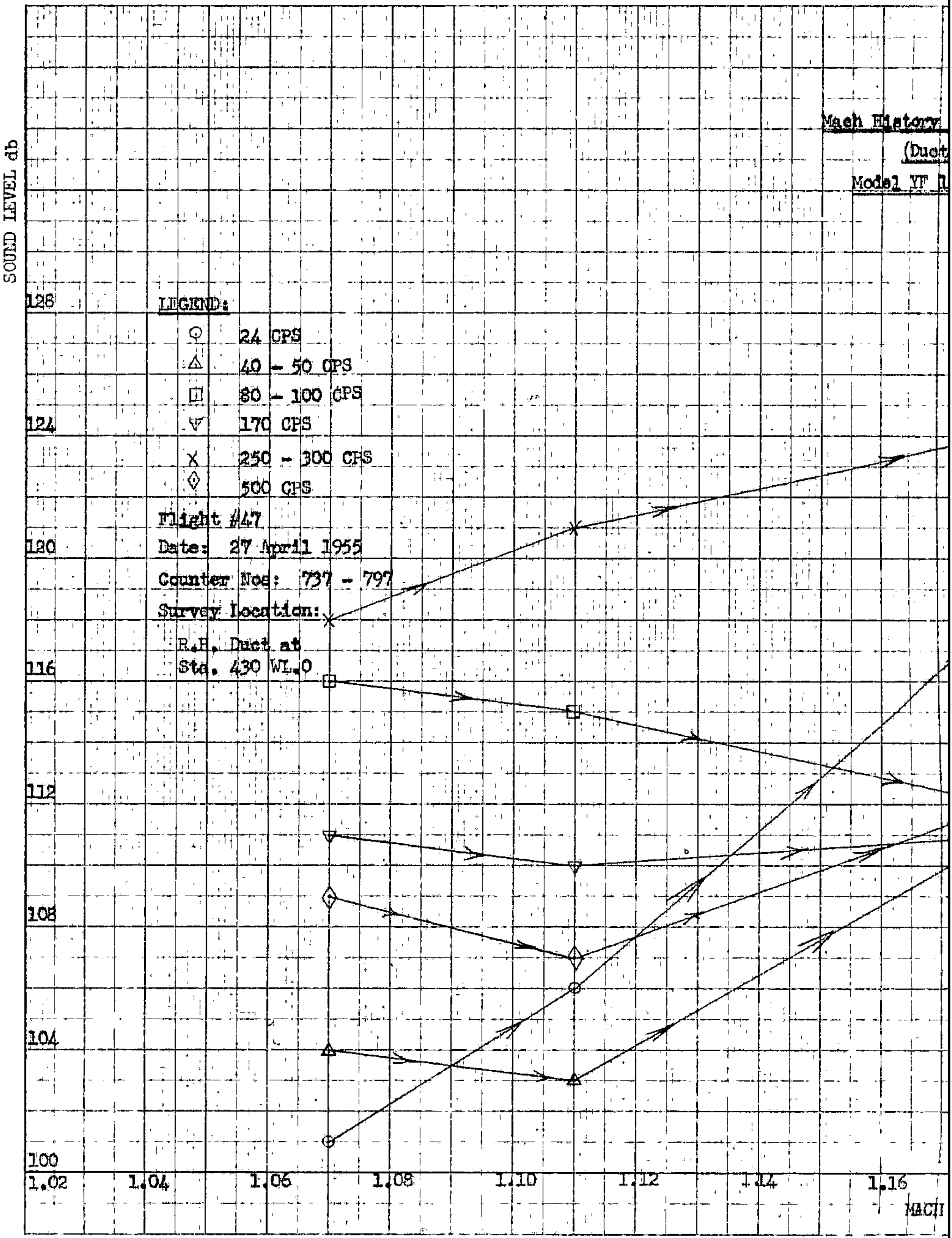
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND

Flight # 47
Date: 27 April 1955
Counter No. 777
Mach - 1.18 (approx.)
Survey Location: RH Duct
at Sta. 430 WL.0

Note: Cabin pressure dumped.
Engine was throttled back.

Fig. 53



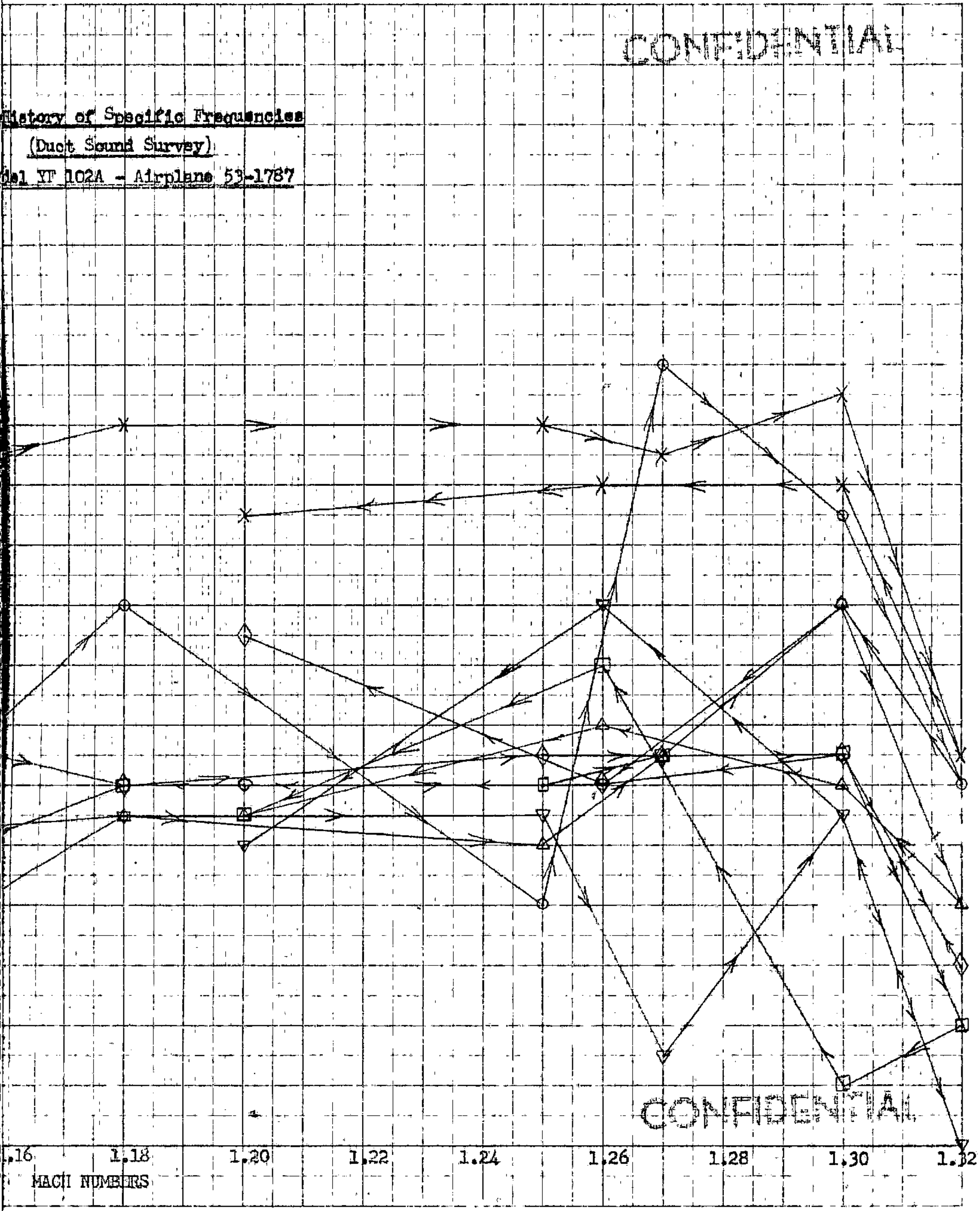
10 X 10 TO THE ½ INCH 359-11L

CONFIDENTIAL

istory of Specific Frequencies

(Duct Sound Survey)

del YF 102A - Airplane 53-1787

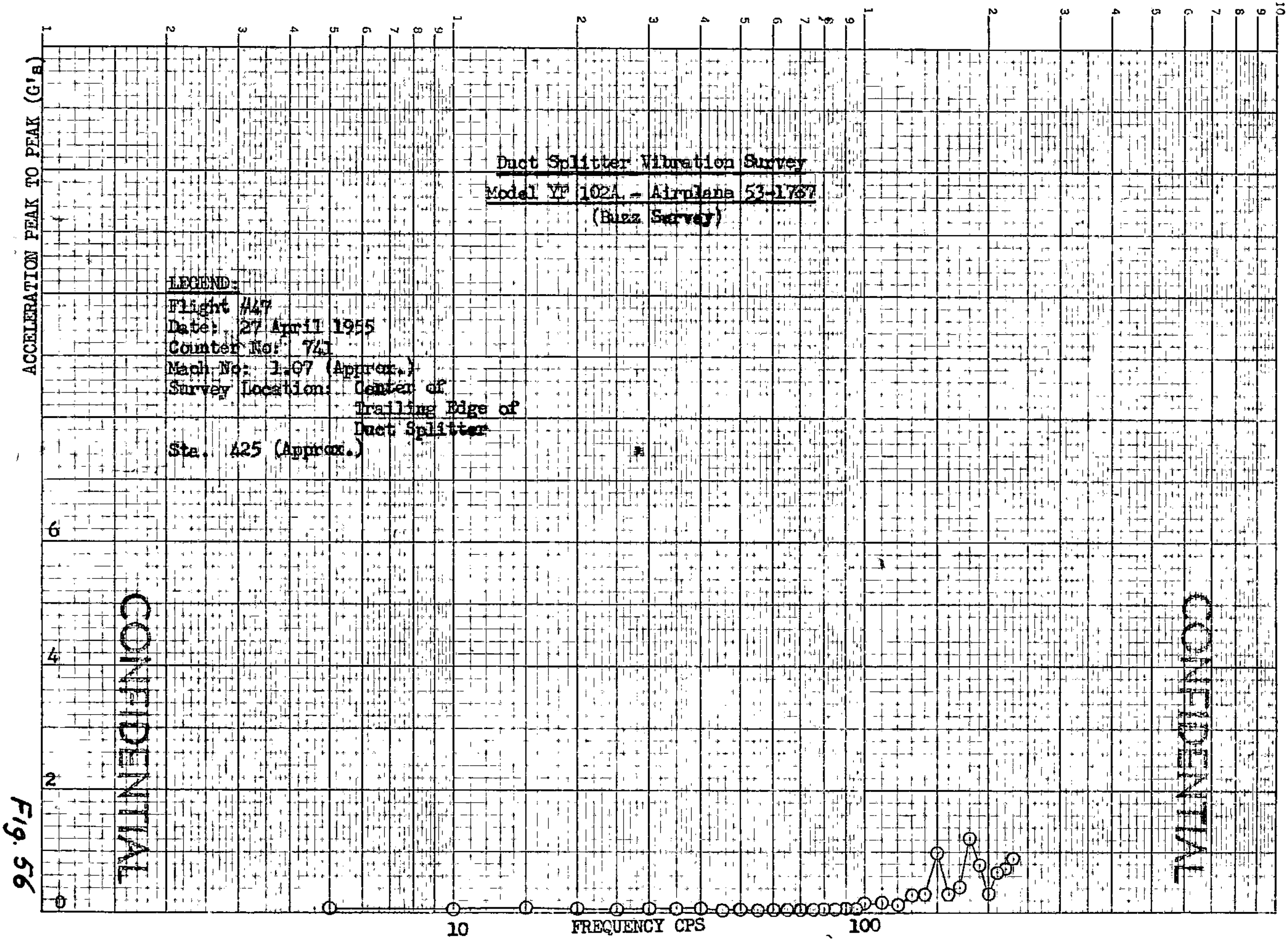


CONFIDENTIAL

Fig. 54

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL

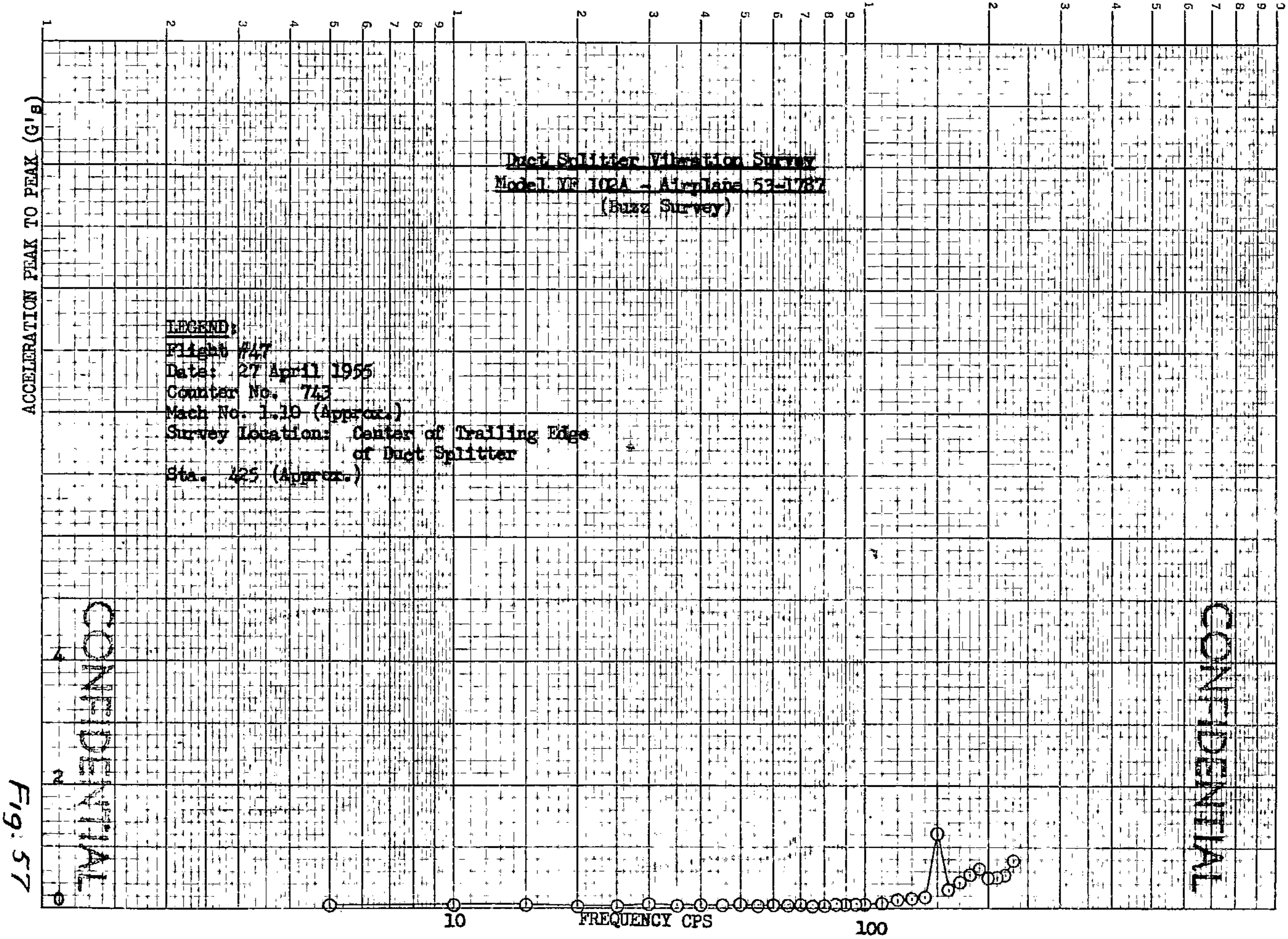


CONFIDENTIAL

Fig. 56

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

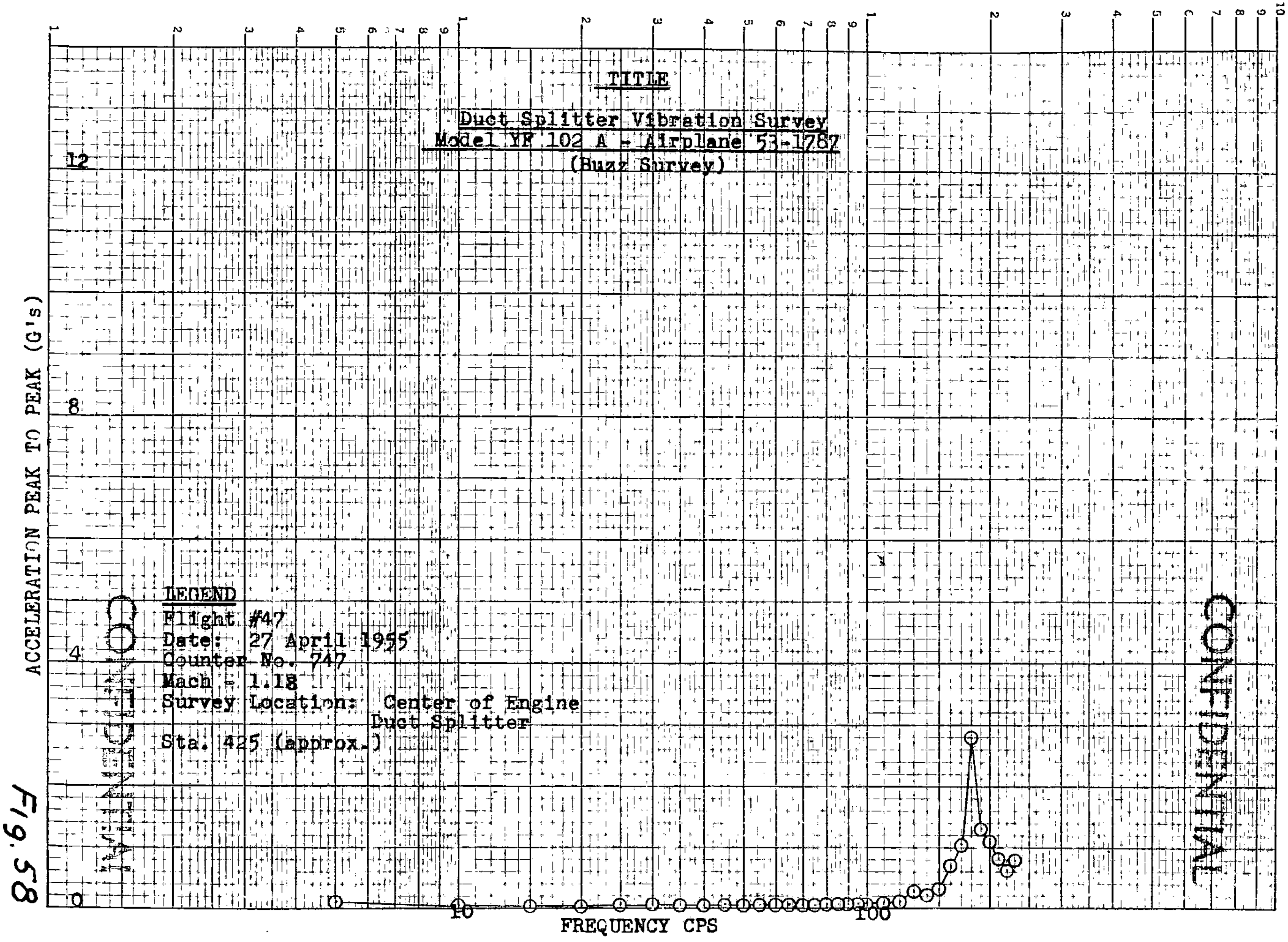
LEGEND:
Flight #47
Date: 27 April 1955
Counter No. 743
Mach No. 1.19 (Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta. 425 (Approx.)

CONFIDENTIAL

Fig. 57

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

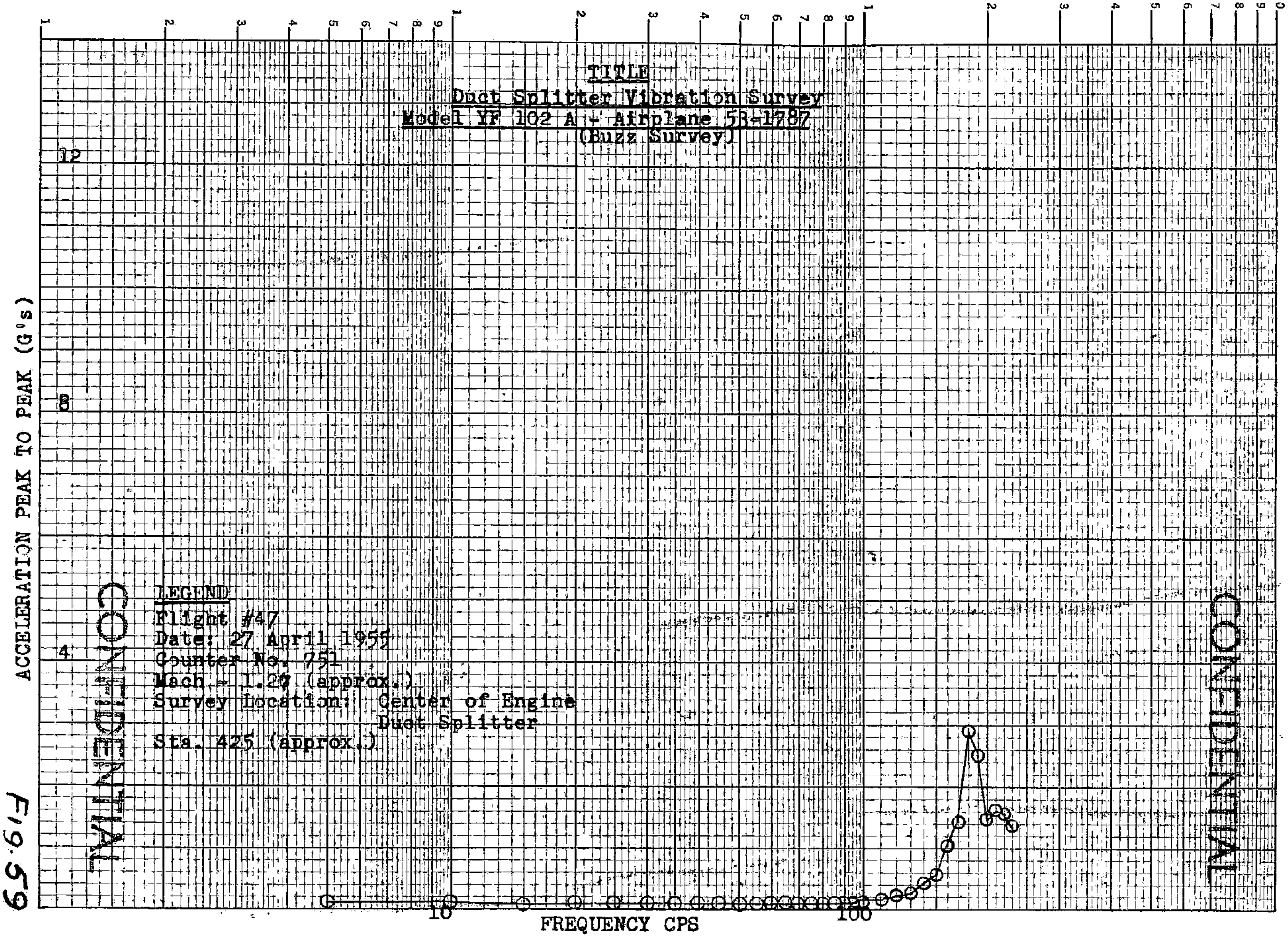
Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 747
Mach - 1.18
Survey Location: Center of Engine
Duct Splitter
Sta. 425 (approx.)

CONFIDENTIAL

Fig. 58

CONFIDENTIAL



TITLE

Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND

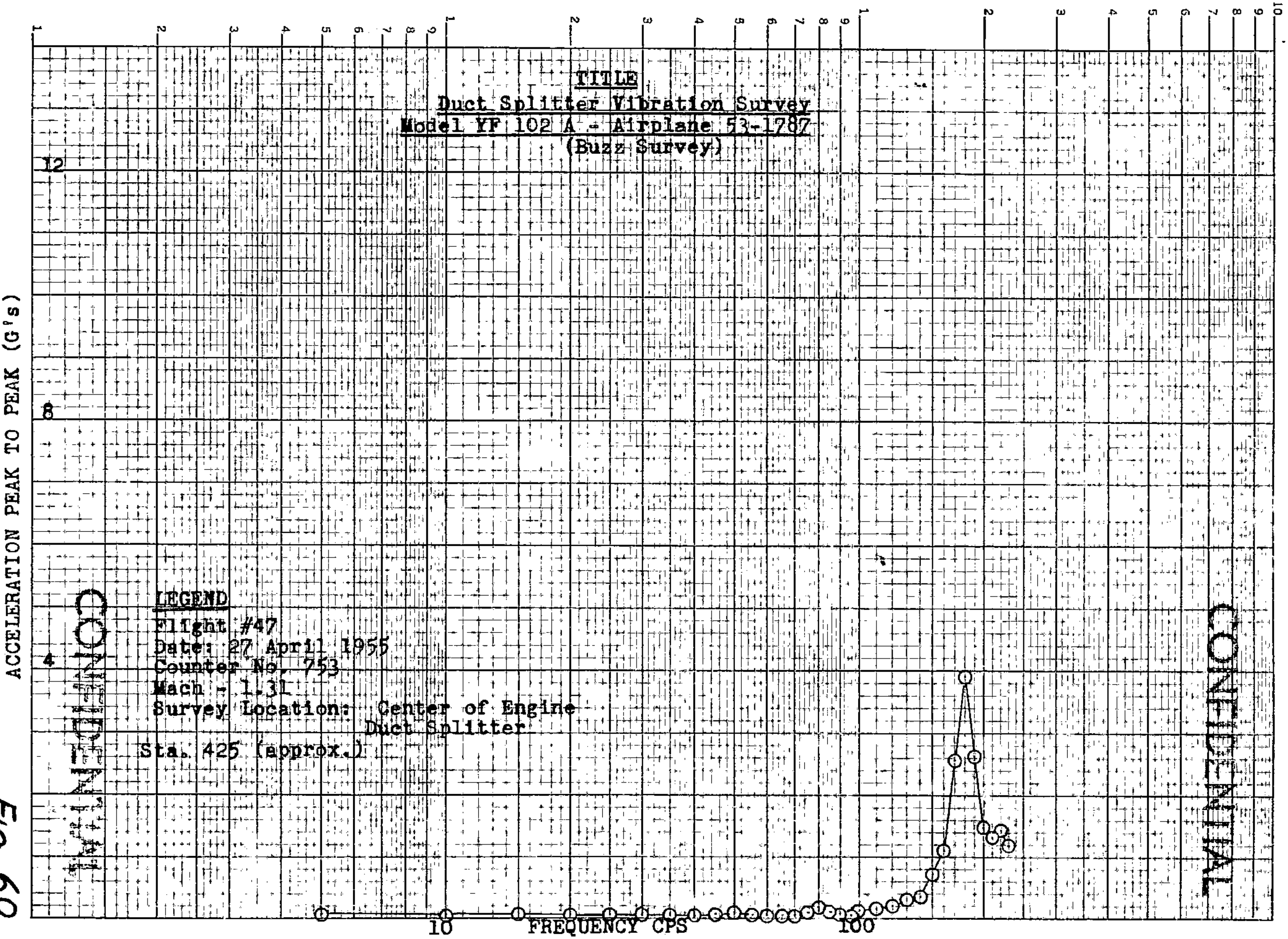
Flight #47
Date: 27 April 1955
Counter No. 751
Mach - 1.27 (approx.)
Survey Location: Center of Engine
Duct Splitter
Sta. 425 (approx.)

CONFIDENTIAL

Fig. 59

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

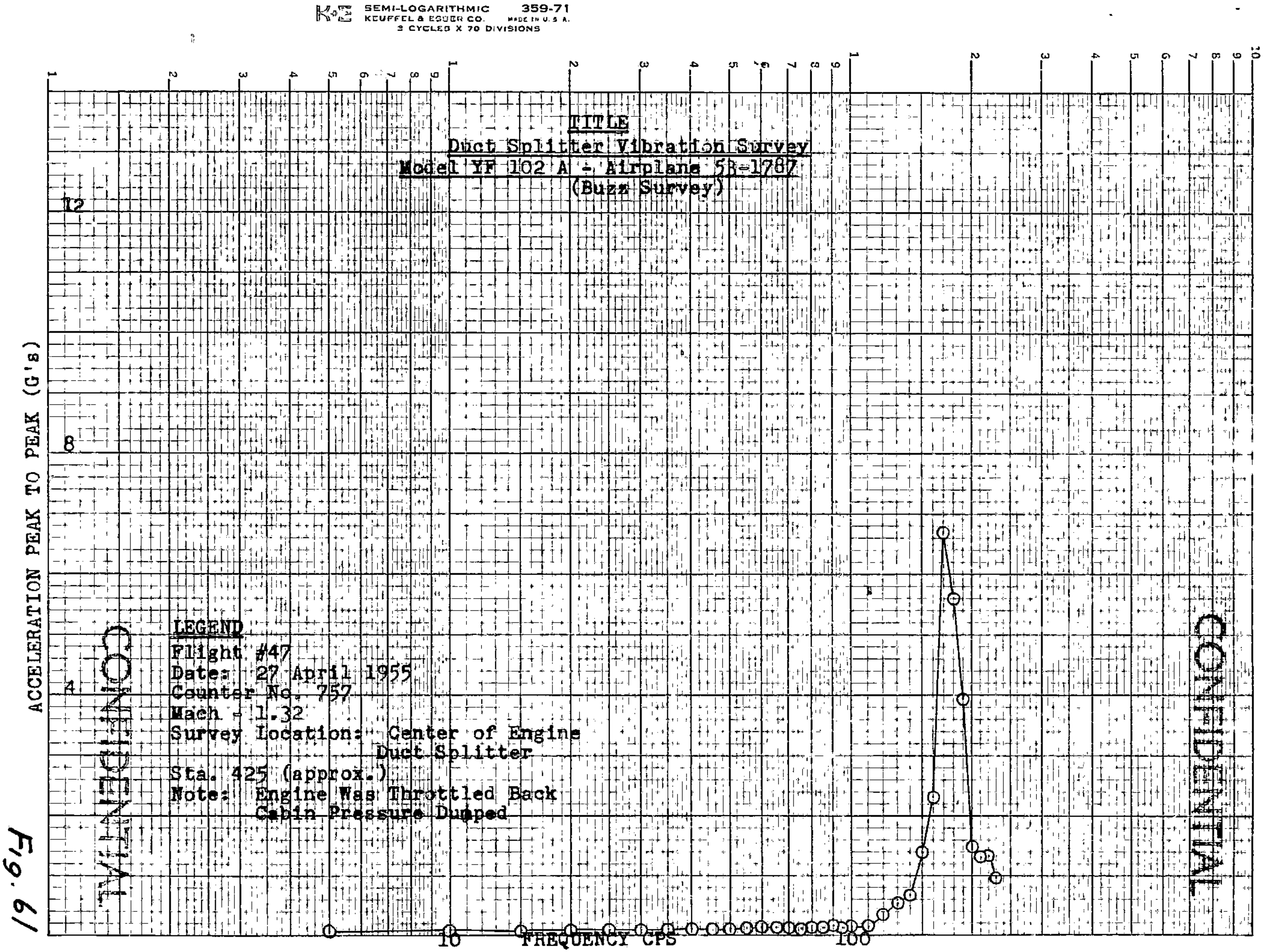
Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 753
Mach - 1.31
Survey Location: Center of Engine Duct Splitter
Sta. 425 (approx.)

CONFIDENTIAL

Fig. 60

CONFIDENTIAL



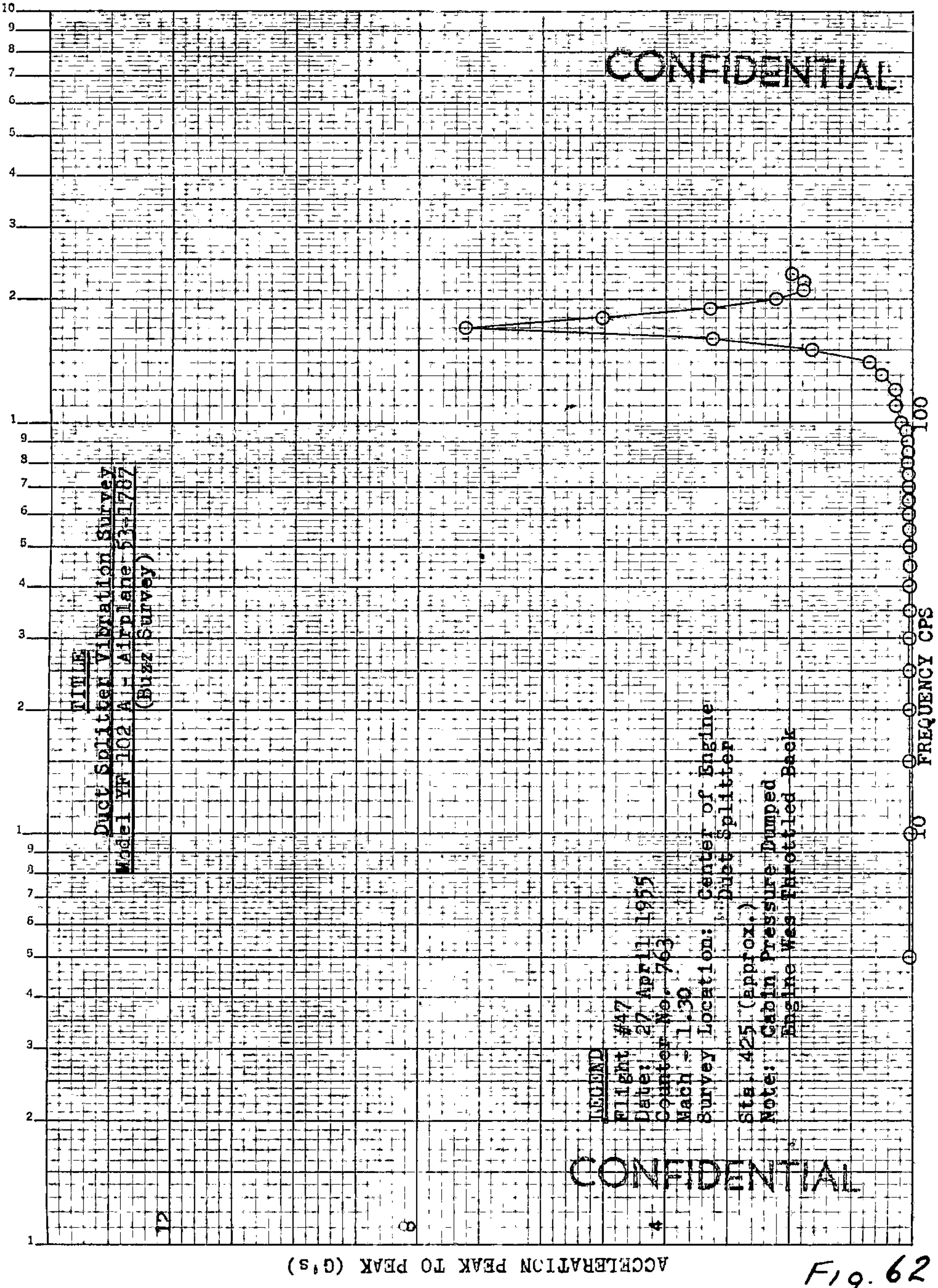
TITLE
Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 757
Mach = 1.32
Survey Location: Center of Engine
Duct Splitter
Sta. 425 (approx.)
Note: Engine Was Throttled Back
Cabin Pressure Dumped

CONFIDENTIAL

Fig. 61

CONFIDENTIAL



TITLE
Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

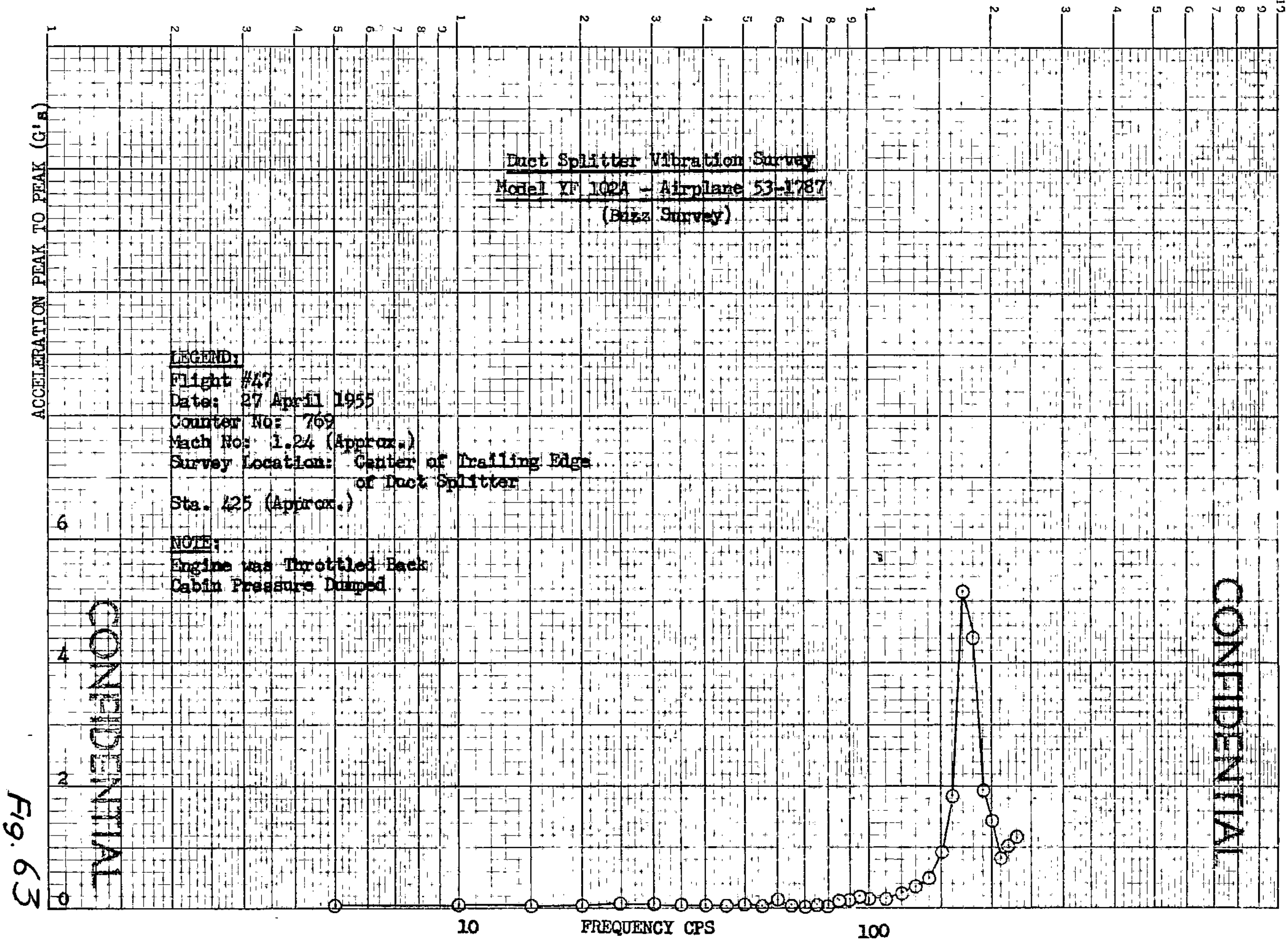
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 763
Mach - 1.30
Survey Location: Center of Engine
Duct Splitter
Sta. 425 (approx.)
Note: Cabin Pressure Dumped
Engine Was Throttled Back

CONFIDENTIAL

Fig. 62

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Splitter Vibration Survey

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 769
Mach No: 1.24 (Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta. 425 (Approx.)

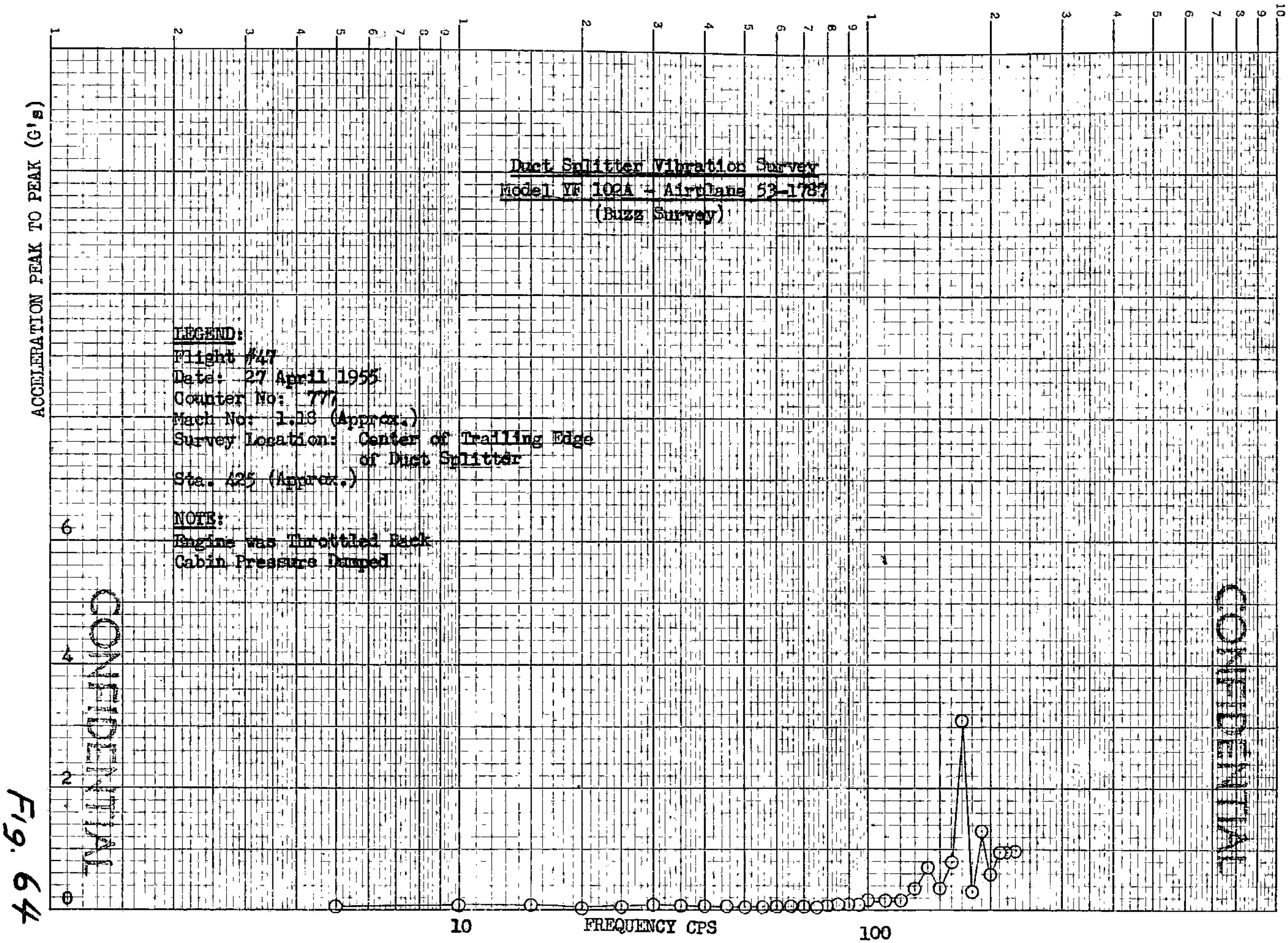
NOTE:
Engine was Throttled Back
Cabin Pressure Dumped

CONFIDENTIAL

Fig. 63

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



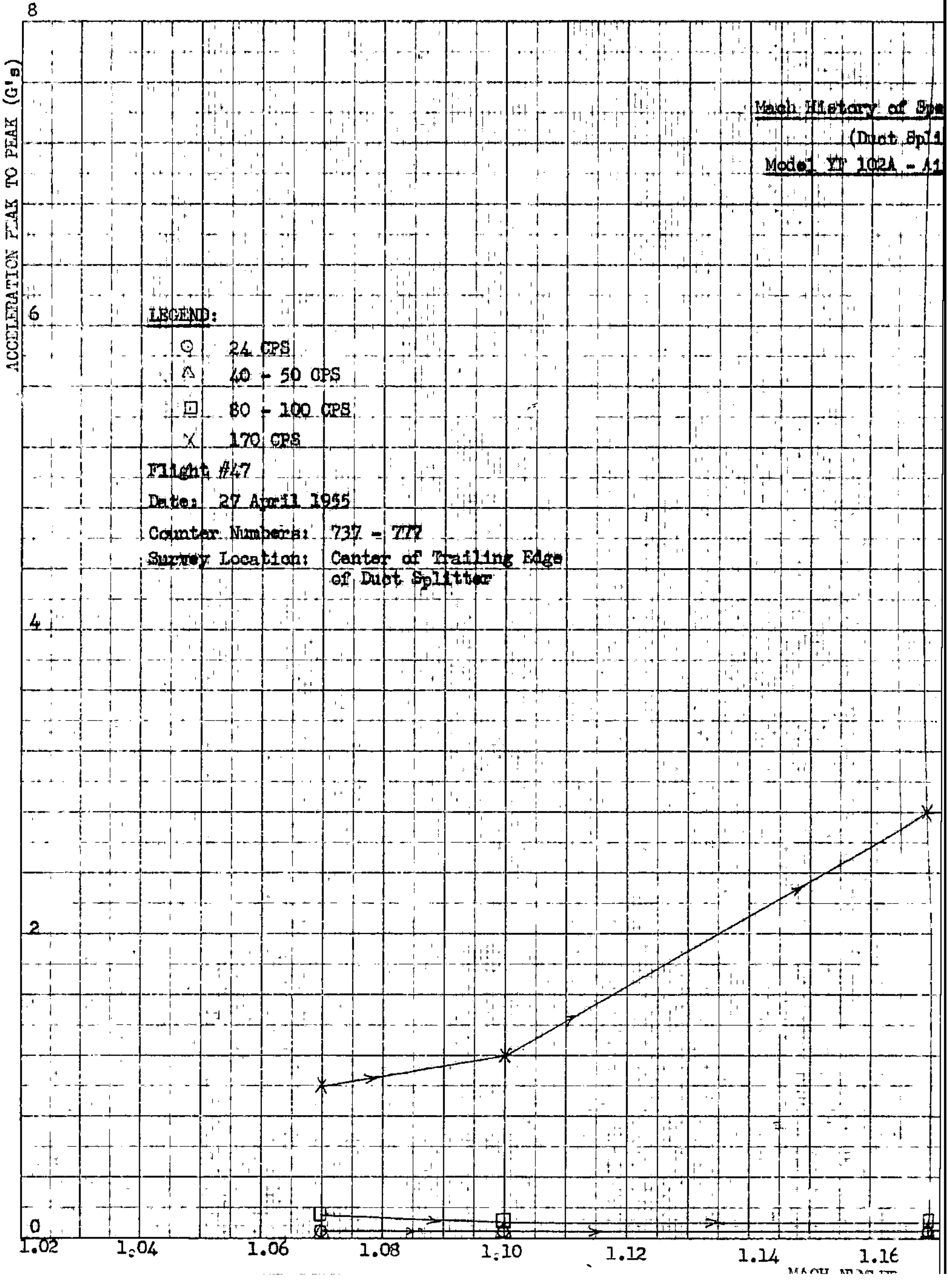
Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 777
Mach No: 1.18 (Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta. 425 (Approx.)

NOTE:
Engine was Throttled Back
Cabin Pressure Dumped

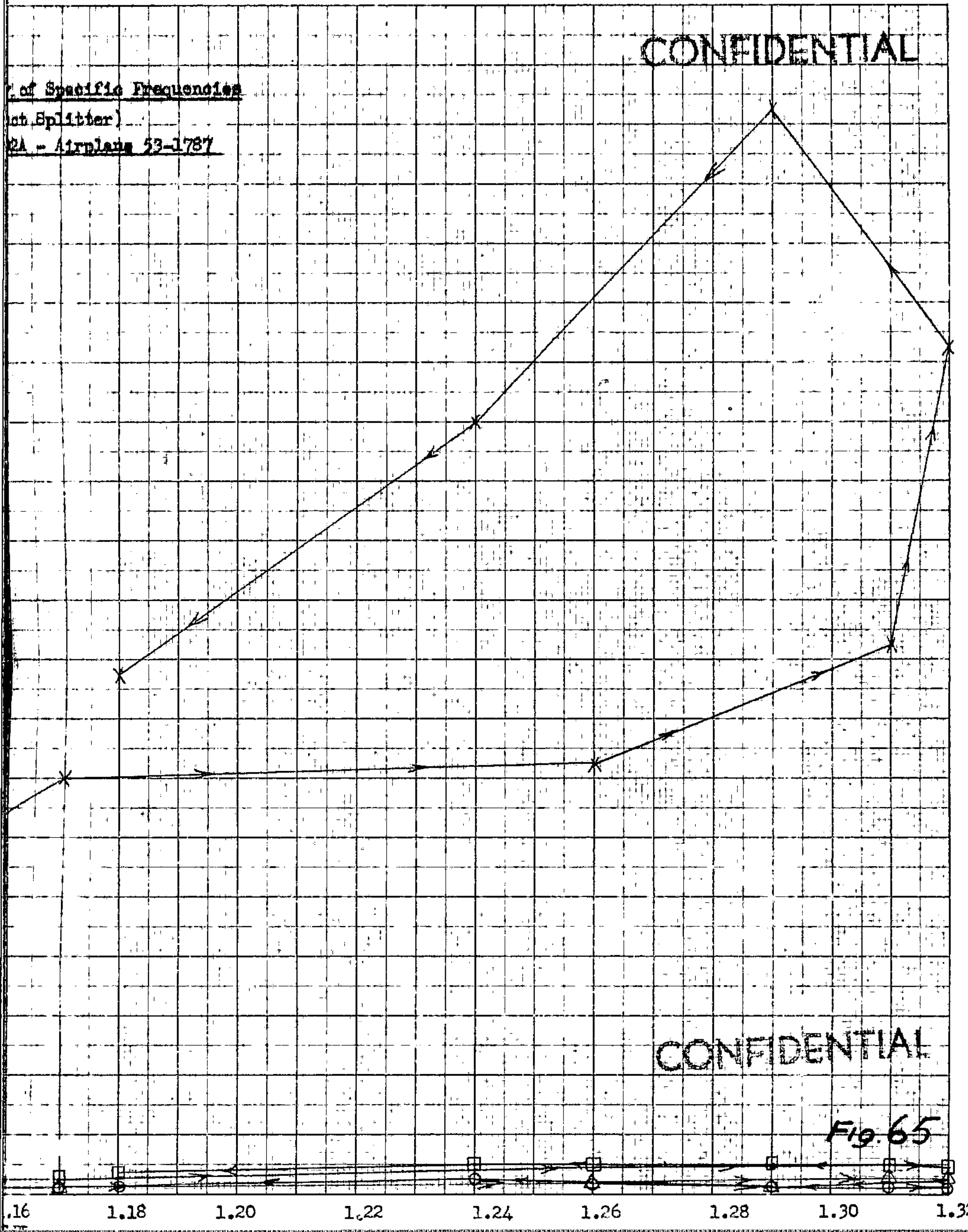
CONFIDENTIAL

Fig. 64



10 X 10 TO THE 1/2 INCH 359-11L

CONFIDENTIAL



of Specific Frequencies

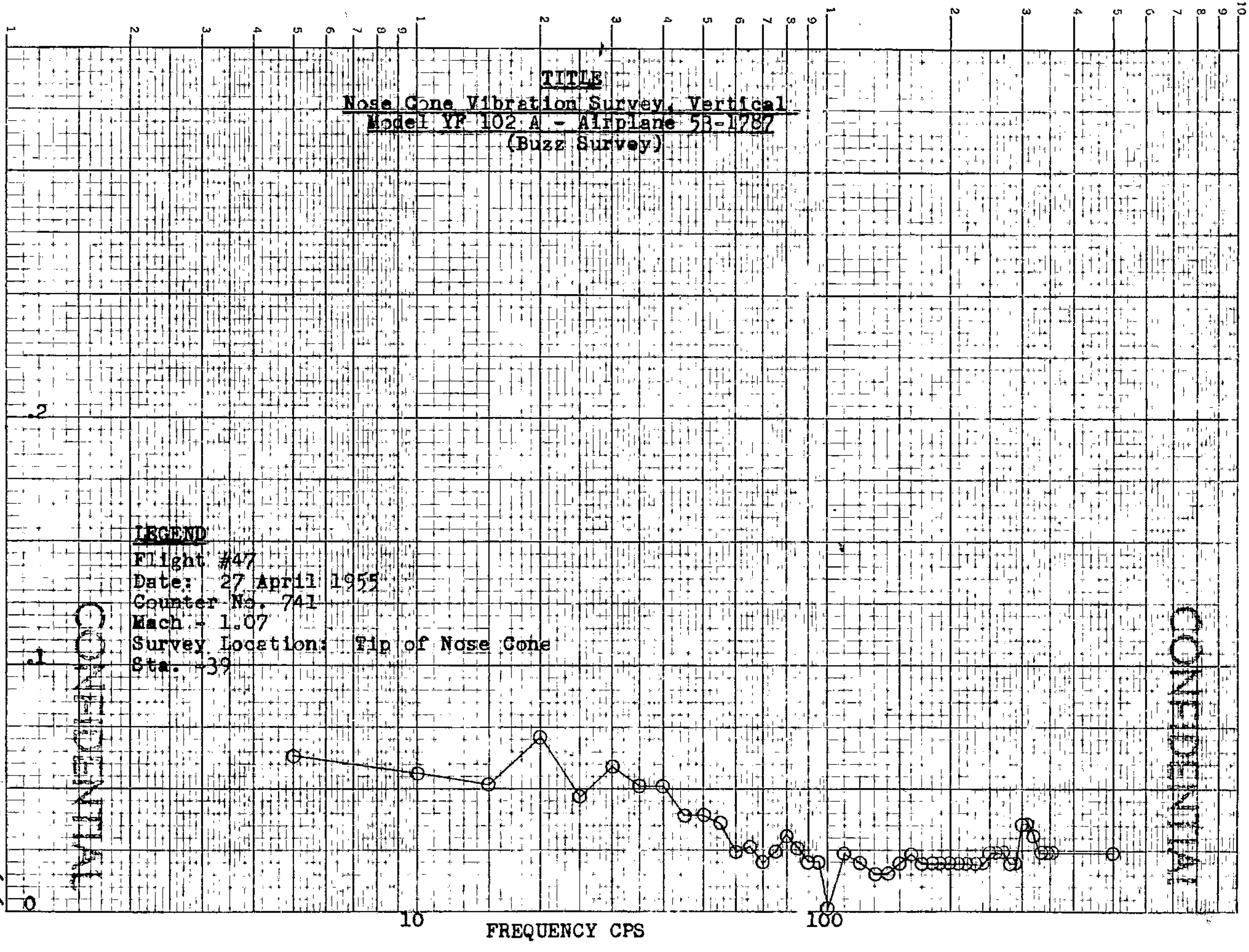
ct Splitter)

2A - Airplane 53-1787

CONFIDENTIAL

Fig. 65

CONFIDENTIAL



TITLE

Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND

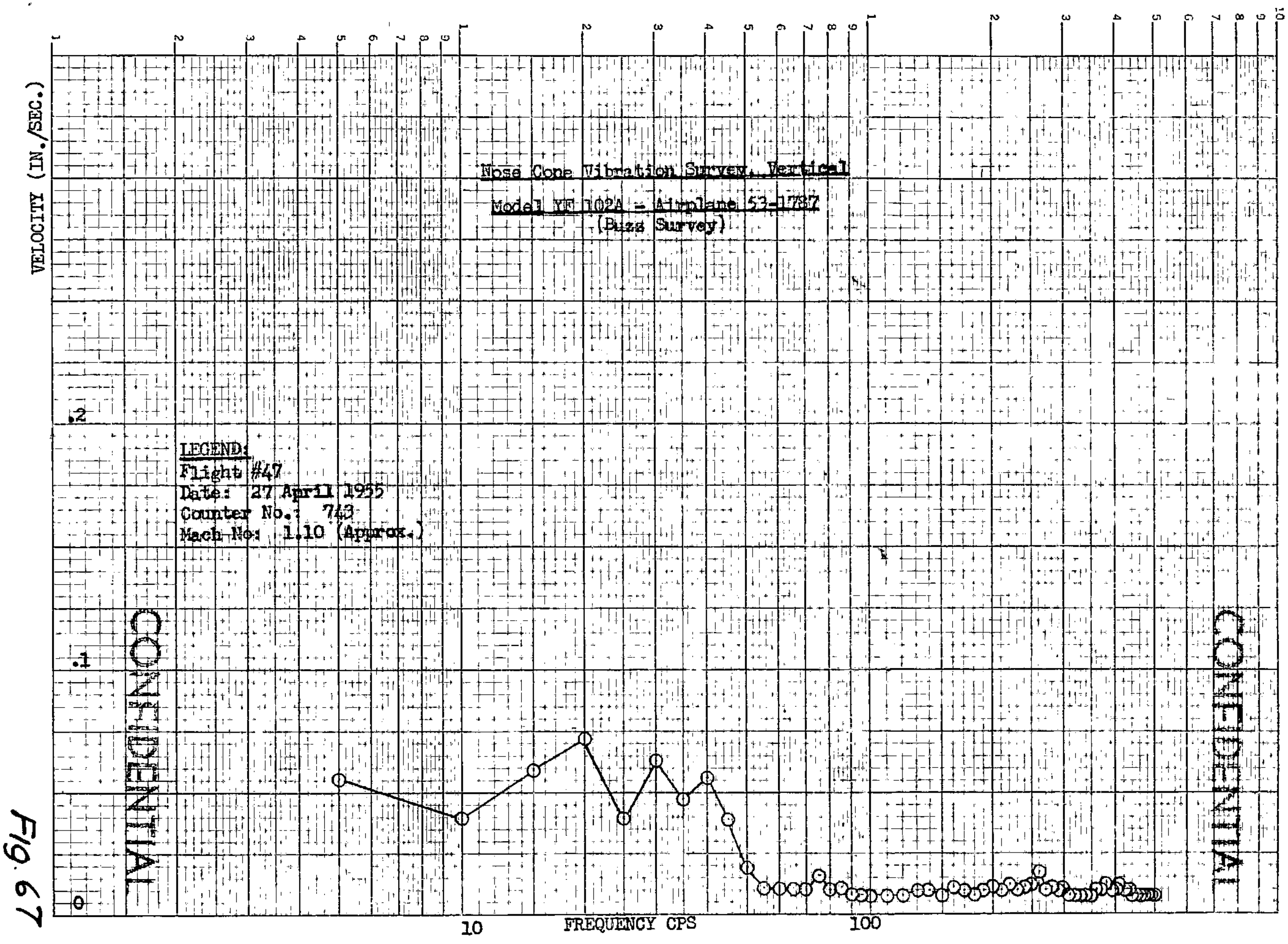
Flight #47
Date: 27 April 1955
Counter No. 741
Mach - 1.07
Survey Location: Tip of Nose Cone
Sta. -39

CONFIDENTIAL

Fig. 6.6

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL

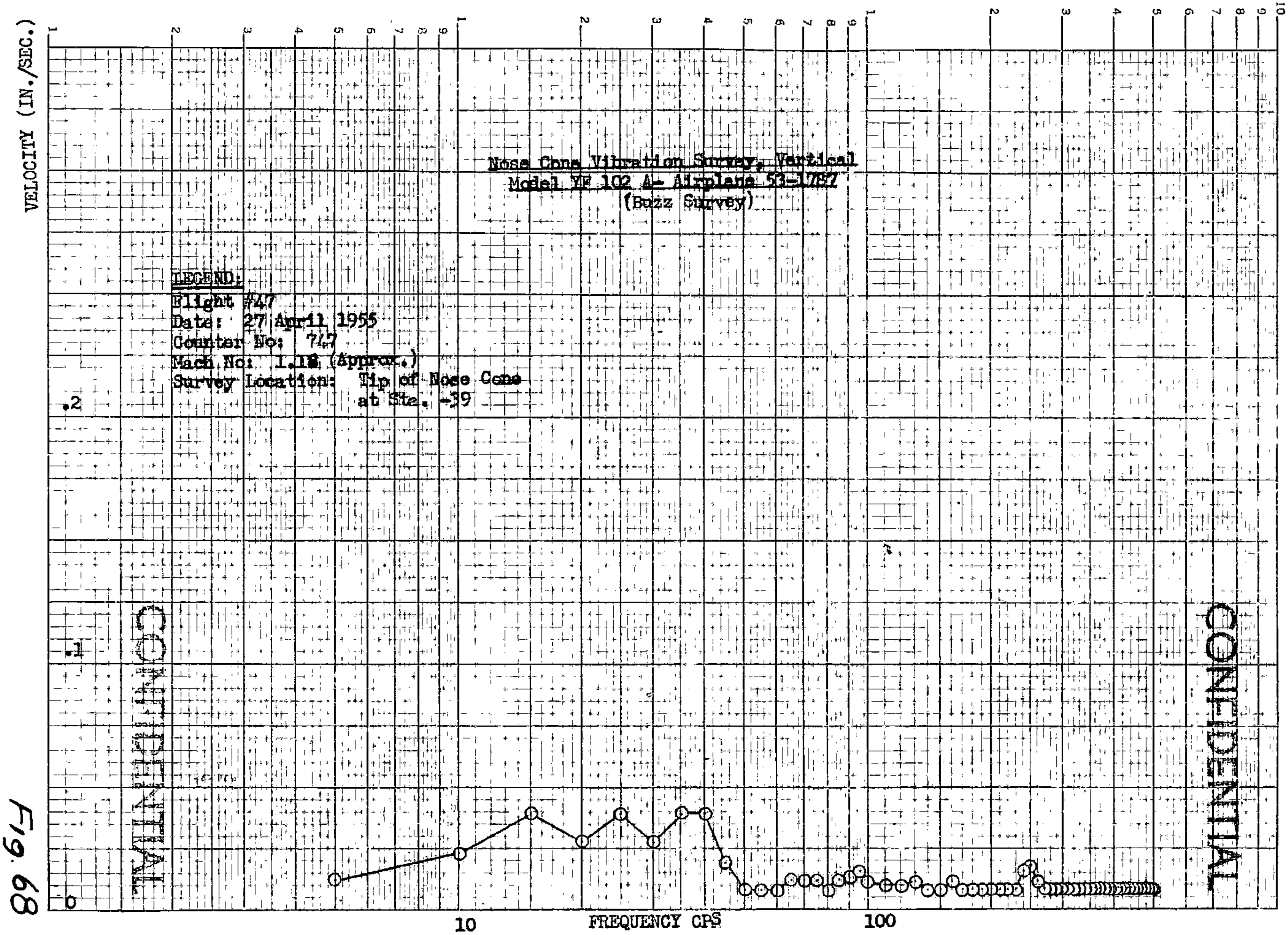


CONFIDENTIAL

Fig. 67

K-E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical
Model YF 102 A- Airplane 53-1787
(Buzz Survey)

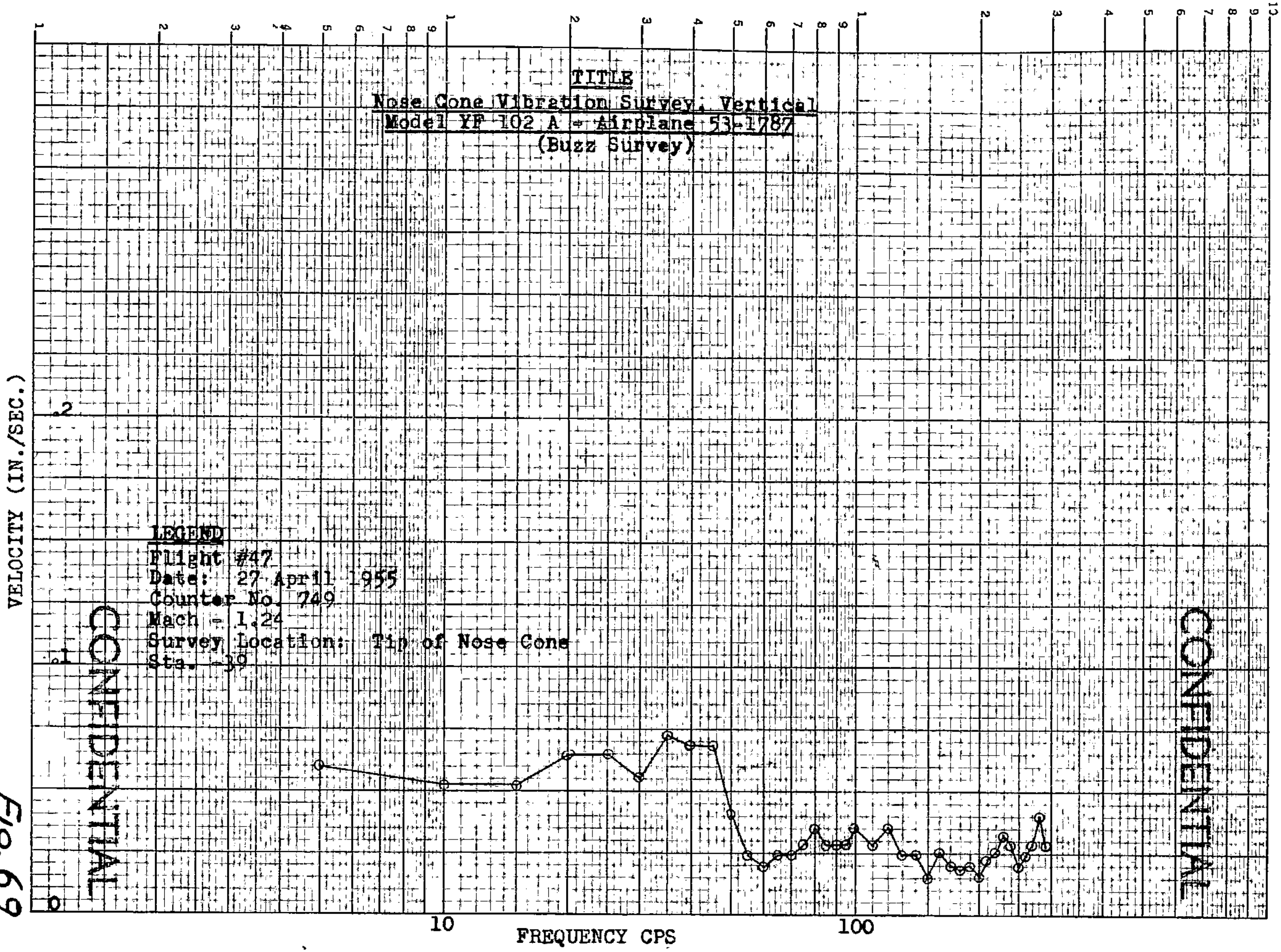
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 747
Mach No: 1.18 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

CONFIDENTIAL

Fig. 68

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

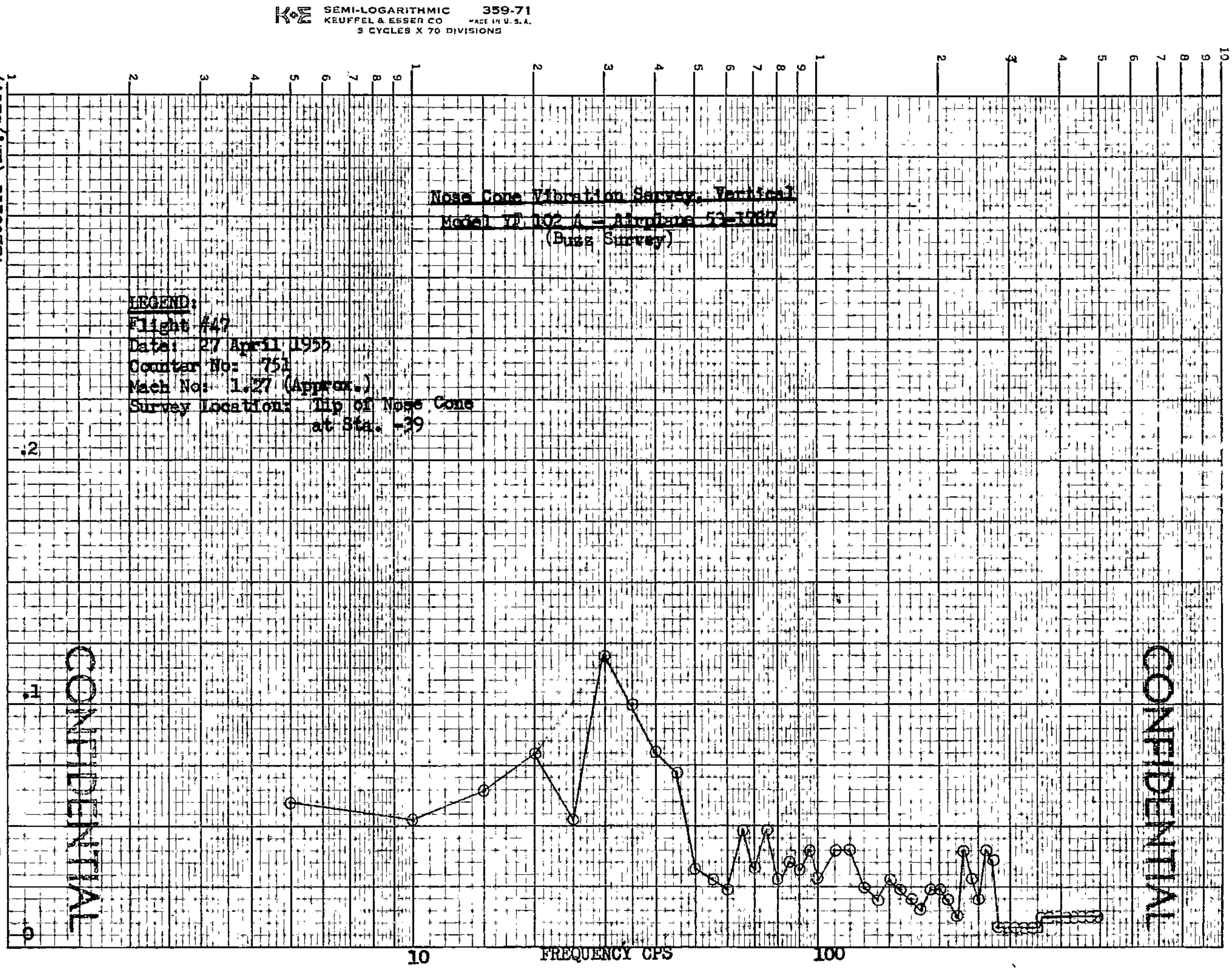
Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND

Flight #47
Date: 27 April 1955
Counter No. 749
Mach - 1.24
Survey Location: Tip of Nose Cone
Sta. -39

CONFIDENTIAL

Fig. 69

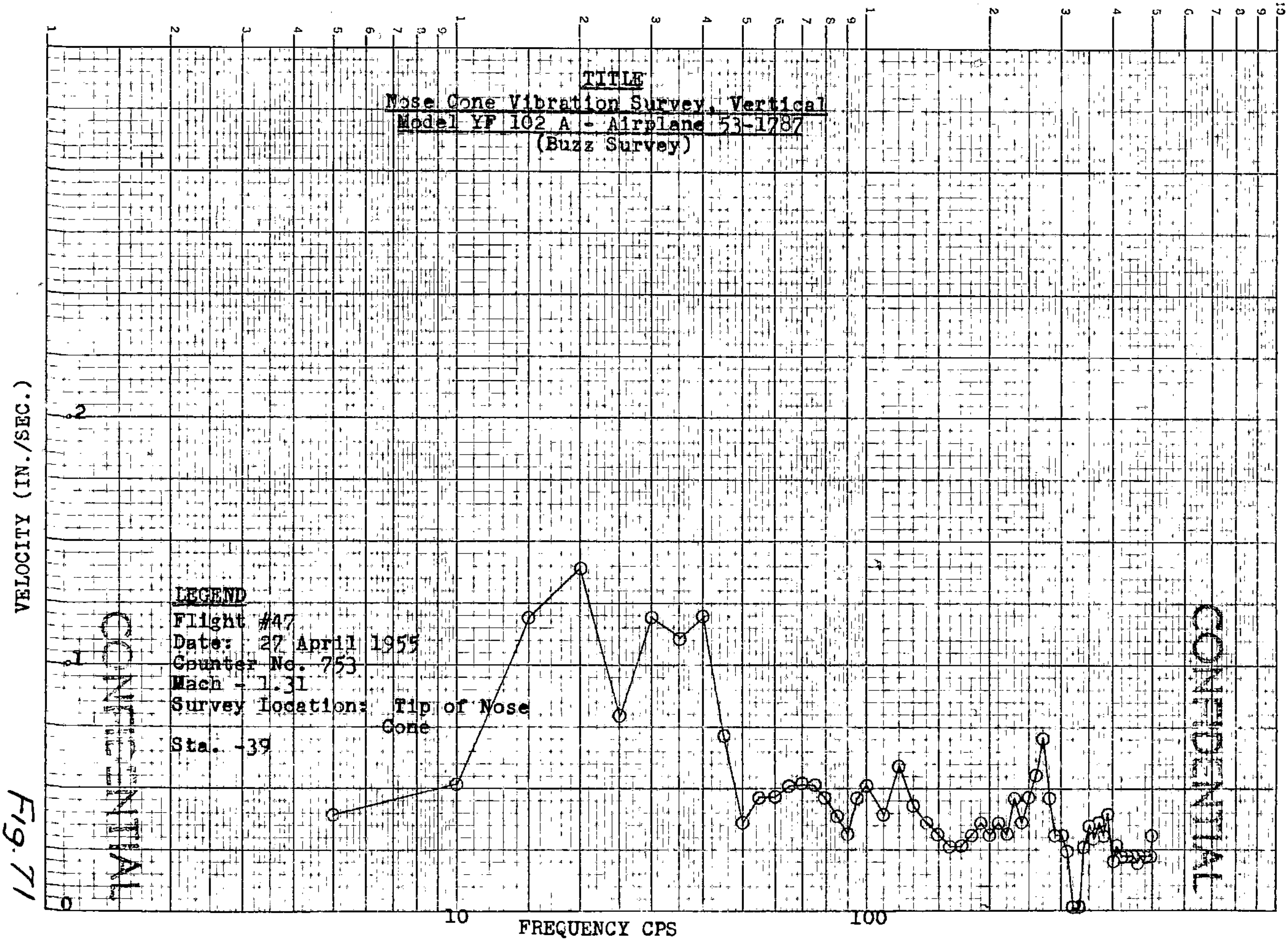


Nose Cone Vibration Survey, Vertical
Model YF-102 A – Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 751
Mach No: 1.27 (Approx.)
Survey Location: Tip of Nose Cone at Sta. -39

Fig. 70

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

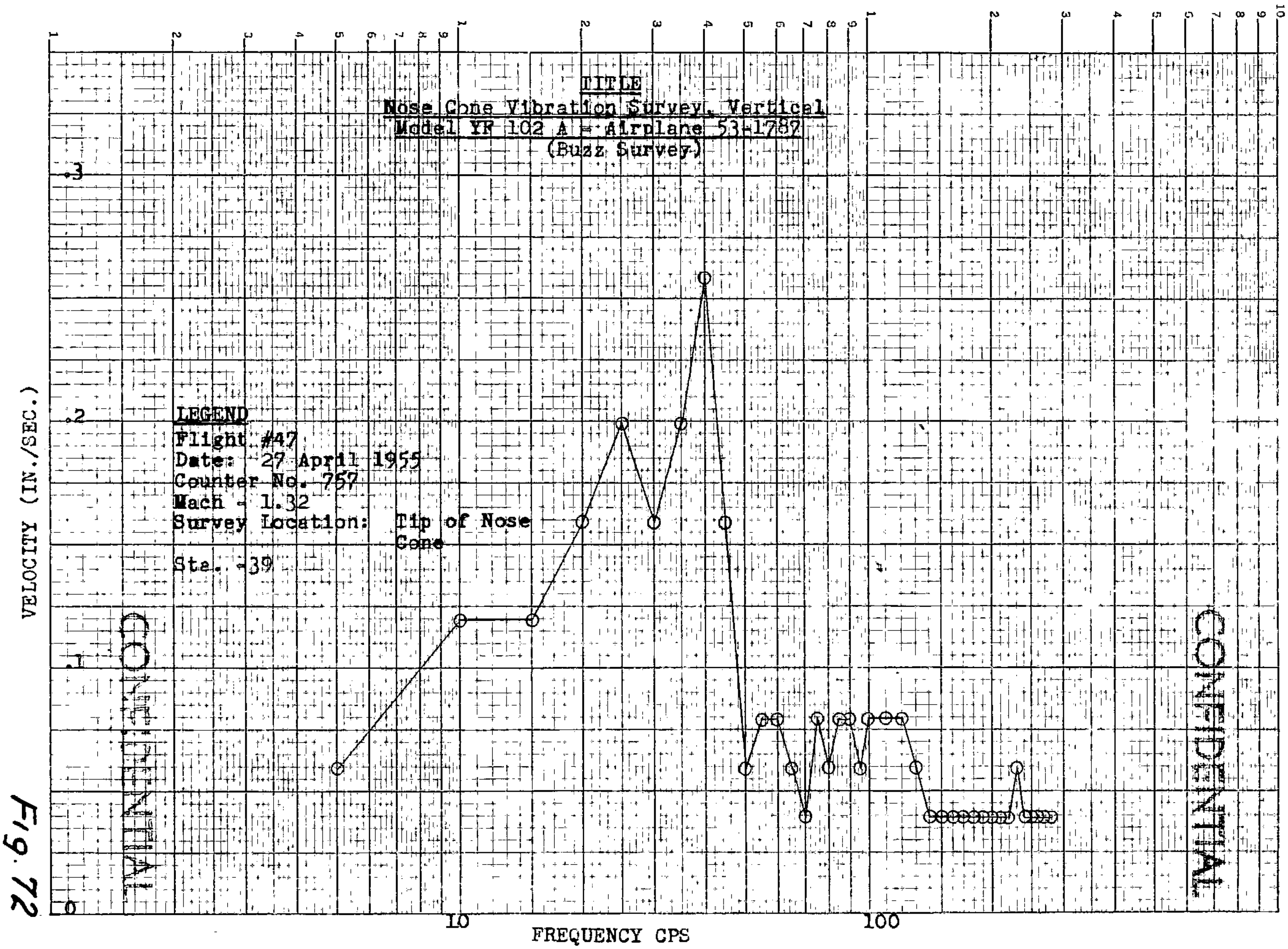


TITLE
Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 753
Mach - 1.31
Survey Location: Tip of Nose Cone
Sta. -39

Fig. 71

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



TITLE

Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

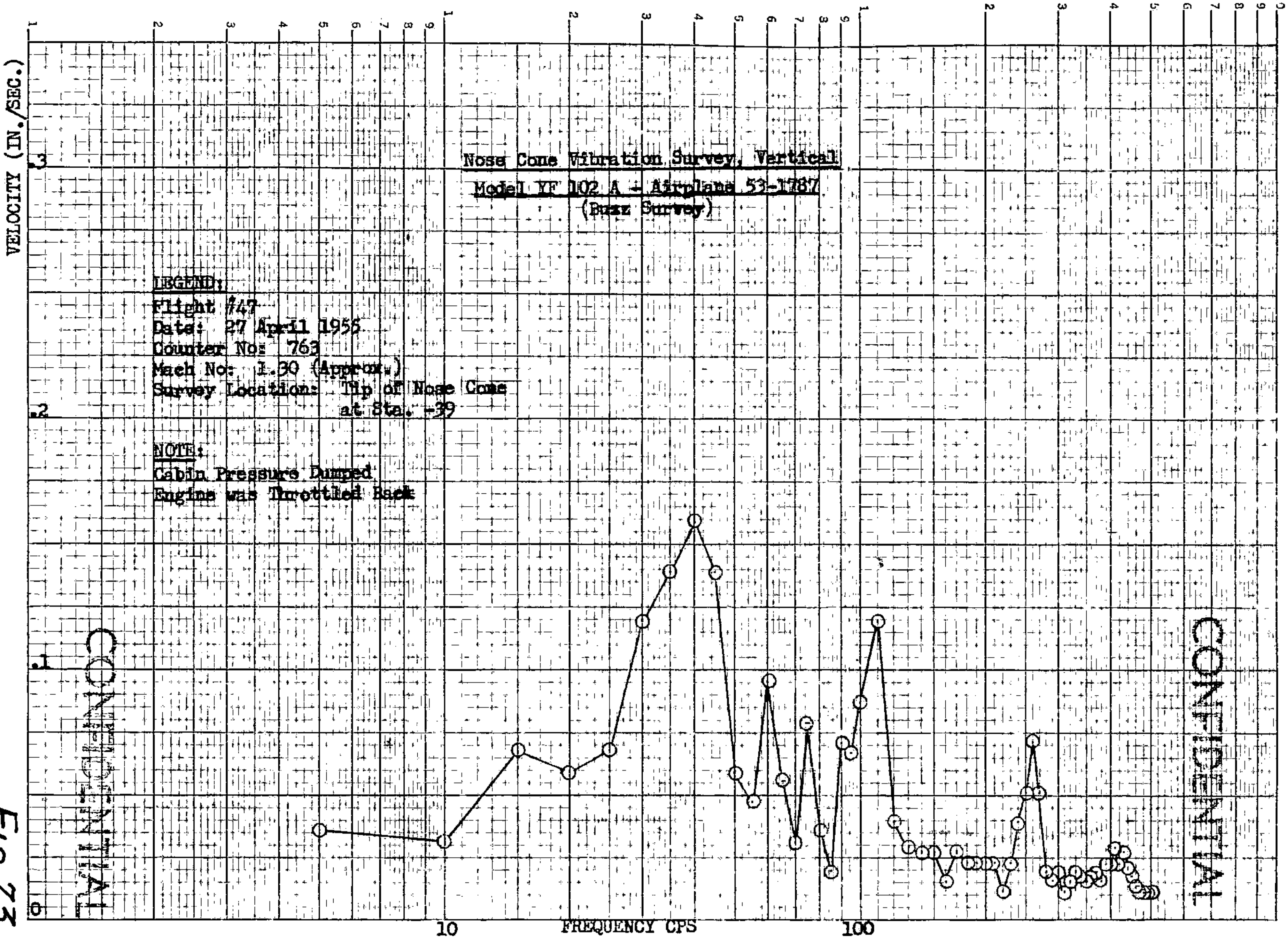
LEGEND

Flight #47
Date: 27 April 1955
Counter No. 757
Mach - 1.32
Survey Location: Tip of Nose Cone
Sta. -39

Fig. 72

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical

Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 763
Mach No: 1.30 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

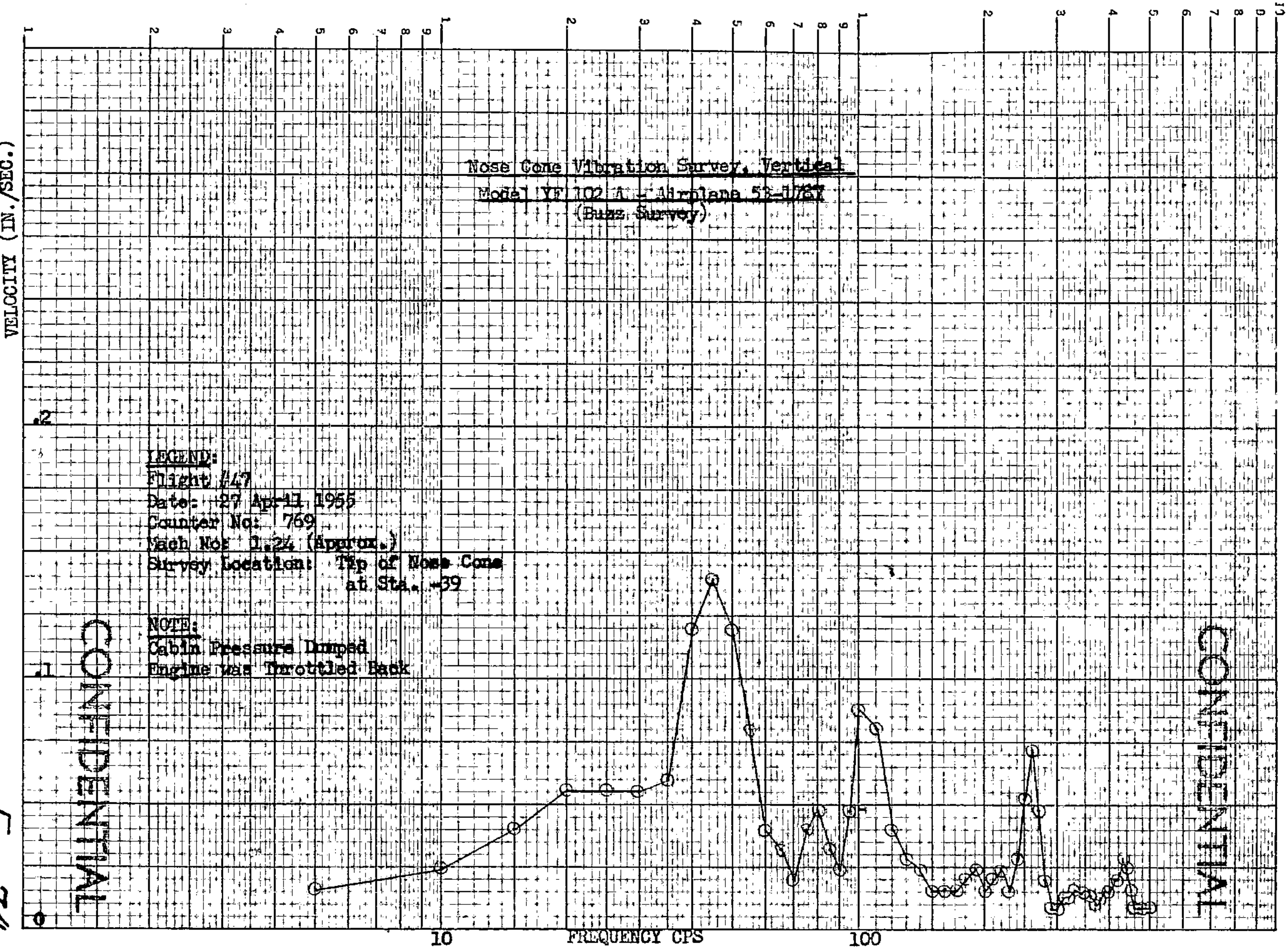
NOTE:
Cabin Pressure Dumped
Engine was Throttled Back

CONFIDENTIAL

Fig. 73

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL

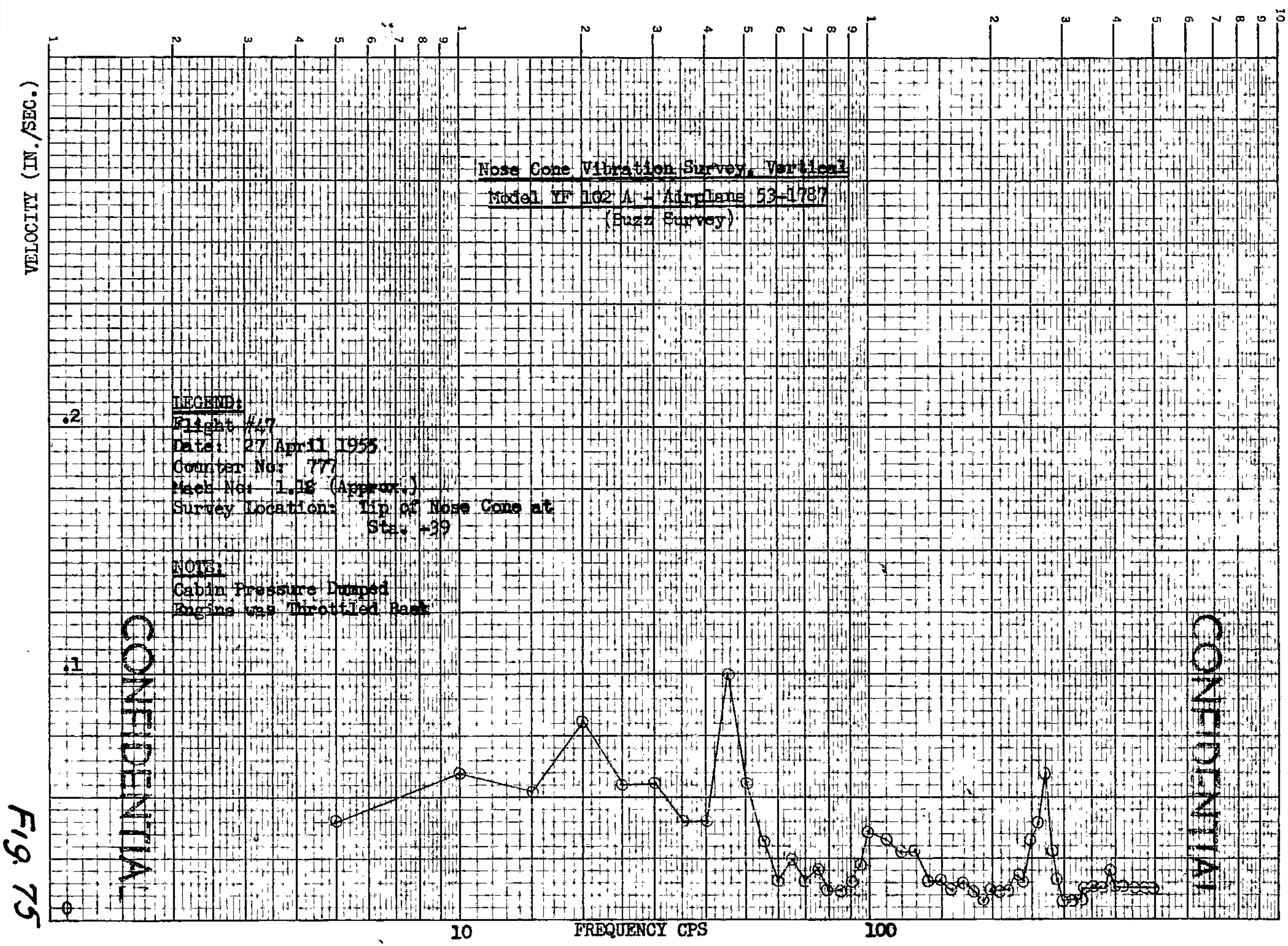


CONFIDENTIAL

Fig. 74

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical

Model YF 102 A - Airplane 53-1787

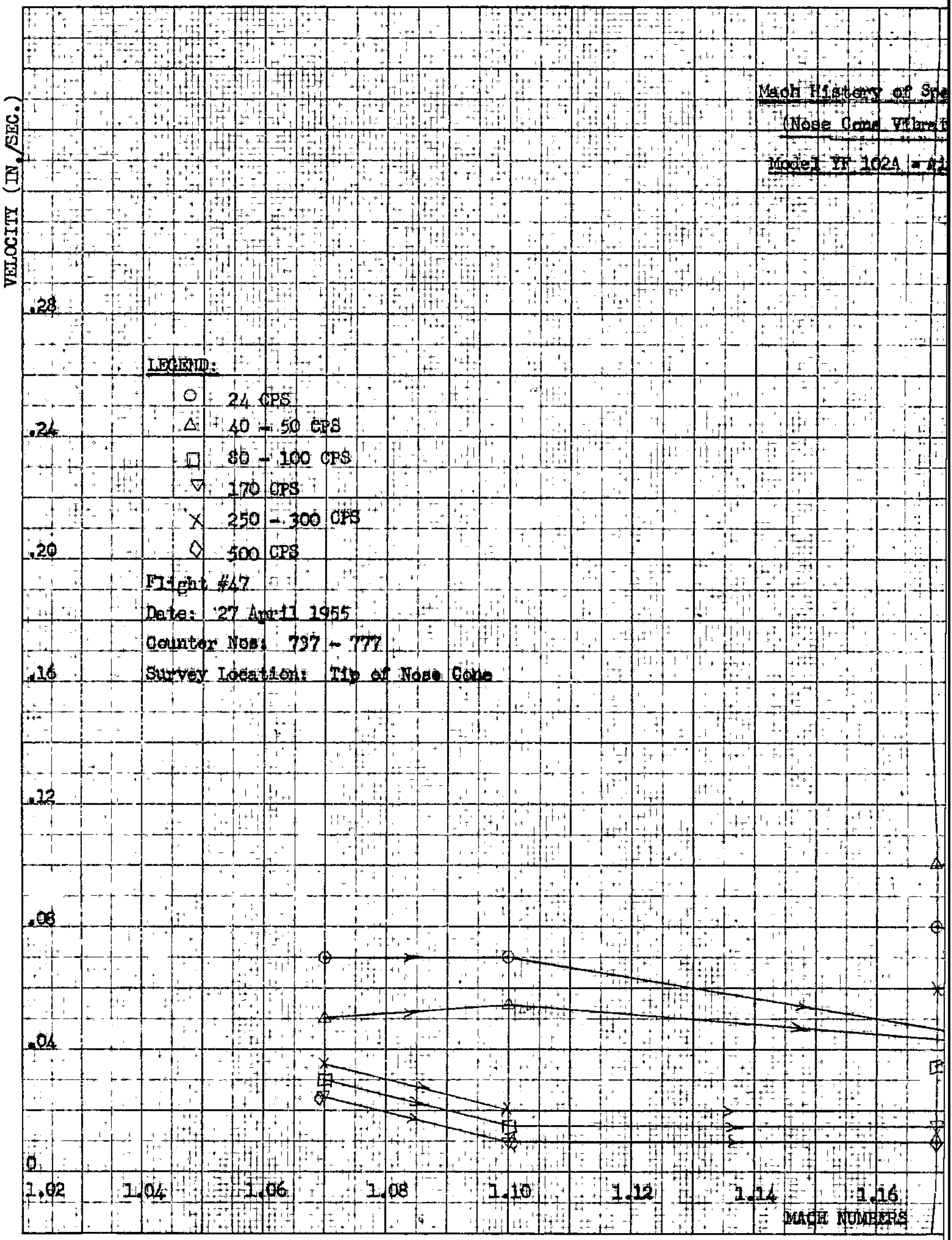
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 777
Mach No: 1.18 (Approx.)
Survey Location: Tip of Nose Cone at
Sta. +39

NOTE:
Cabin Pressure Dumped
Engine was Throttled Back

CONFIDENTIAL

Fig. 75



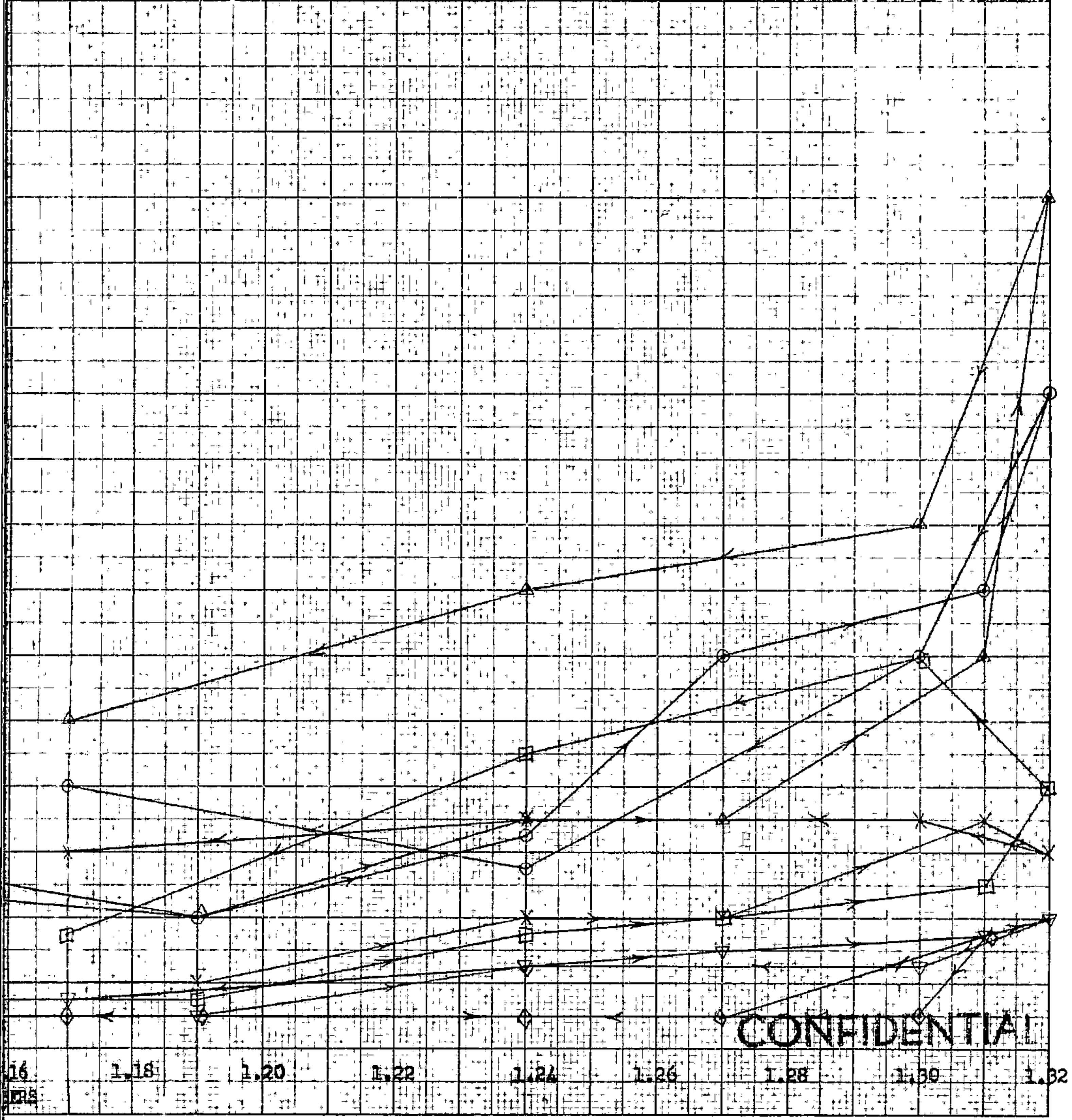
10 X 10 TO THE ½ INCH 359-11L

CONFIDENTIAL

y of Specific Frequencies

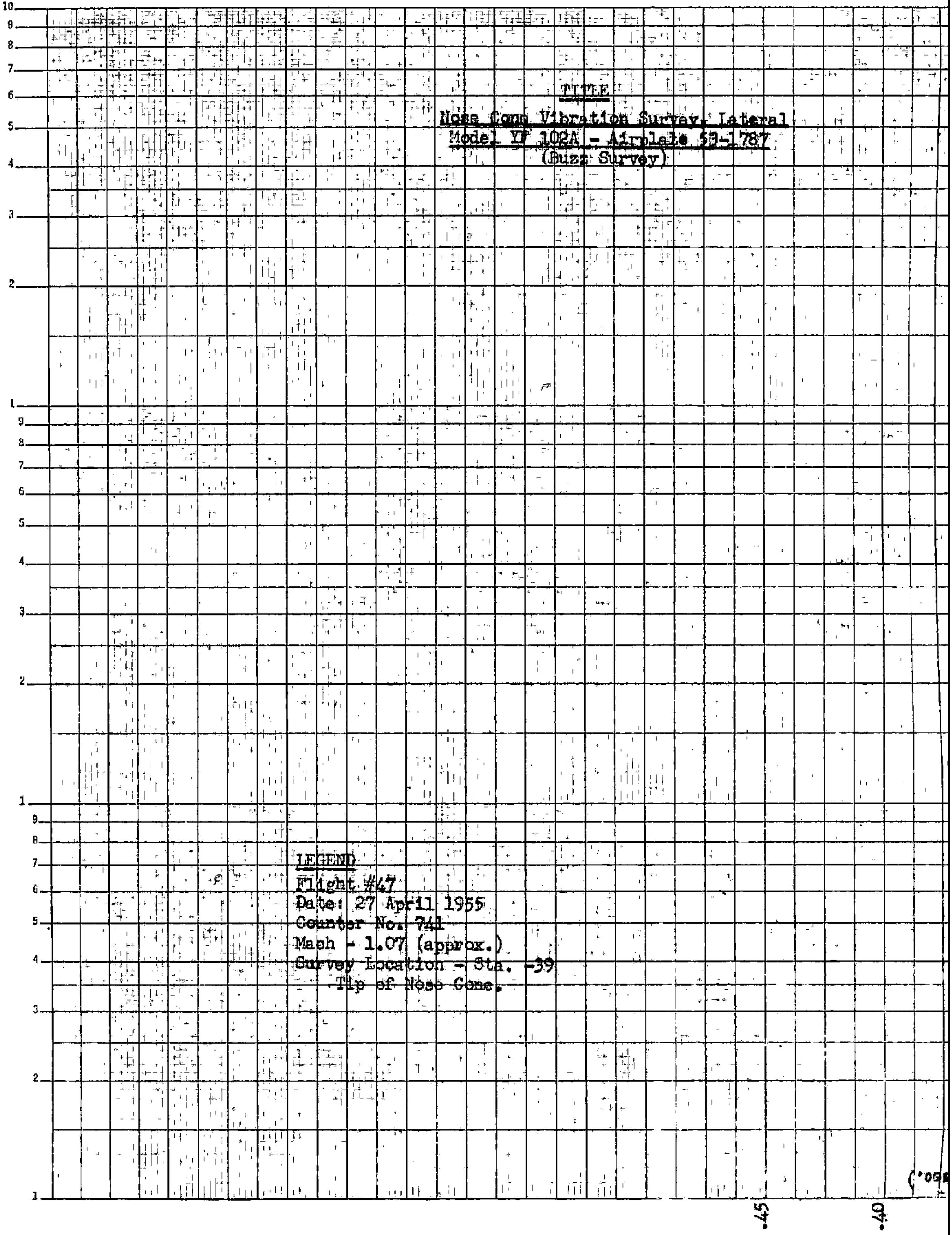
Vibration, Vertical)

024 = Airplane 53-1787



CONFIDENTIAL

Fig 76



TITLE

Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 741
Mach - 1.07 (approx.)
Survey Location - Sta. -39
Tip of Nose Cone.

359-73L KEUFFEL & ESSER CO.
Semi-Logarithmic, 3 Cycles × 10 to the ½ inch.
MADE IN U.S.A.

CONFIDENTIAL

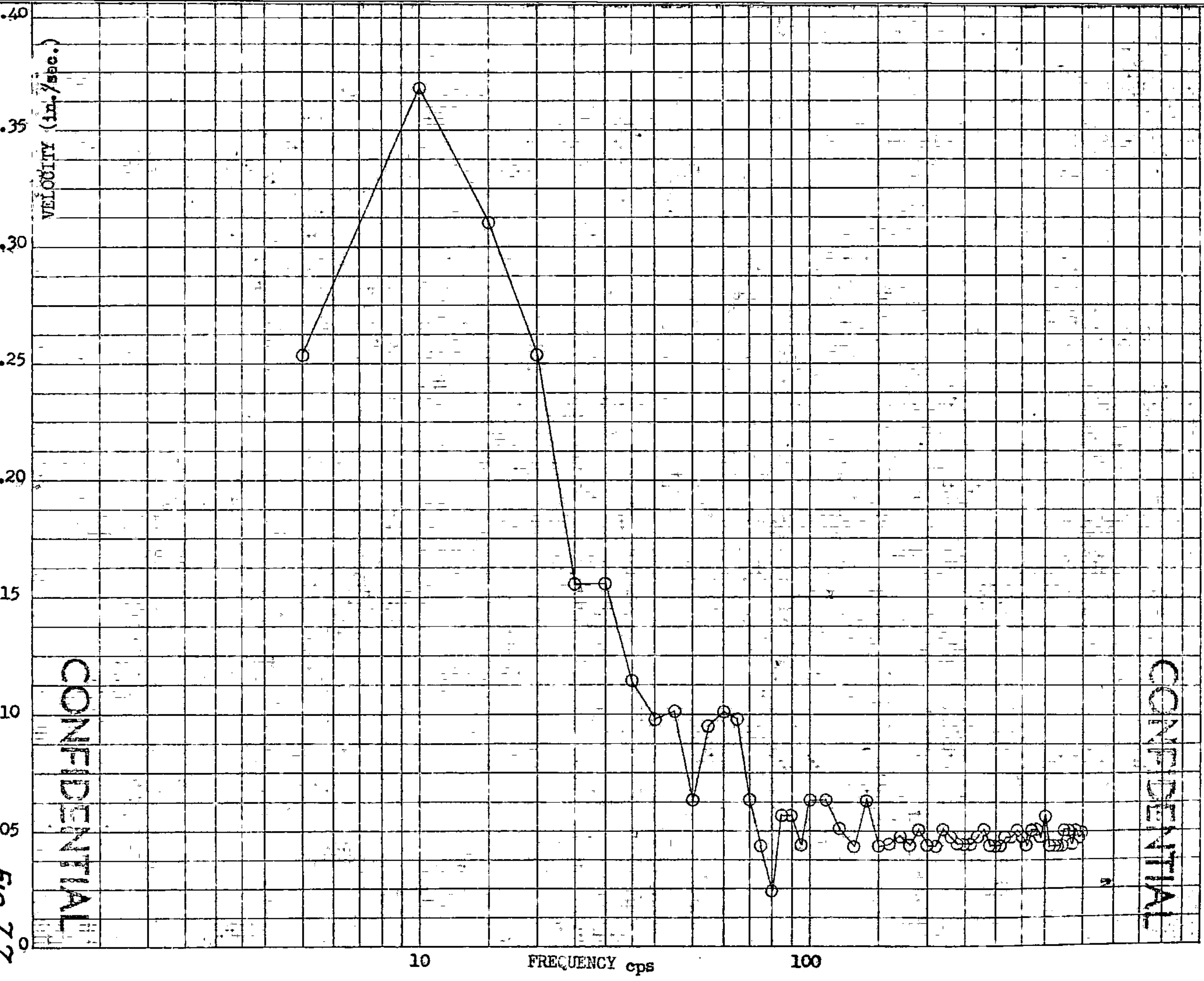
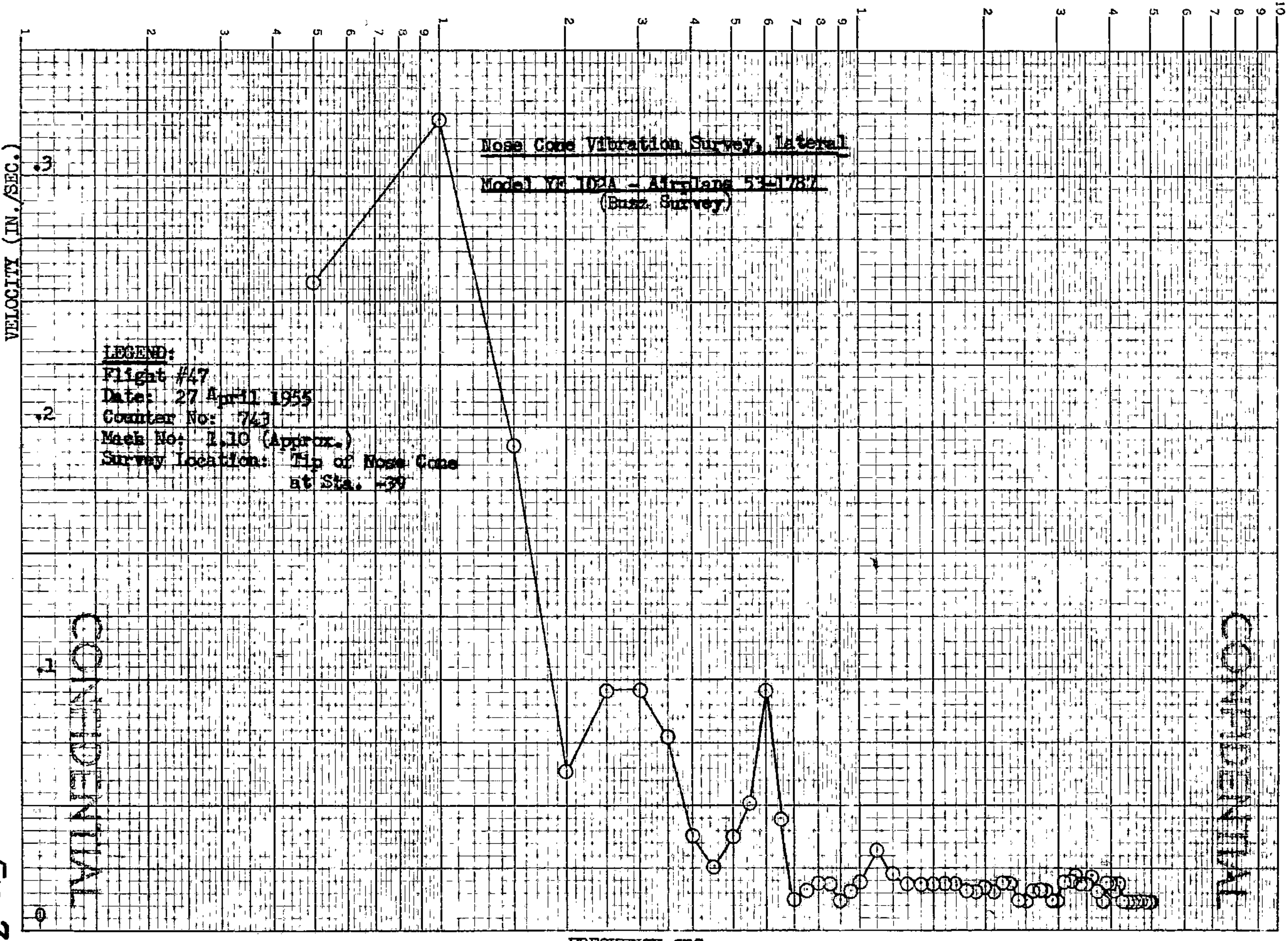


Fig. 77

CONFIDENTIAL

CONFIDENTIAL


Nose Cone Vibration Survey, Lateral

Model YF-102A - Airplane 53-1787
(Buzz Survey)

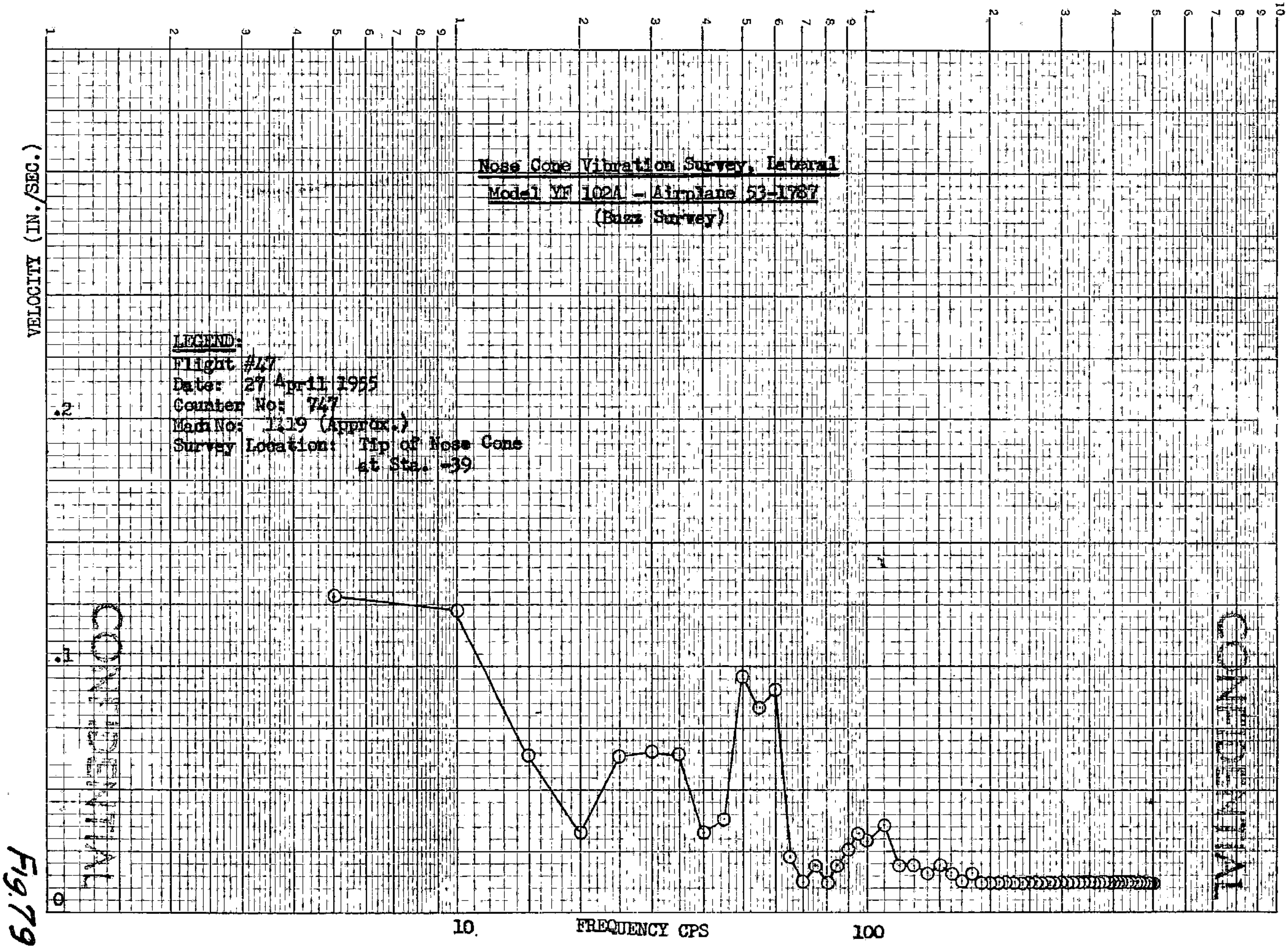
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 743
Mach No: 1.10 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

CONFIDENTIAL

Fig. 78

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Lateral

Model YF 102A - Airplane 53-1787

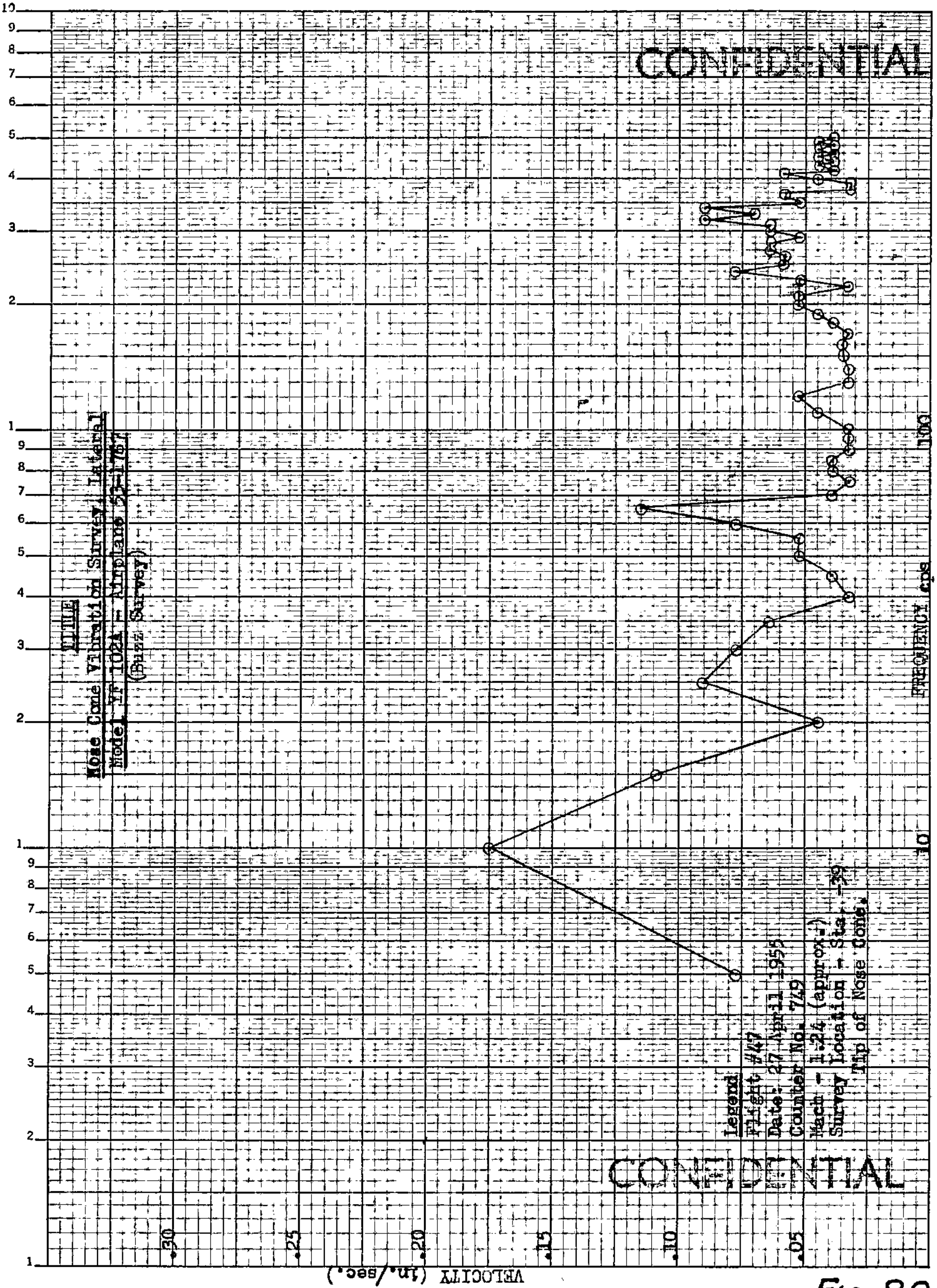
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 747
Mach No: 1.19 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

CONFIDENTIAL

Fig. 79

CONFIDENTIAL

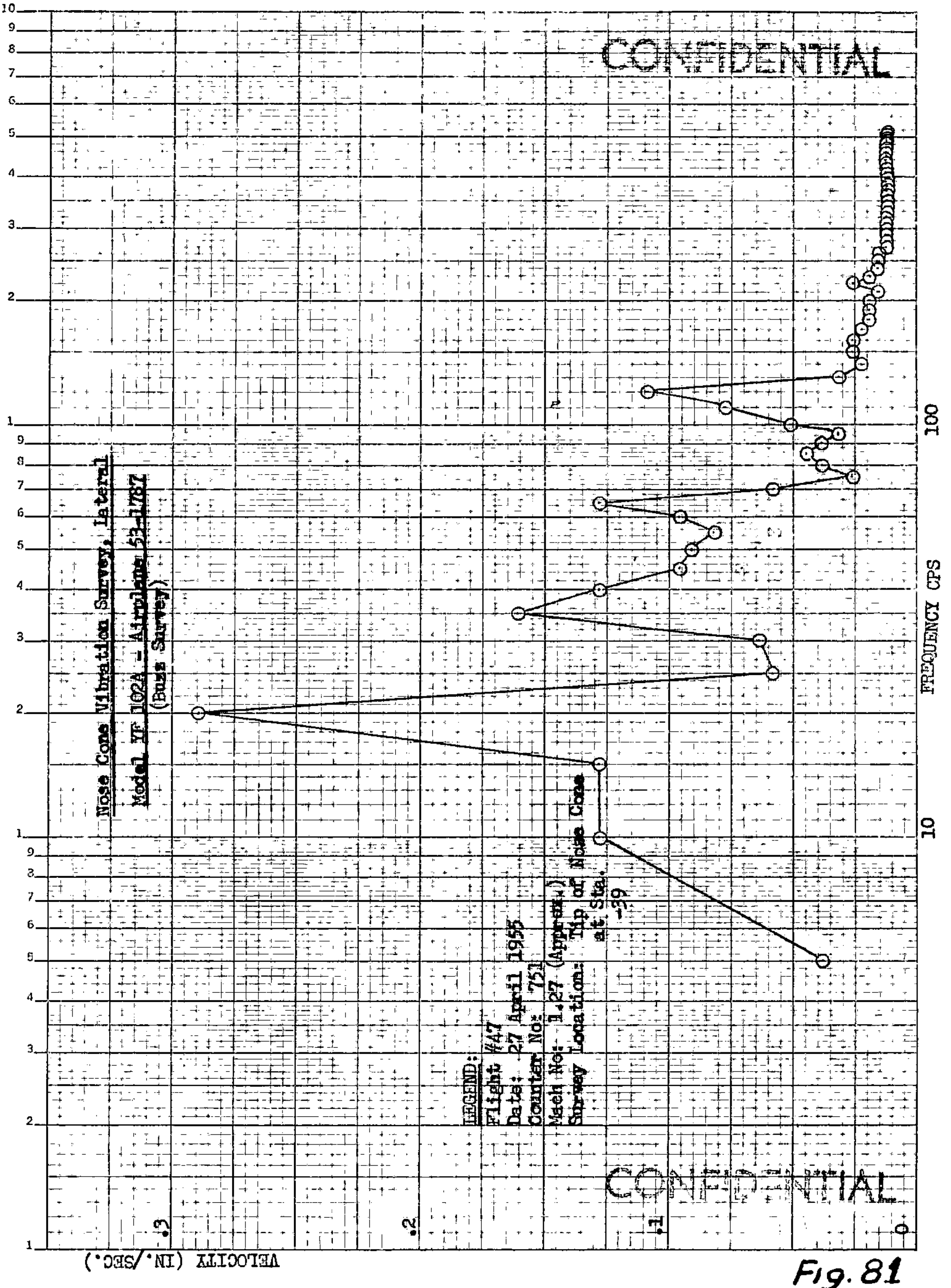


CONFIDENTIAL

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

Fig. 80

CONFIDENTIAL



Nose Cone Vibration Survey, Lateral

Model YF 102A - Airplane 53-1787

(Buzz Survey)

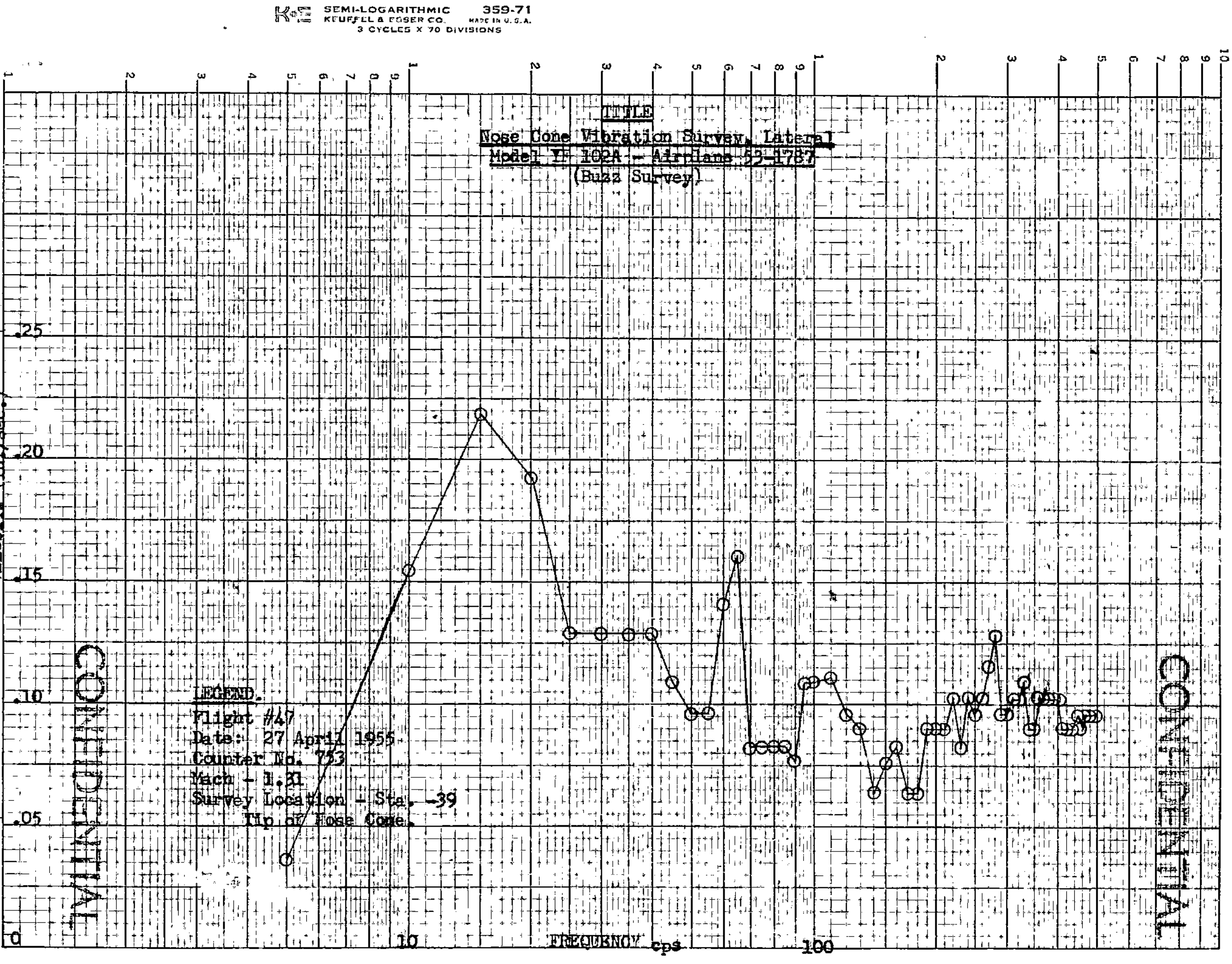
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 751
Mach No: 1.27 (Approx.)
Survey Location: Tip of Nose Cone at Sta. -39

CONFIDENTIAL

Fig. 81

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND.
Flight #47
Date: 27 April 1955
Counter No. 753
Mach - 1.31
Survey Location - Sta. -39
Tip of Nose Cone.

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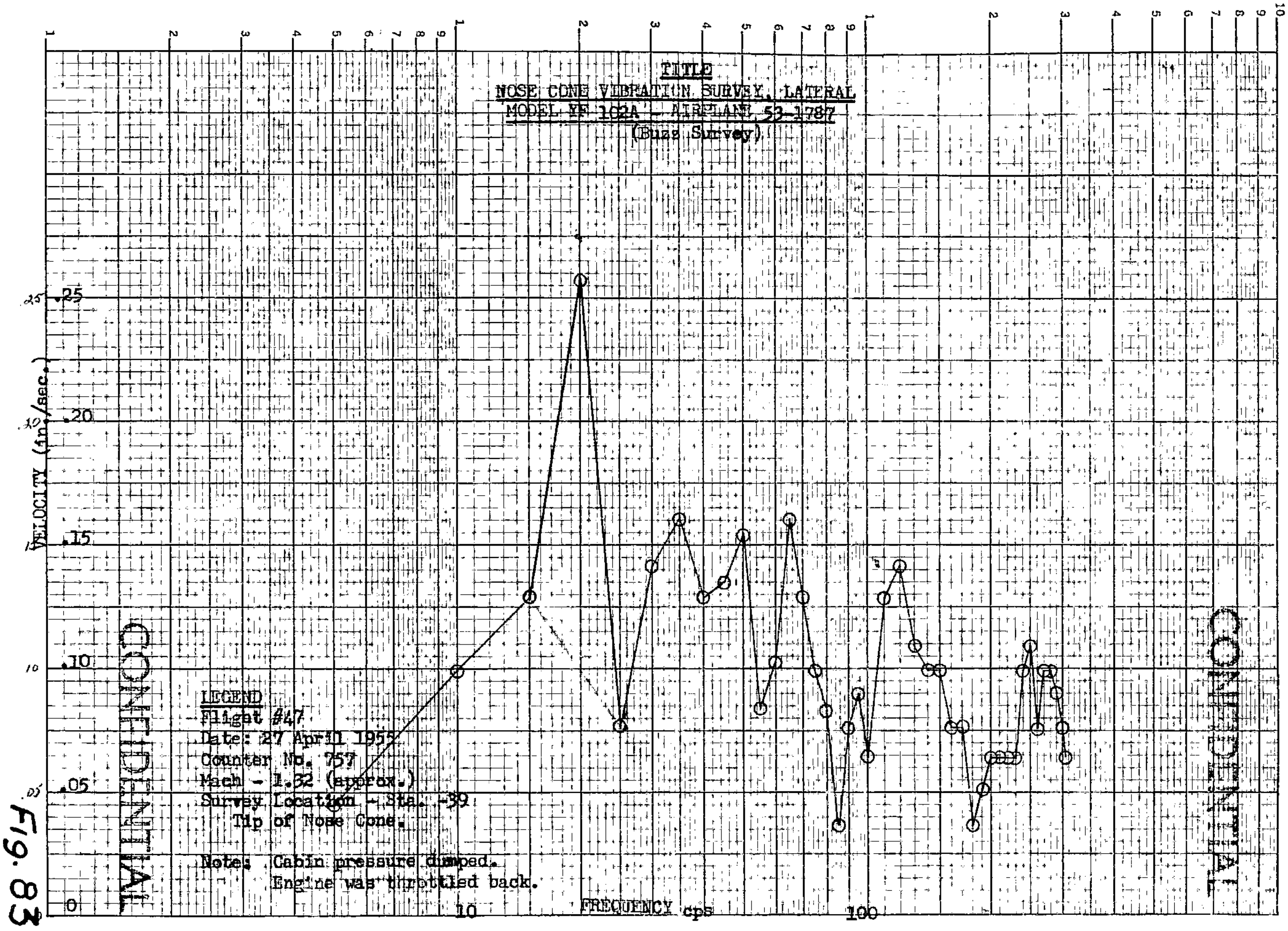
Fig. 82

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



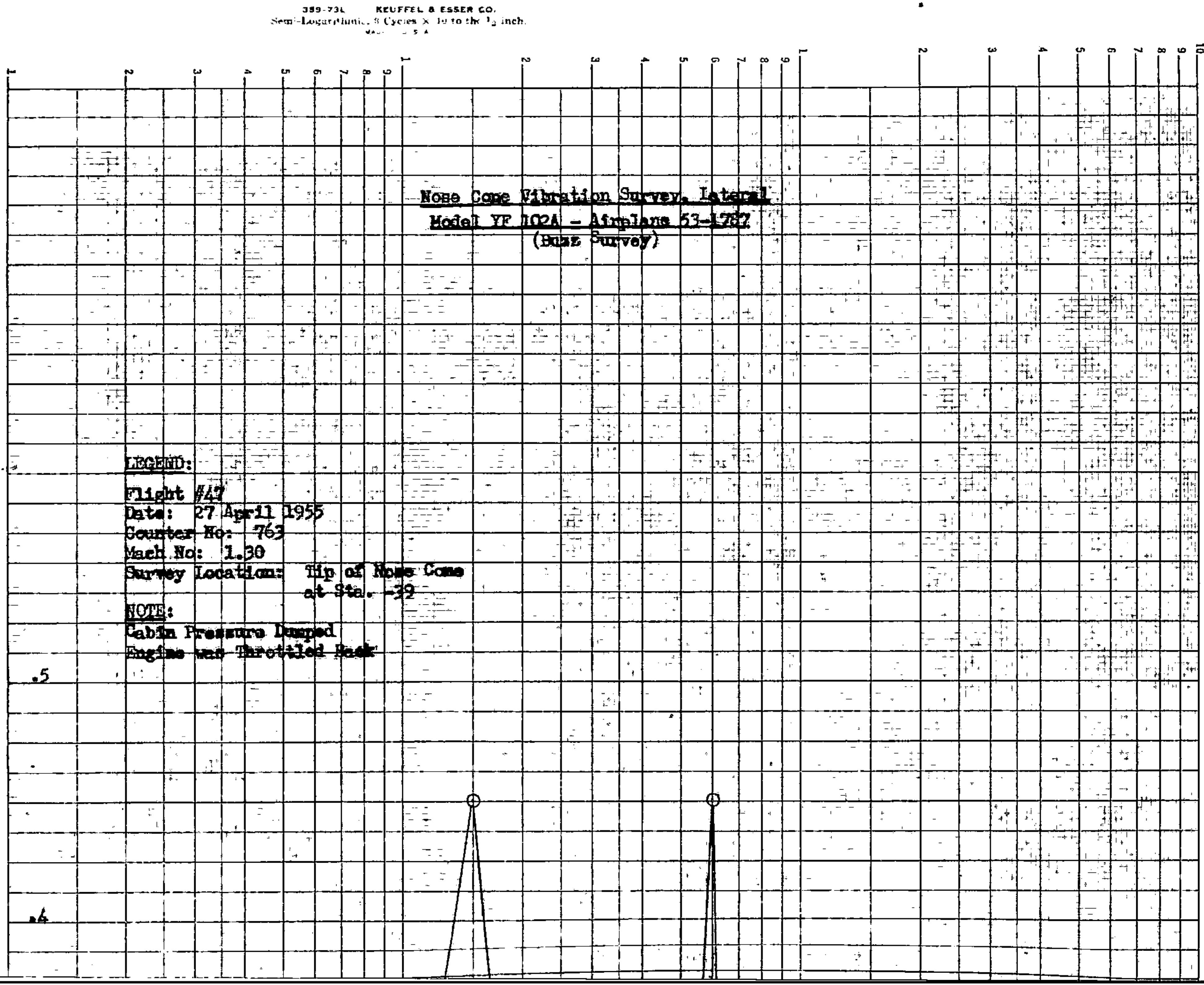
TITLE
NOSE CONE VIBRATION SURVEY, LATERAL
MODEL YF 102A - AIRPLANE 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 757
Mach - 1.32 (approx.)
Survey Location - Sta. -39
Tip of Nose Cone.

Note: Cabin pressure dumped.
Engine was throttled back.

CONFIDENTIAL

Fig. 83



Nose Cone Vibration Survey, Lateral

Model YF 102A - Airplane 53-1787

(Buzz Survey)

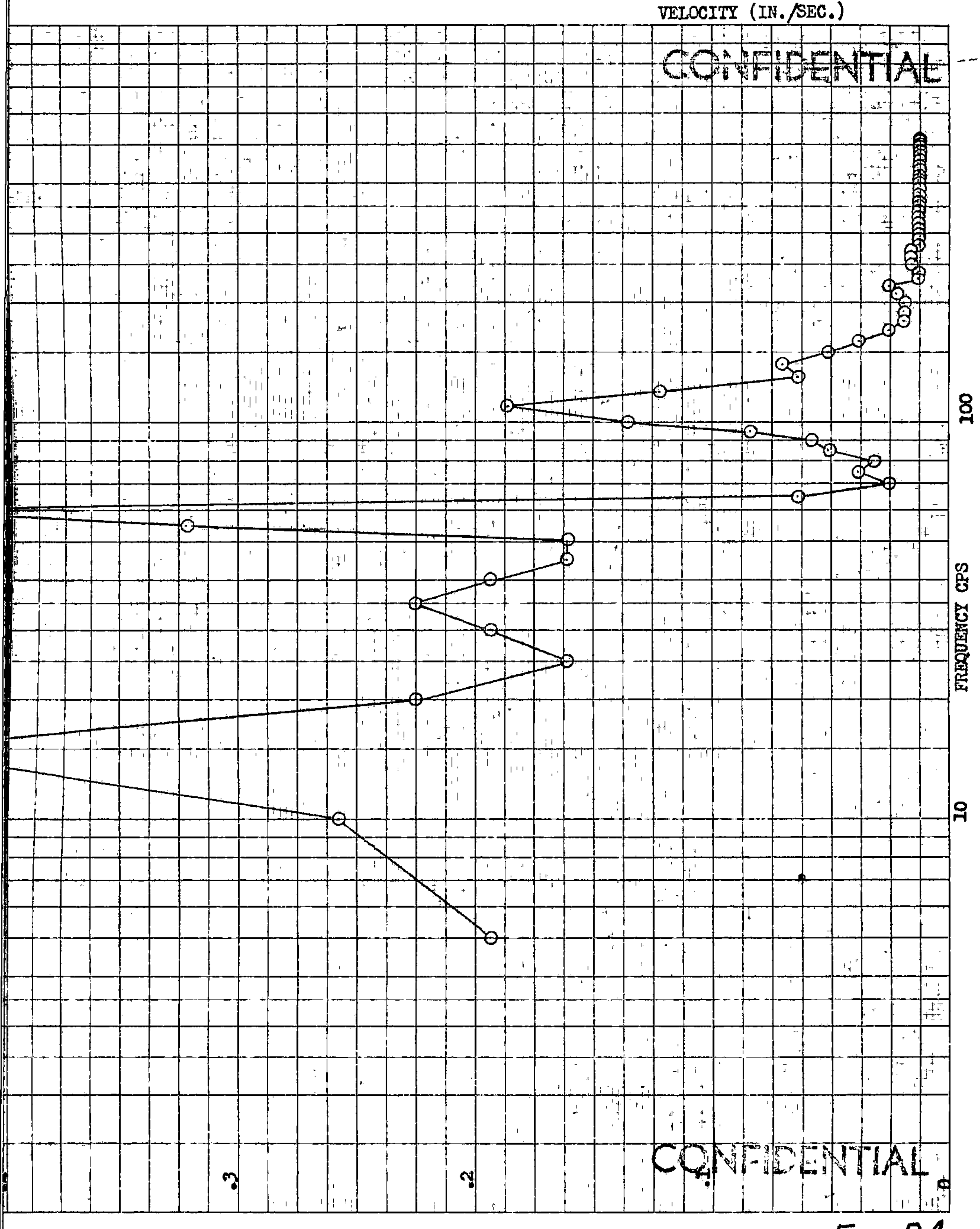
LEGEND:

Flight #47
Date: 27 April 1955
Counter No: 763
Mach No: 1.30
Survey Location: Tip of Nose Cone at Sta. -29

NOTE:

Cabin Pressure Dumped
Engine was Throttled Back

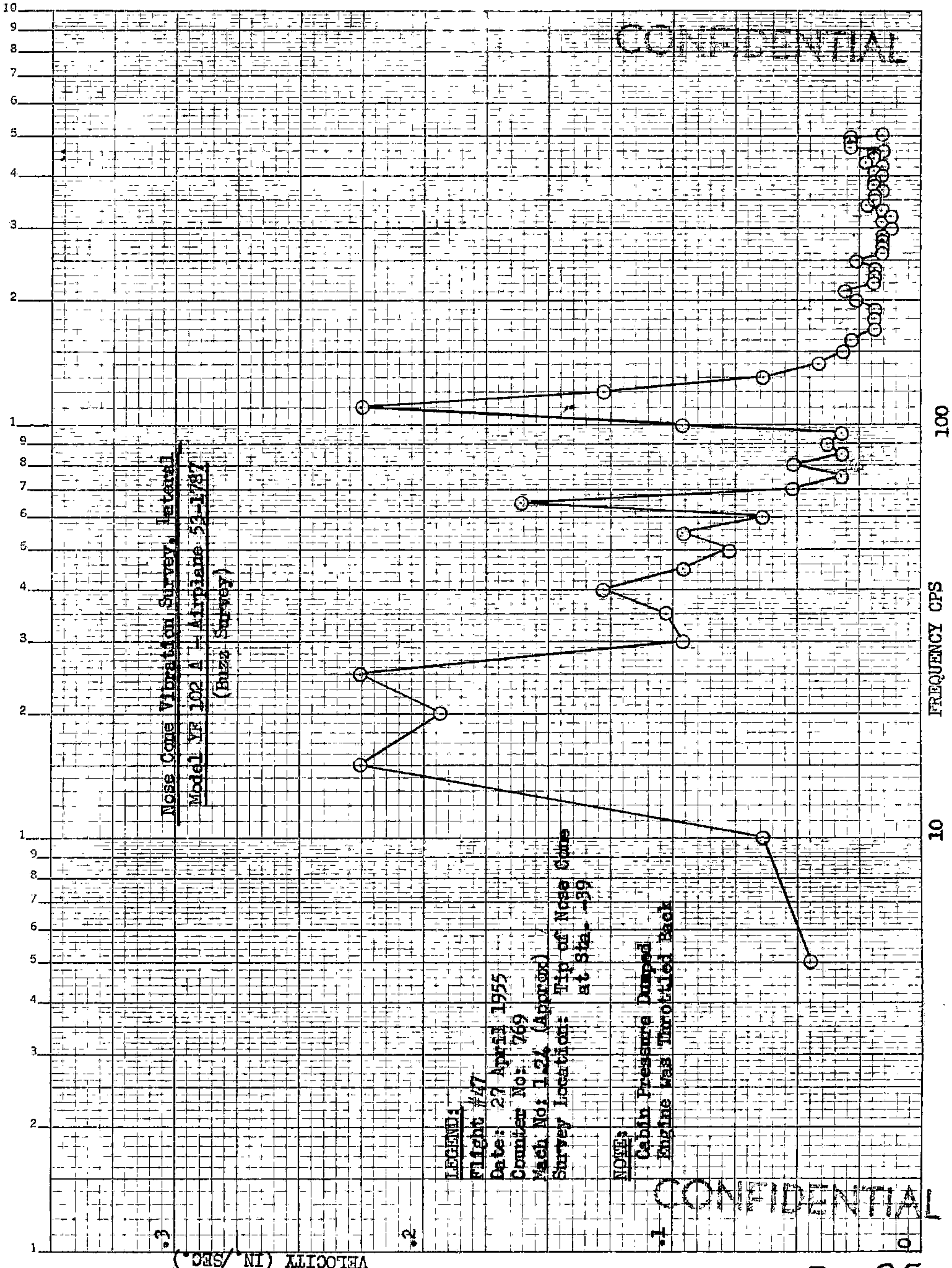
CONFIDENTIAL



CONFIDENTIAL

Fig. 84

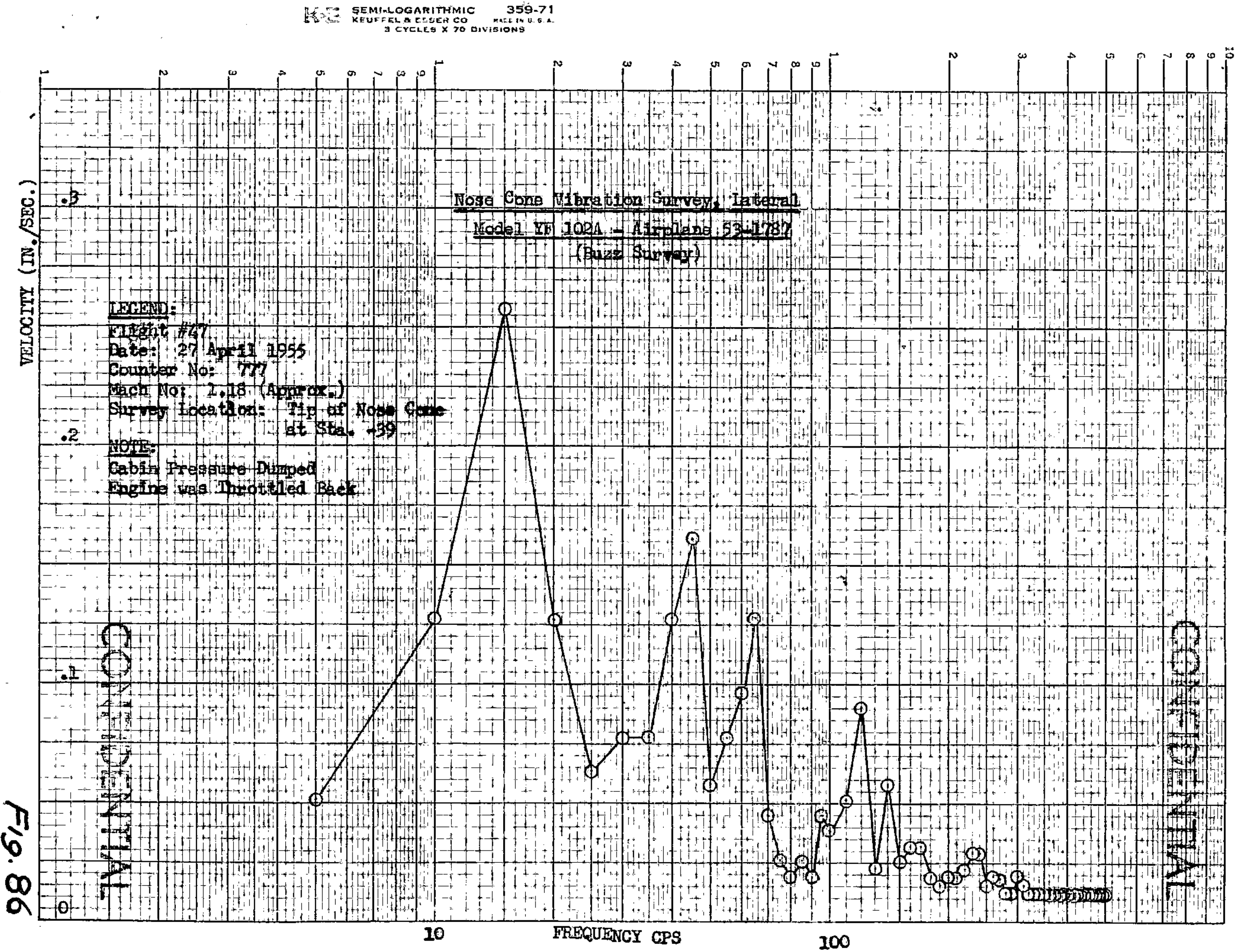
CONFIDENTIAL



CONFIDENTIAL

Fig. 85

CONFIDENTIAL



Nose Cone Vibration Survey, Lateral

Model YF 102A - Airplane 53-1787

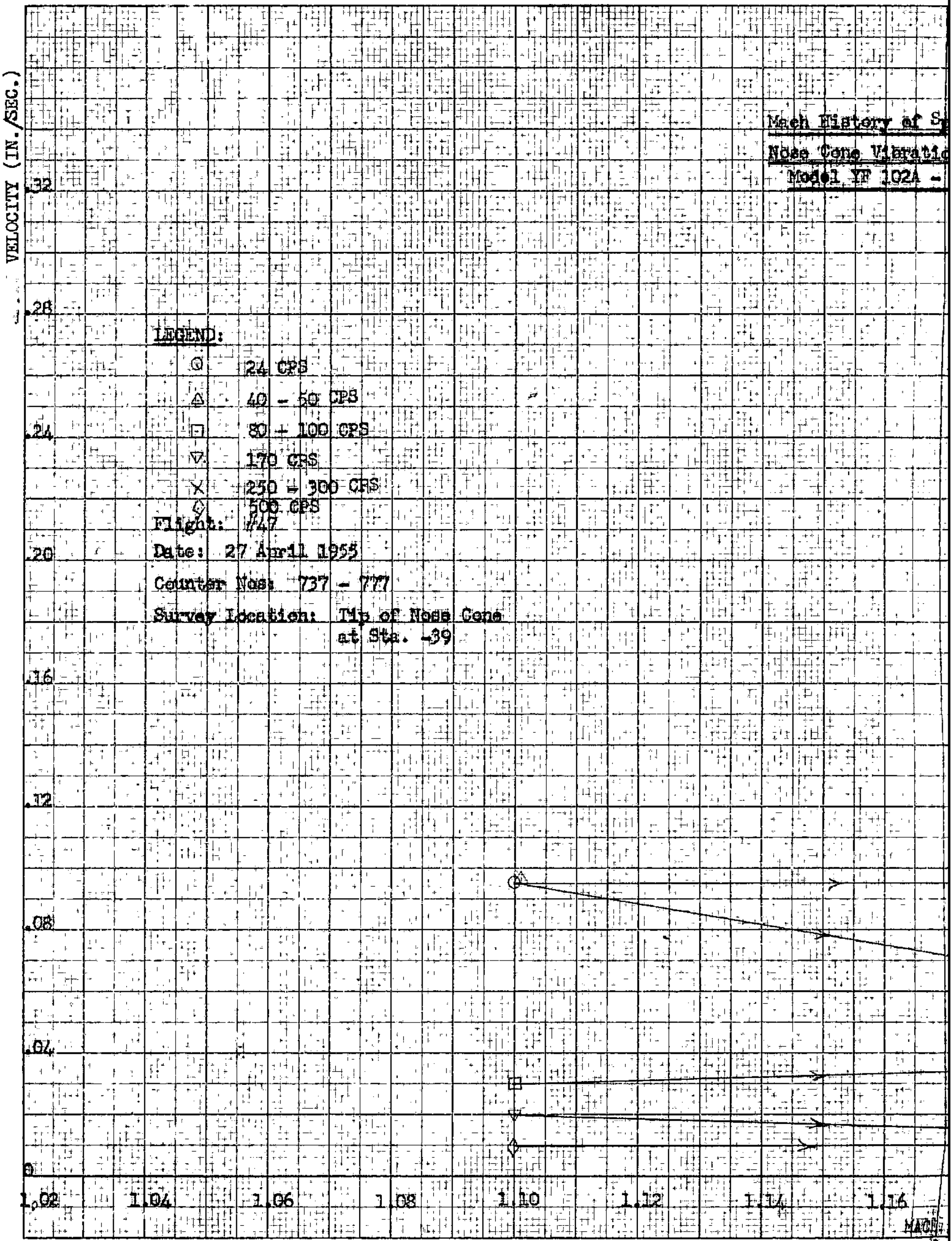
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 777
Mach No: 1.18 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

NOTE:
Cabin Pressure Dumped
Engine was Throttled Back

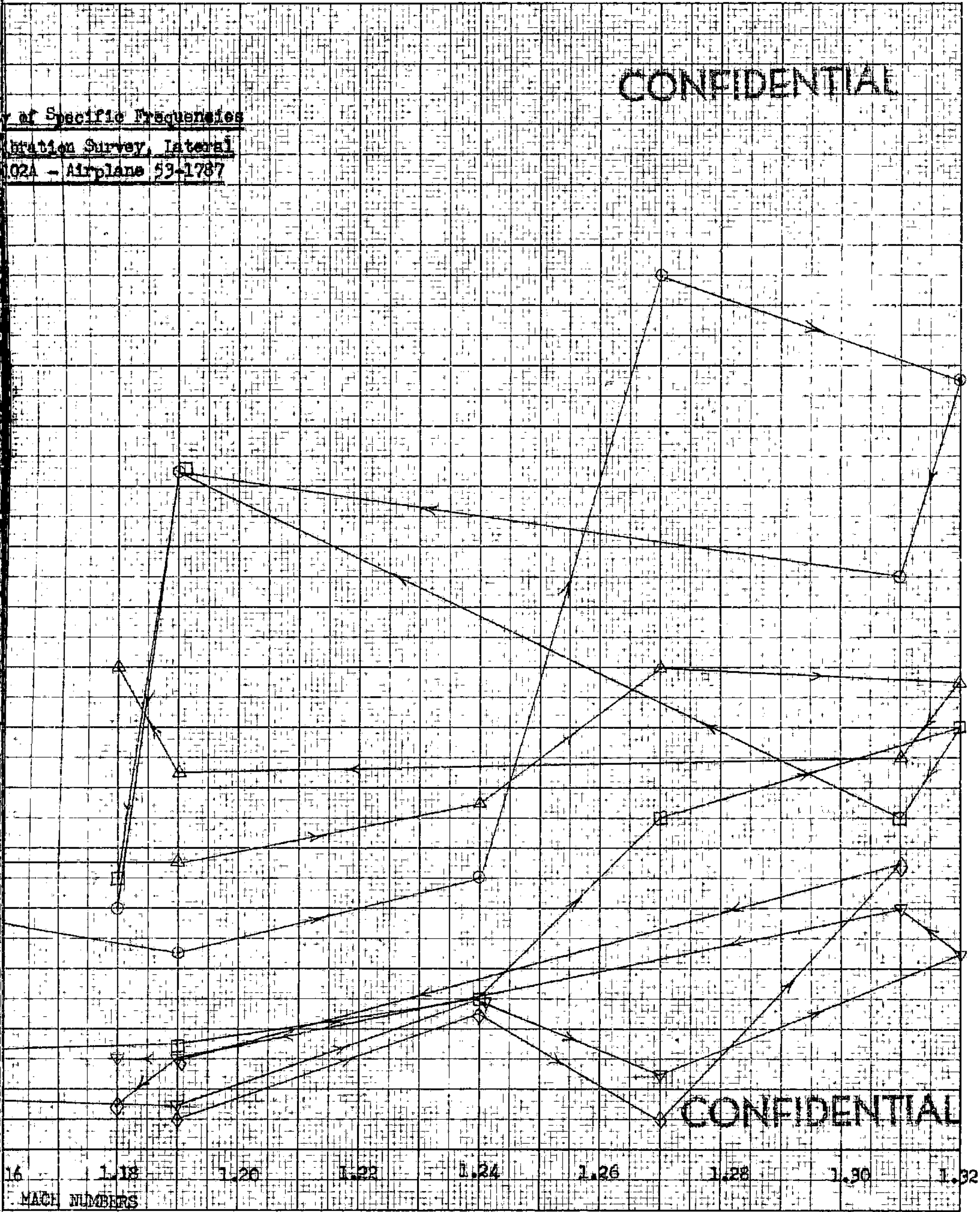
CONFIDENTIAL

Fig. 86



10 X 10 TO THE 1/2 INCH 359-11L
KEUFFEL & ESSER CO.

CONFIDENTIAL

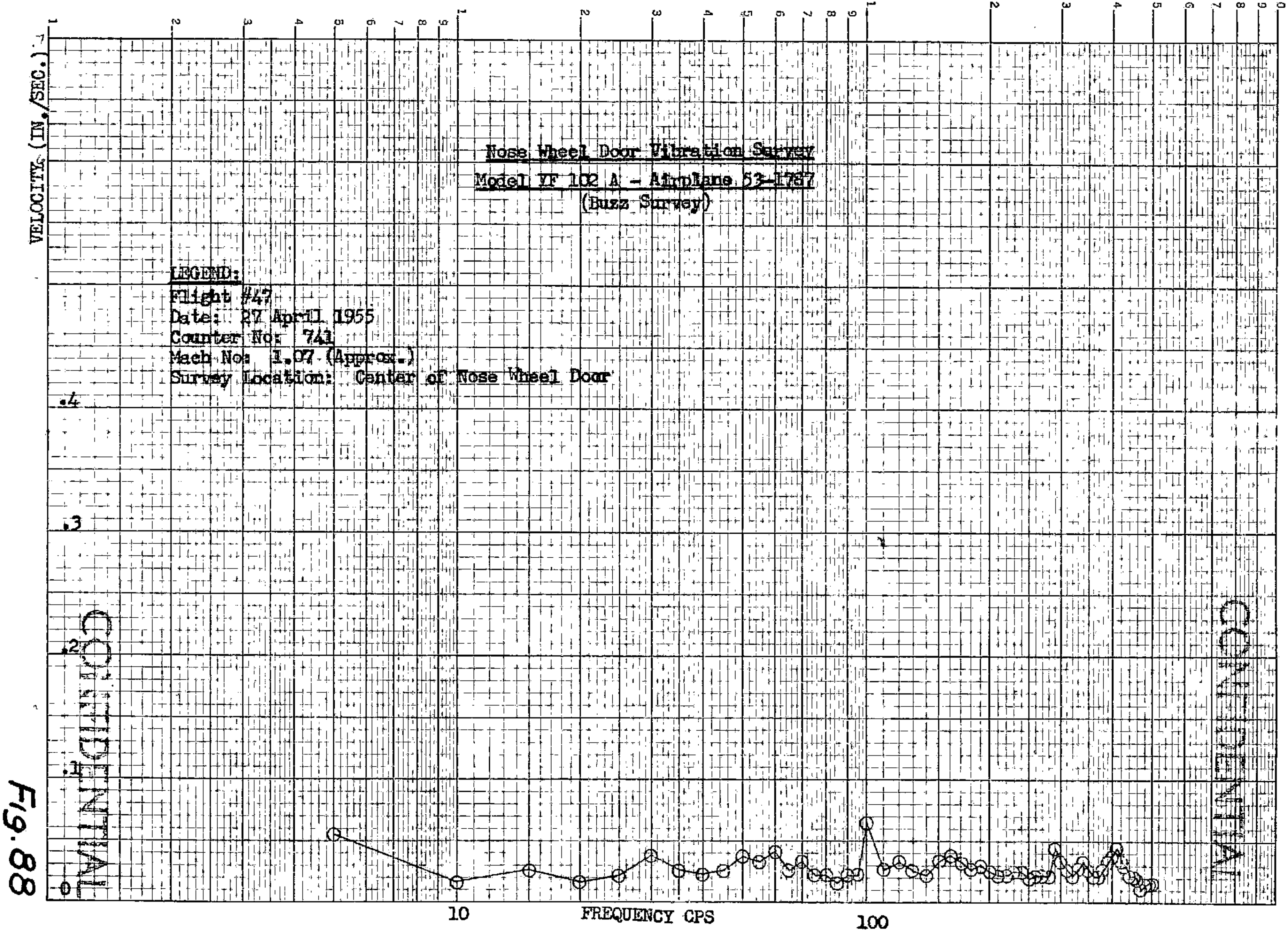


y of Specific Frequencies
bration Survey, Lateral
02A - Airplane 53-1787

CONFIDENTIAL

Fig. 87

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



Nose Wheel Door Vibration Survey

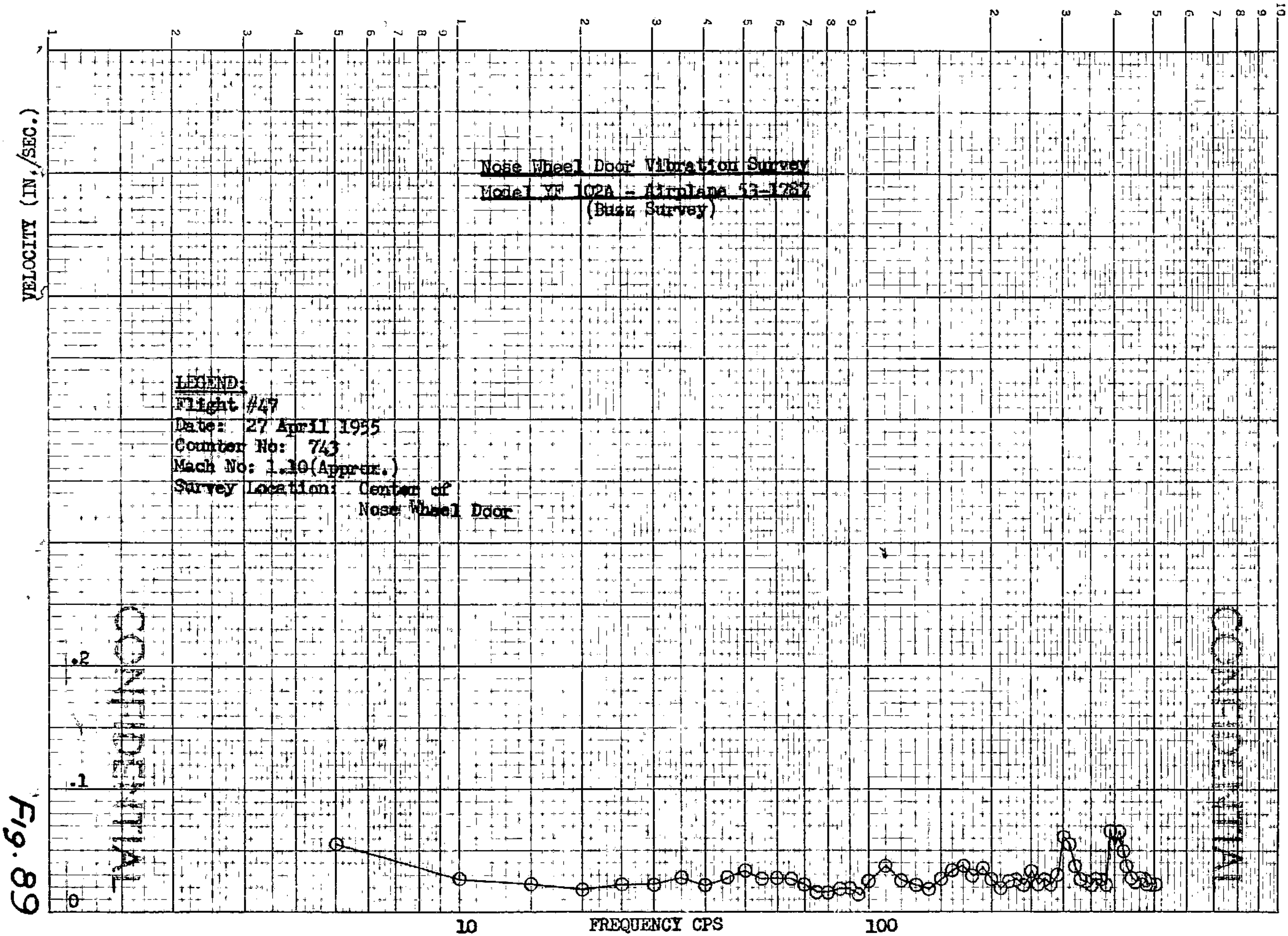
Model YF 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 741
Mach No: 1.07 (Approx.)
Survey Location: Center of Nose Wheel Door

Fig. 88

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



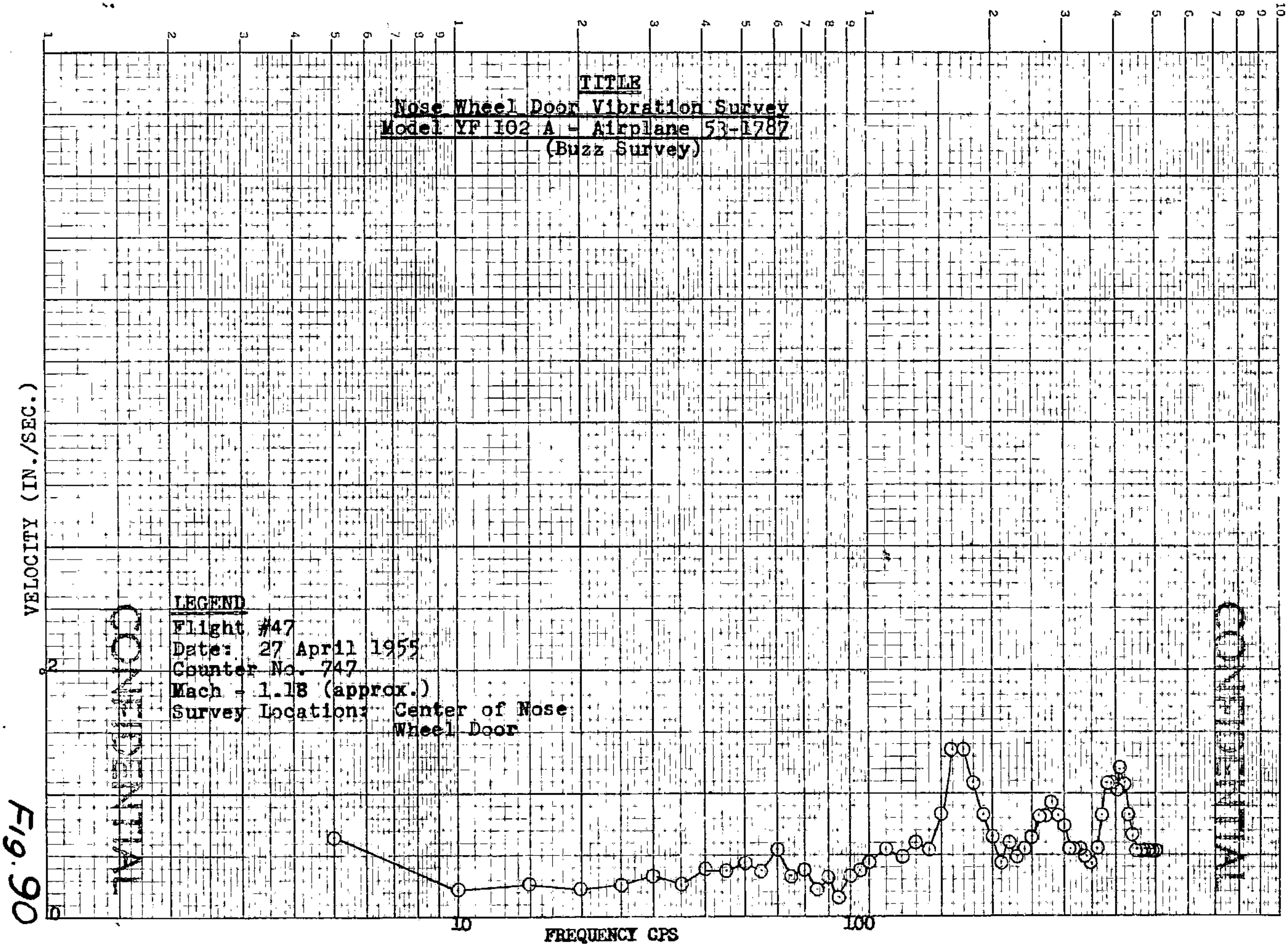
Nose Wheel Door Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 743
Mach No: 1.10 (Apprux.)
Survey Location: Center of
Nose Wheel Door

Fig. 89

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

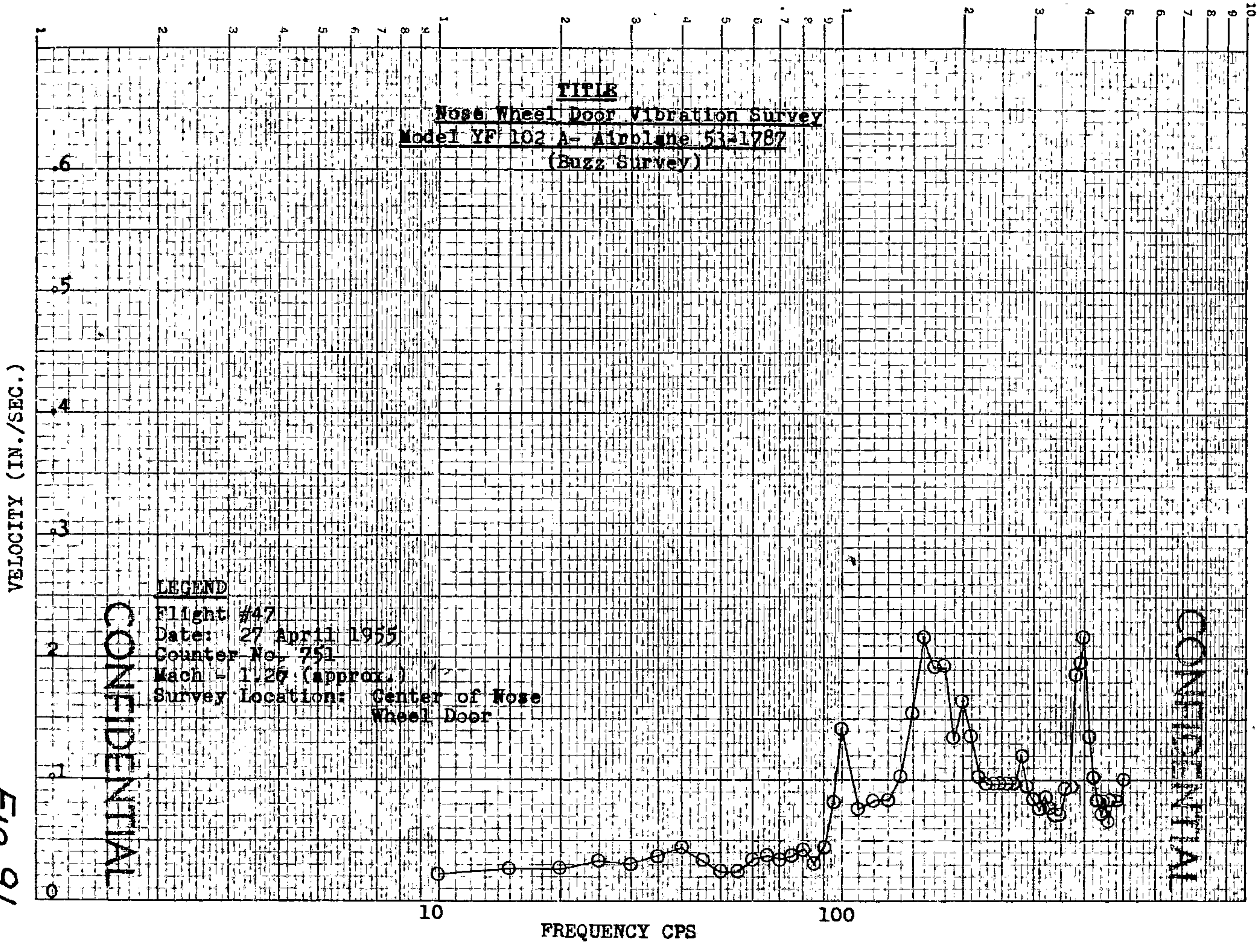
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 747
Mach - 1.18 (approx.)
Survey Location: Center of Nose Wheel Door

CONFIDENTIAL

Fig. 90

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO., MADE IN U.S.A.
3 Cycles x 70 Divisions

CONFIDENTIAL



TITLE

Nose Wheel Door Vibration Survey
Model YF 102 A- Airplane 53-1787
(Buzz Survey)

LEGEND

Flight #47
Date: 27 April 1955
Counter No. 751
Mach - 1.26 (approx.)
Survey Location: Center of Nose Wheel Door

CONFIDENTIAL

Fig. 91

CONFIDENTIAL

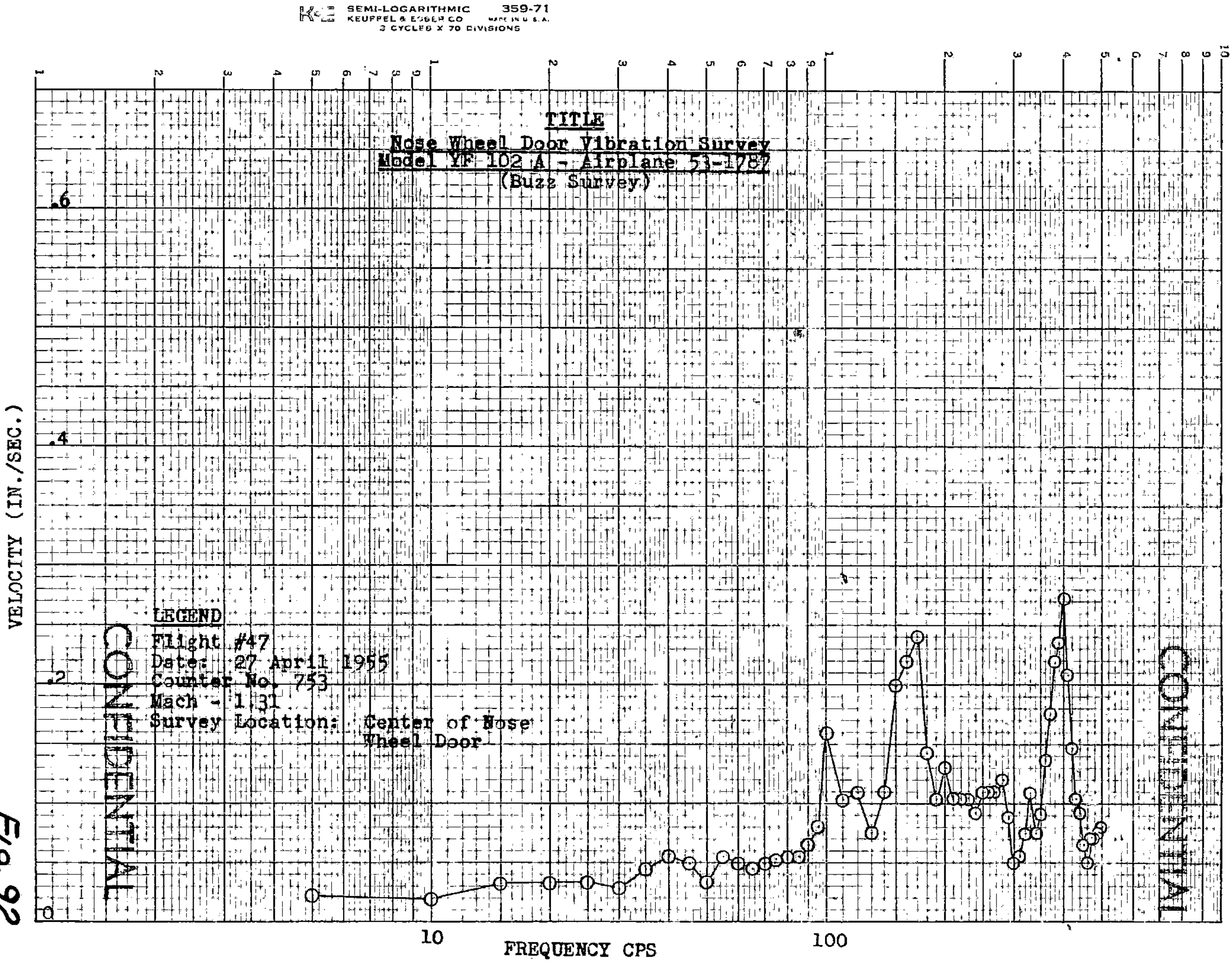
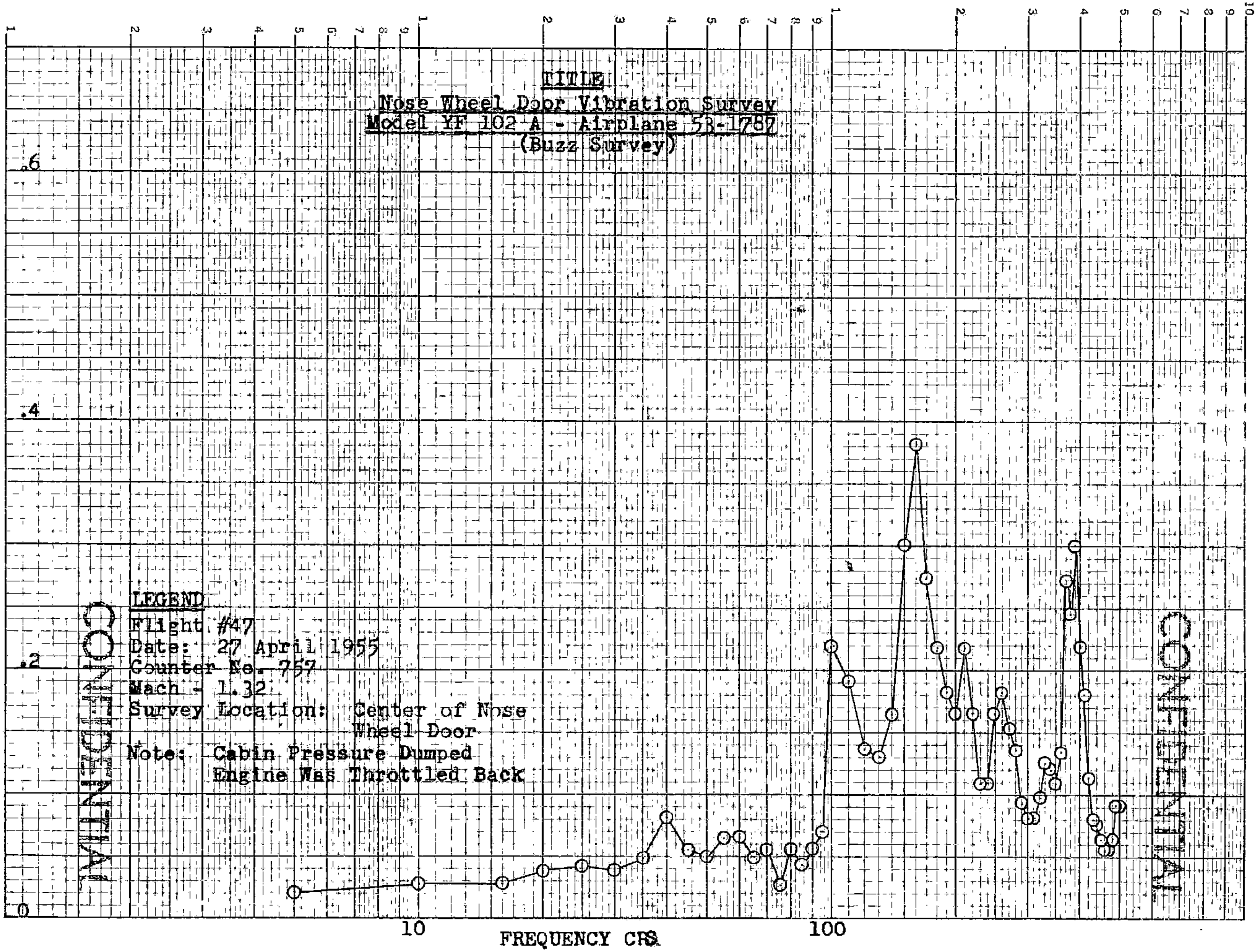


Fig. 92

CONFIDENTIAL

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

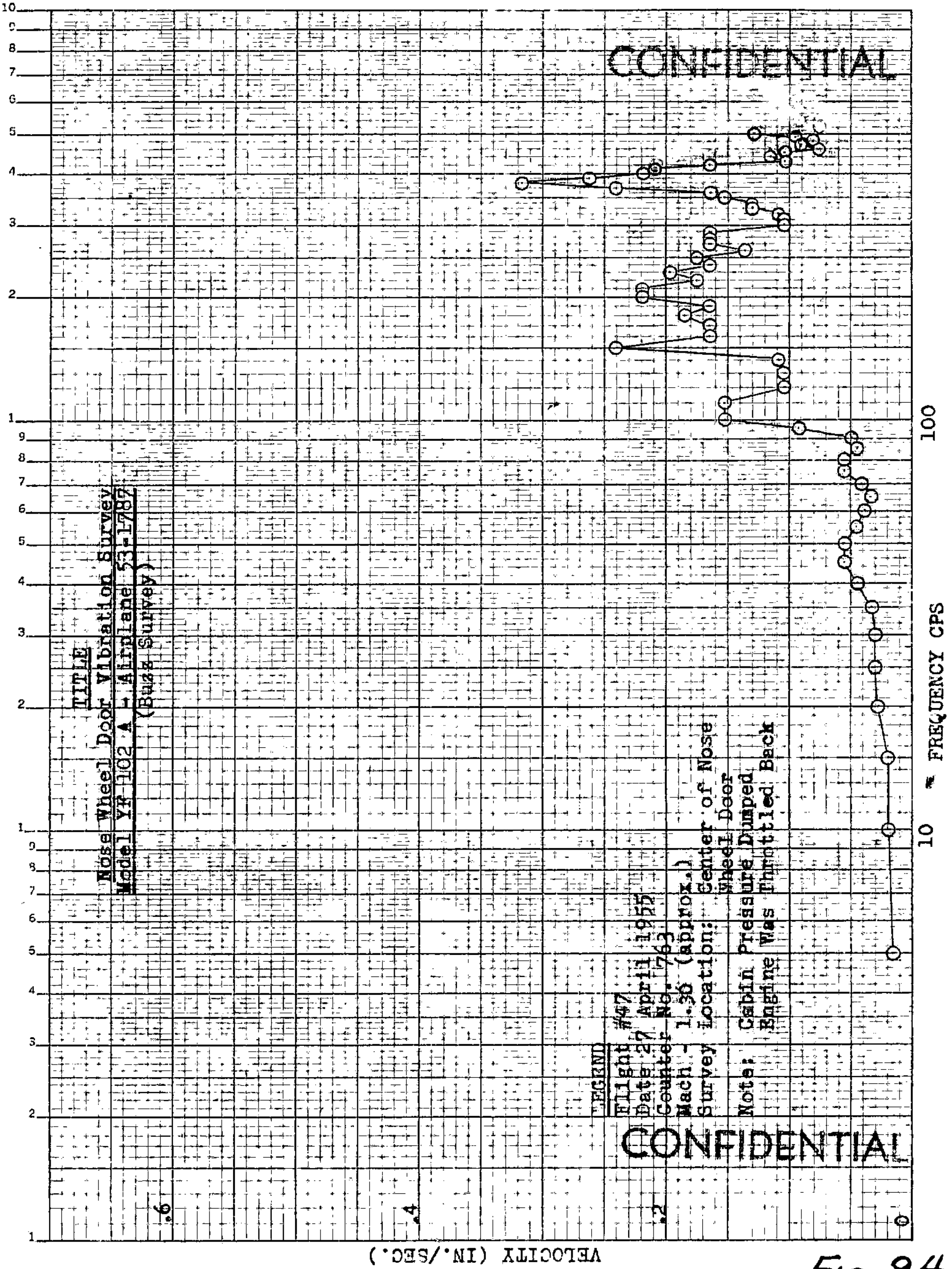
CONFIDENTIAL



CONFIDENTIAL

Fig. 93

CONFIDENTIAL



TITLE
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date 27 April 1955
Counter No. 763
Mach - 1.30 (approx.)
Survey Location: Center of Nose
Wheel Door
Note: Cabin Pressure Dumped
Engine Was Throttled Back

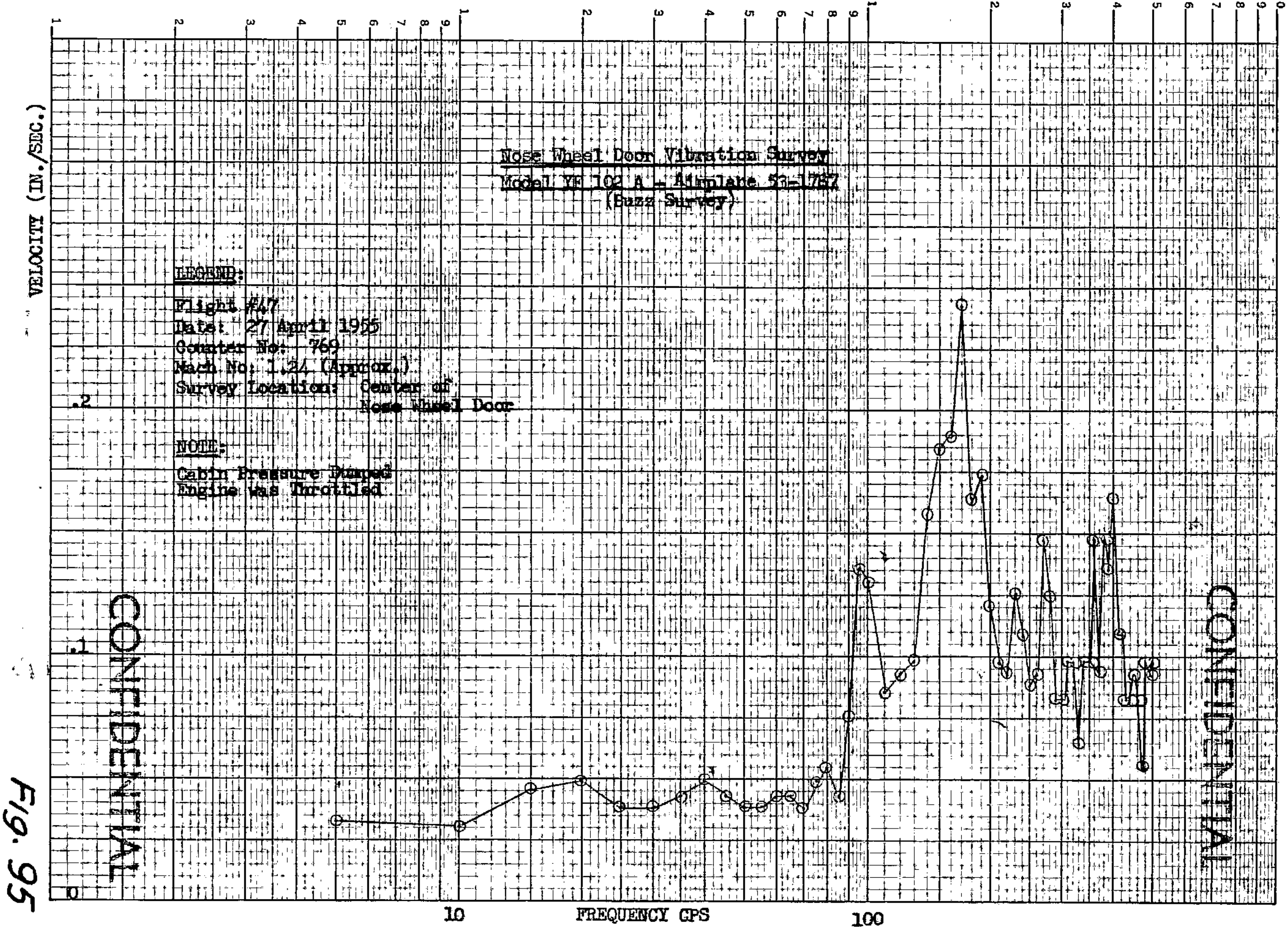
CONFIDENTIAL

Fig. 94

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:

Flight #47
Date: 27 April 1955
Counter No: 769
Mach No: 1.24 (Approx.)
Survey Location: Center of Nose Wheel Door

NOTE:

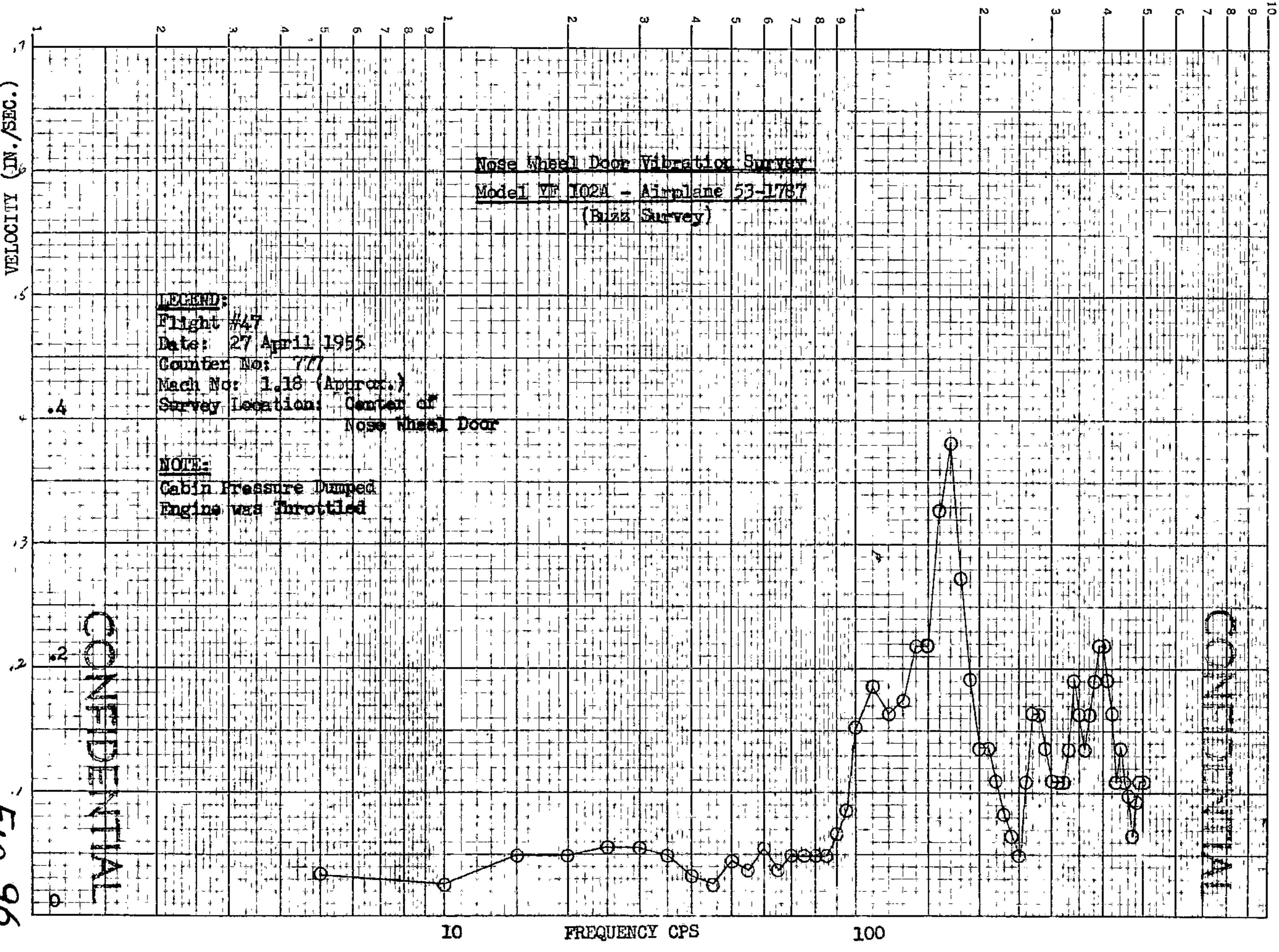
Cabin Pressure Dumped
Engine was Throttled

CONFIDENTIAL

Fig. 95

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
2 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Wheel Door Vibration Survey

Model YF 102A - Airplane 53-1787

(Buzz Survey)

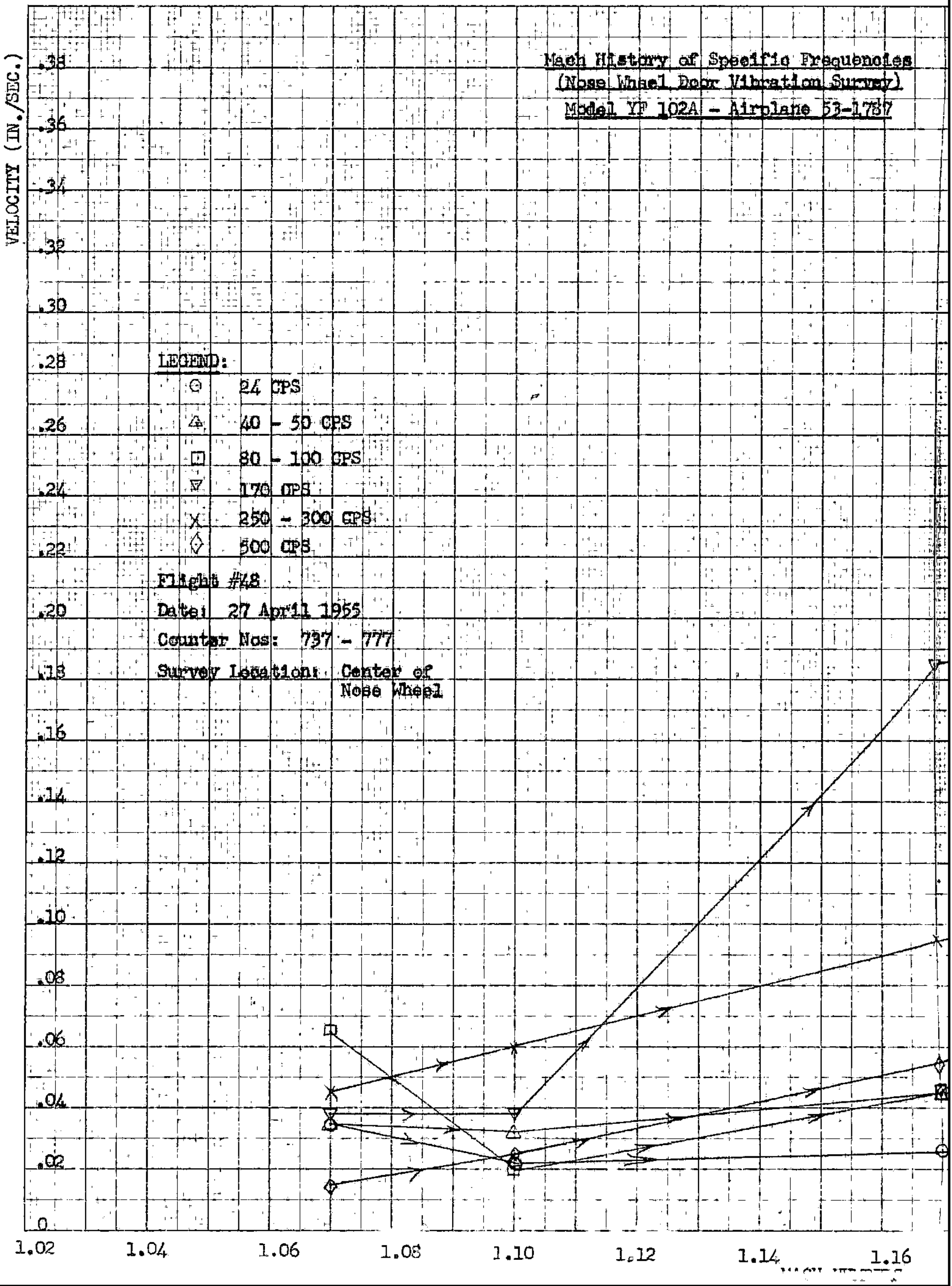
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 777
Mach No: 1.18 (Approx.)
Survey Location: Center of
Nose Wheel Door

NOTE:
Cabin Pressure Dumped
Engine was Throttled

CONFIDENTIAL

Fig. 96

10 X 10 TO THE 1/2 INCH 359-11L
KEUFFEL & ESSER CO.



Mach History of Specific Frequencies
(Nose Wheel Door Vibration Survey)
Model YF 102A - Airplane 53-1787

LEGEND:

⊖	24 CPS
△	40 - 50 CPS
□	80 - 100 CPS
▽	170 CPS
X	250 - 300 CPS
◇	500 CPS

Flight #48
Date: 27 April 1955
Counter Nos: 737 - 777
Survey Location: Center of Nose Wheel

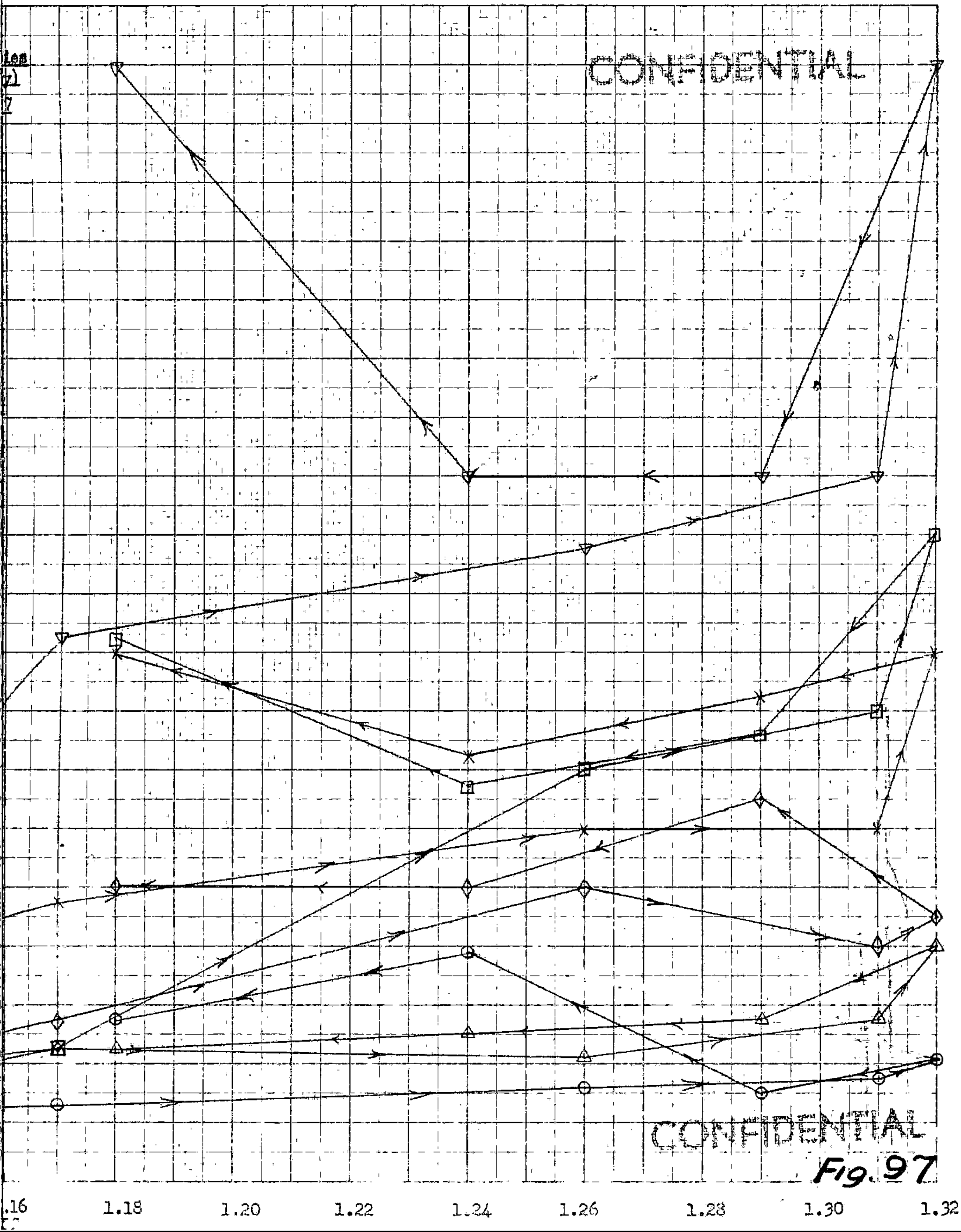
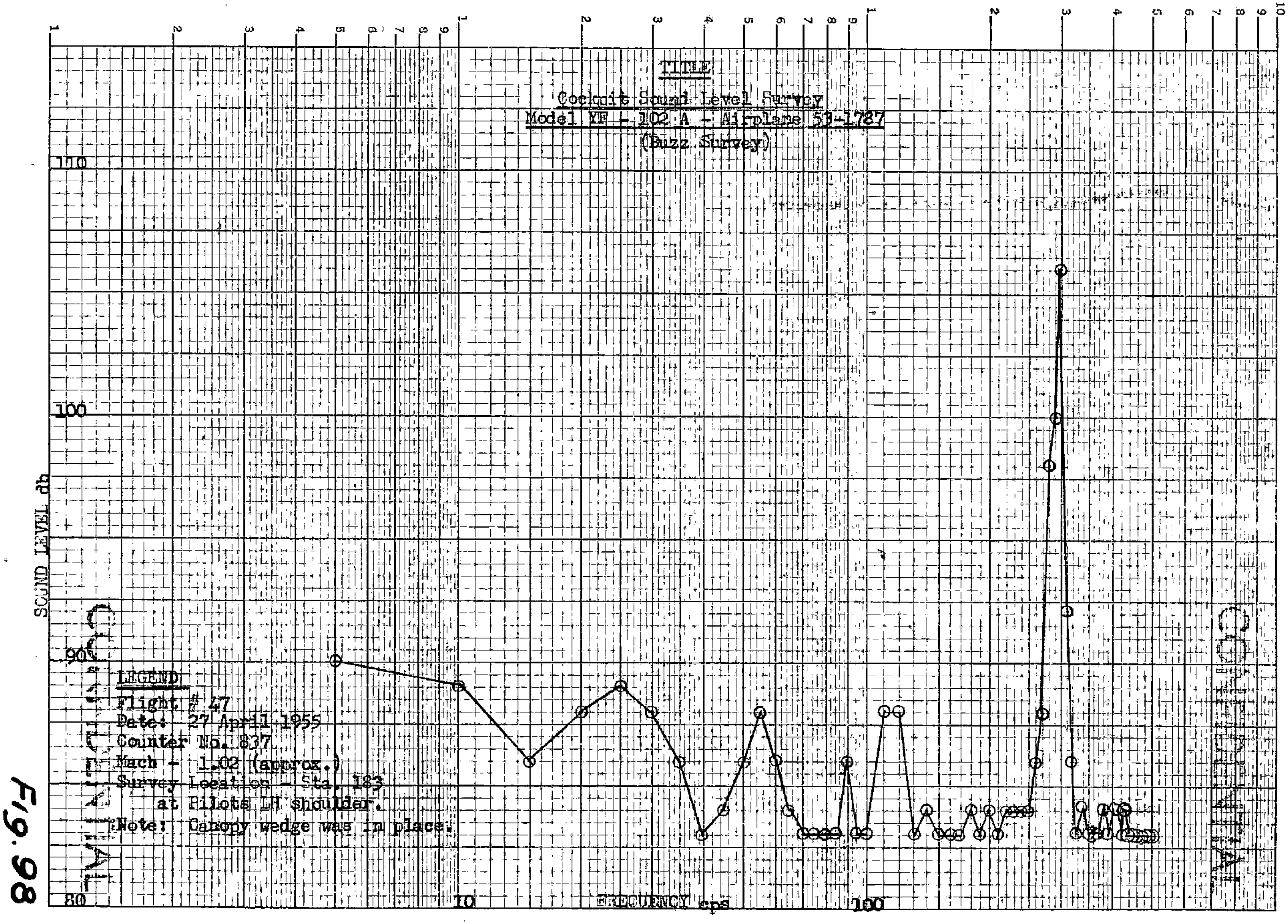


Fig. 97

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



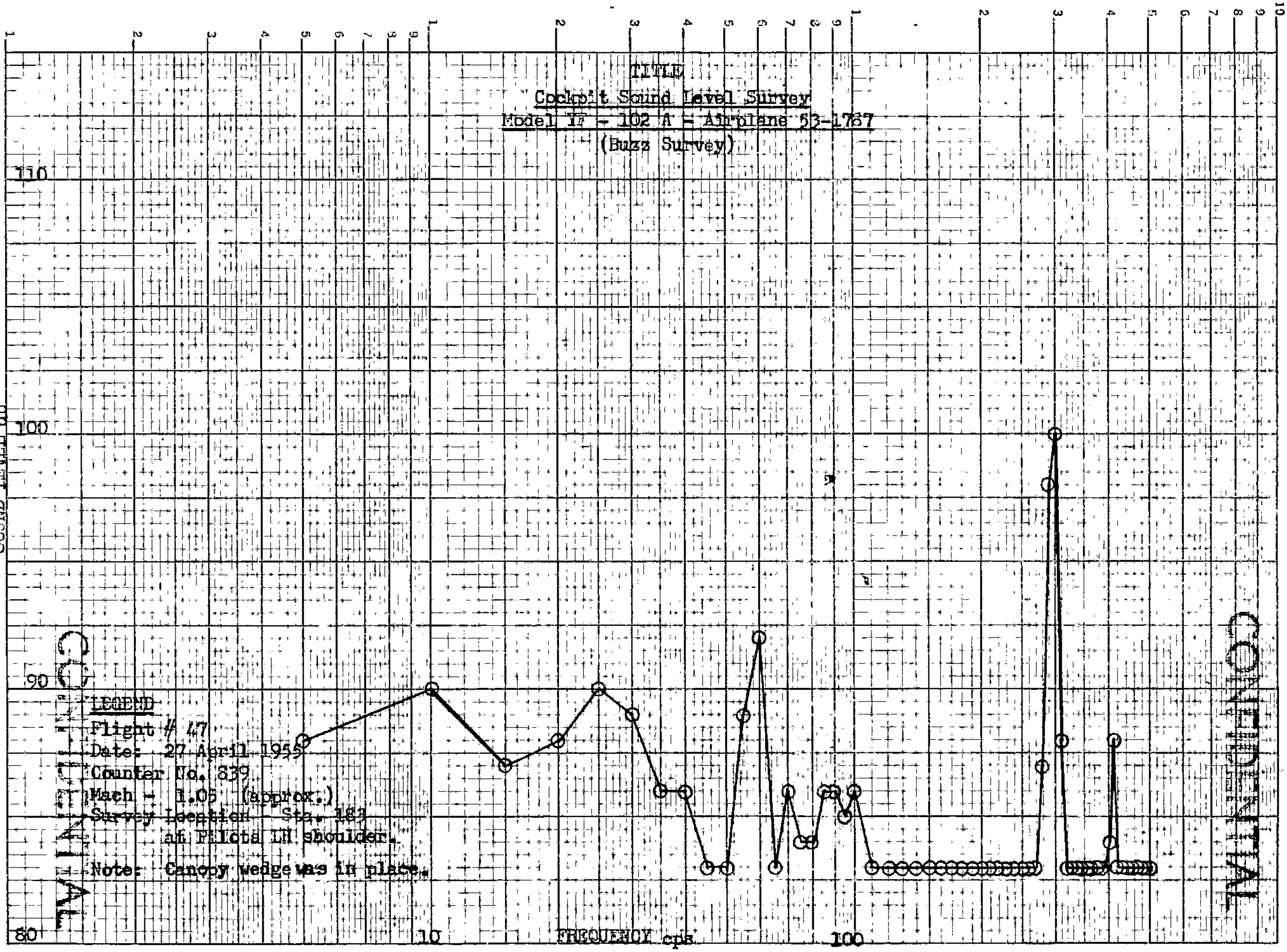
TITLE
Cockpit Sound Level Survey
Model YF - 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 837
Mach - 1.02 (approx.)
Survey Location - Sta. 183
at Pilots LH shoulder.
Note: Canopy wedge was in place.

Fig. 98

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Cockpit Sound Level Survey

Model YF - 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND

Flight # 47

Date: 27 April 1955

Counter No. 839

Mach - 1.05 (approx.)

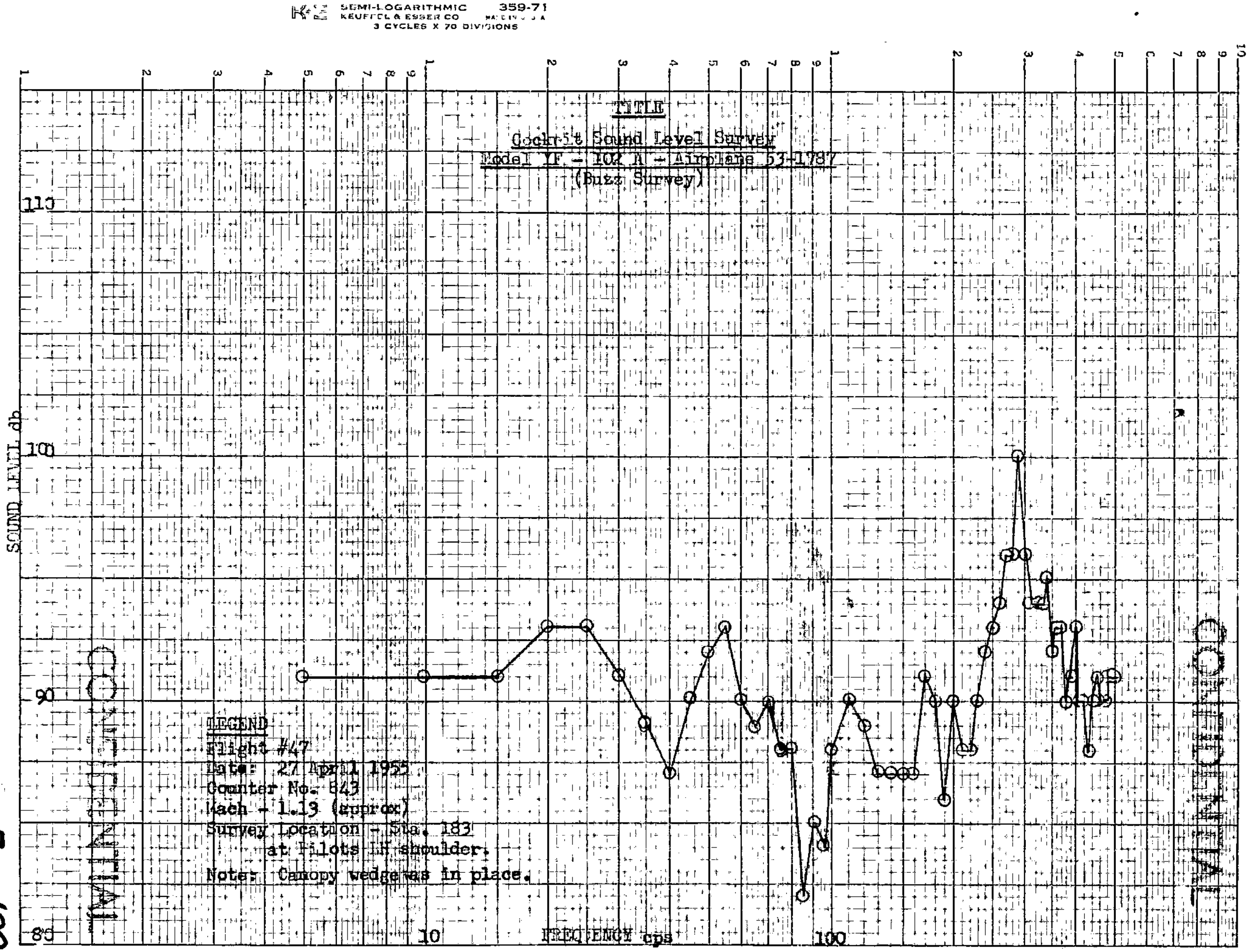
Survey Location - Sta. 183
at Pilots LH shoulder.

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 99

CONFIDENTIAL



TITLE

Cockpit Sound Level Survey
Model YF - 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 843
Mach - 1.13 (approx)
Survey Location - Sta. 183
at Pilots LH shoulder.
Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 100

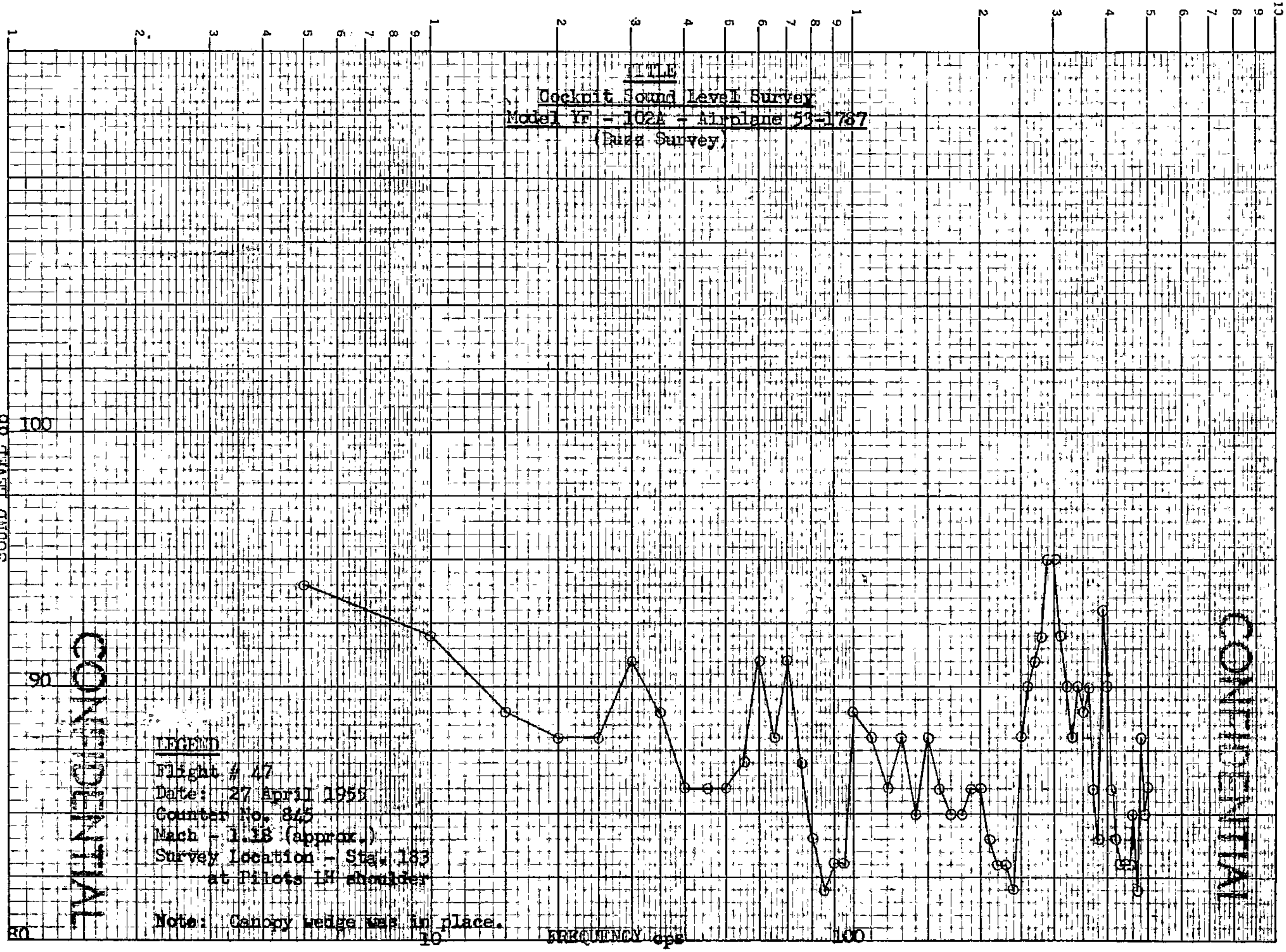
CONFIDENTIAL

TITLE

Cockpit Sound Level Survey

Model YF - 102A - Airplane 53-1787

(Buzz Survey)



LEGEND

Flight # 47

Date: 27 April 1955

Counter No. 845

Mach - 1.18 (approx.)

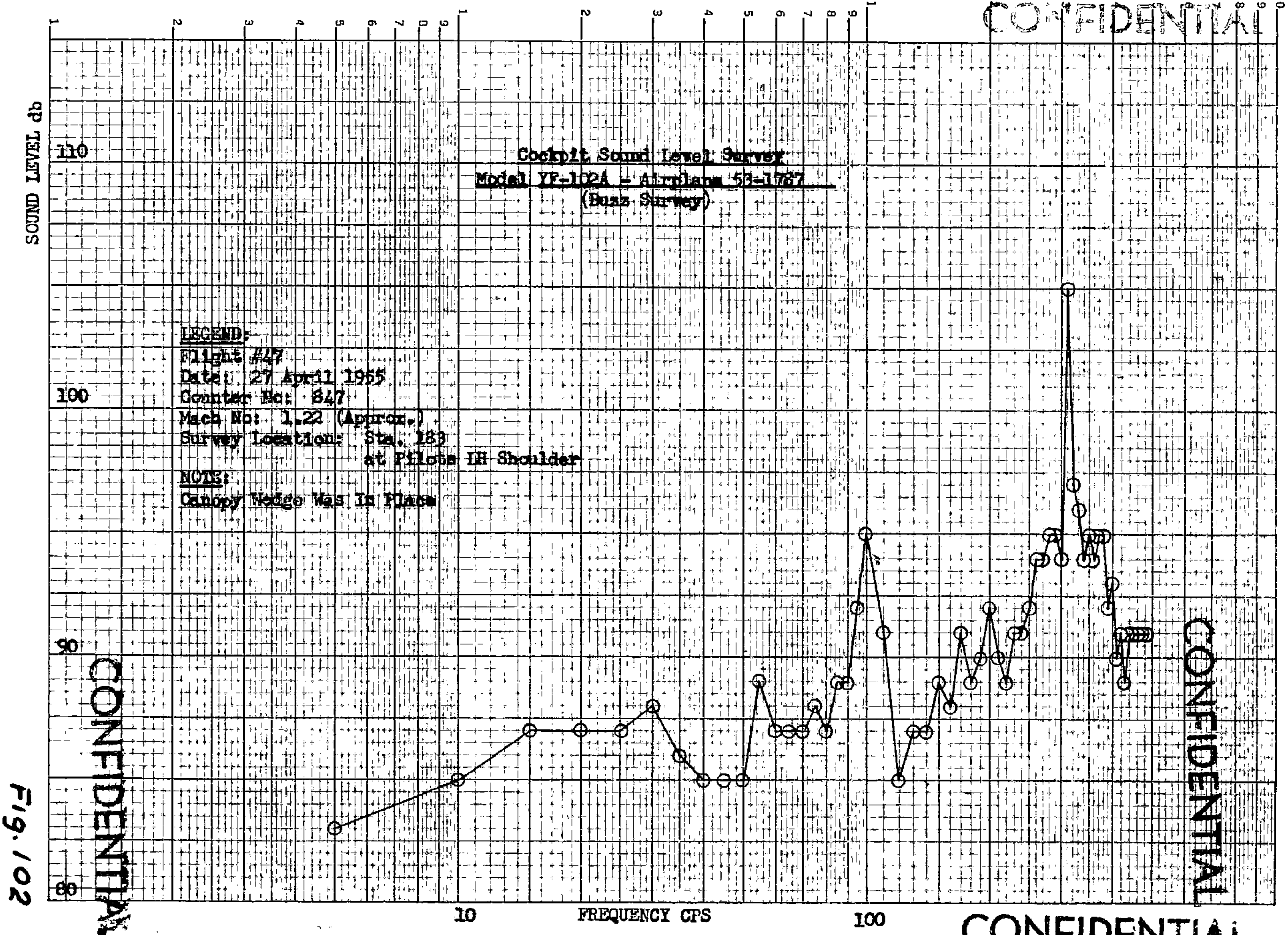
Survey Location - Sta. 183
at Pilots LH shoulder

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 101

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



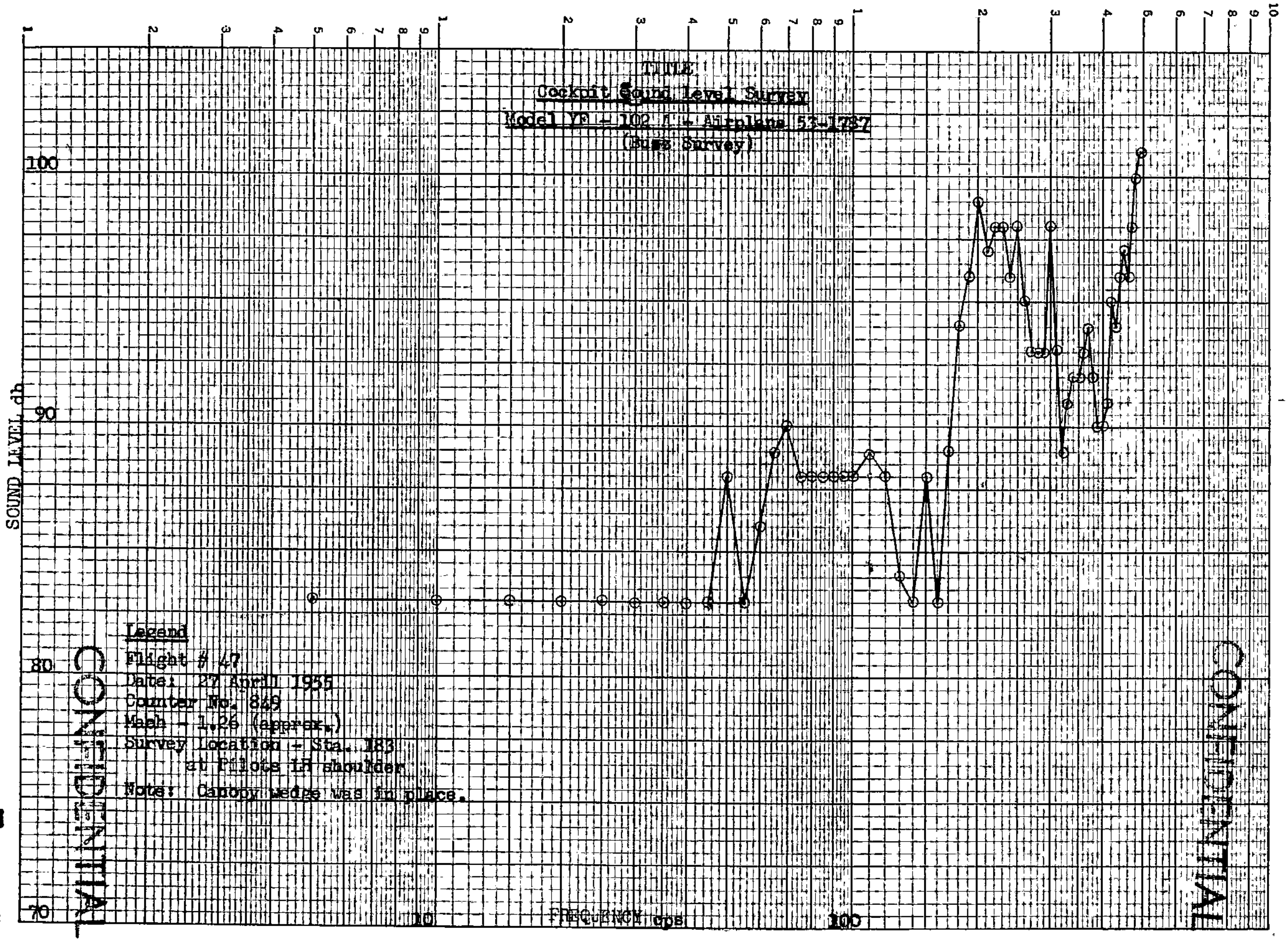
Cockpit Sound Level Survey
Model YF-102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 847
Mach No: 1.22 (Approx.)
Survey Location: Sta. 183
at Pilots LH Shoulder

NOTE:
Canopy Wedge Was In Place

Fig. 102

CONFIDENTIAL

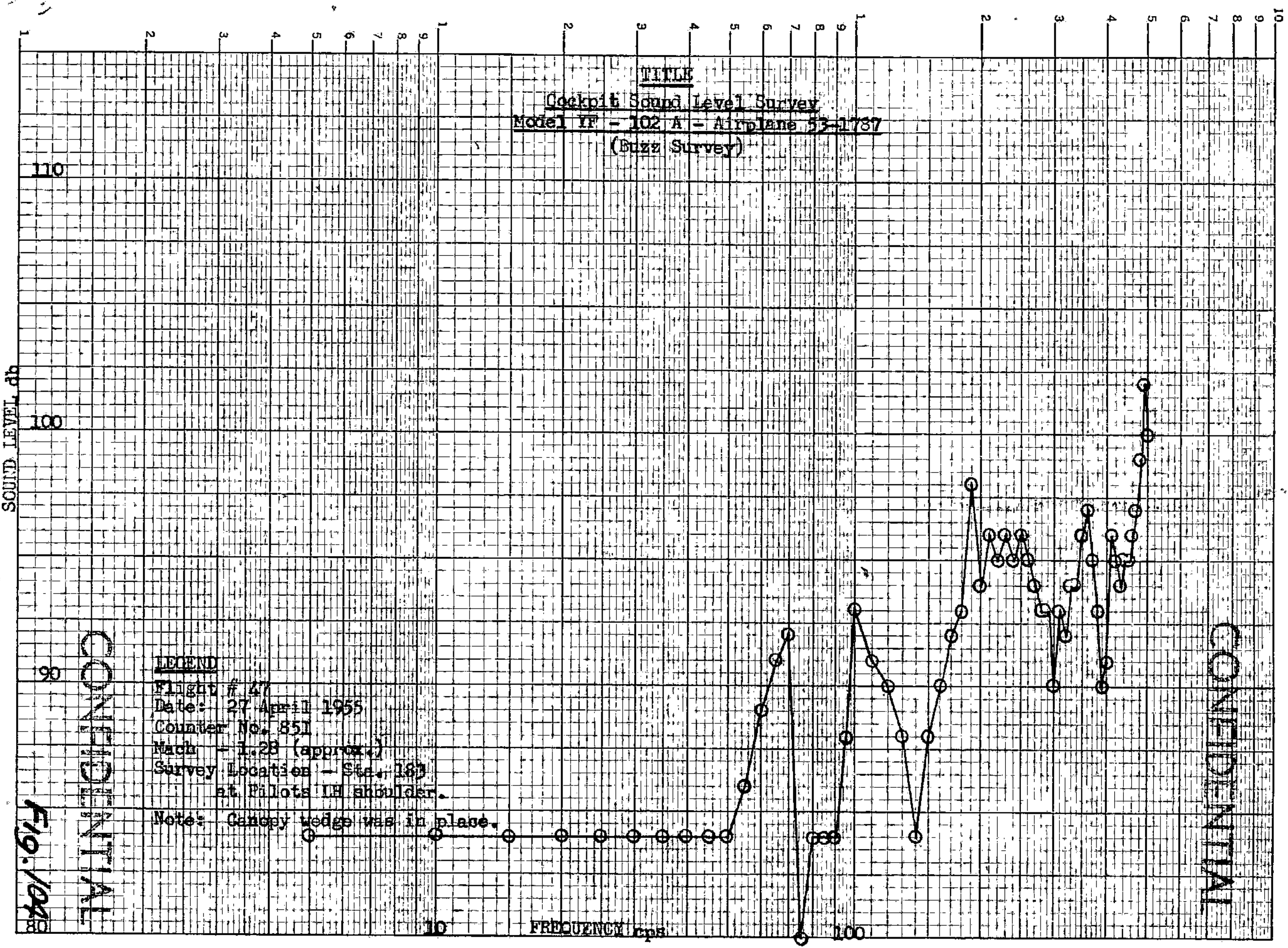


CONFIDENTIAL

Fig. 103

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



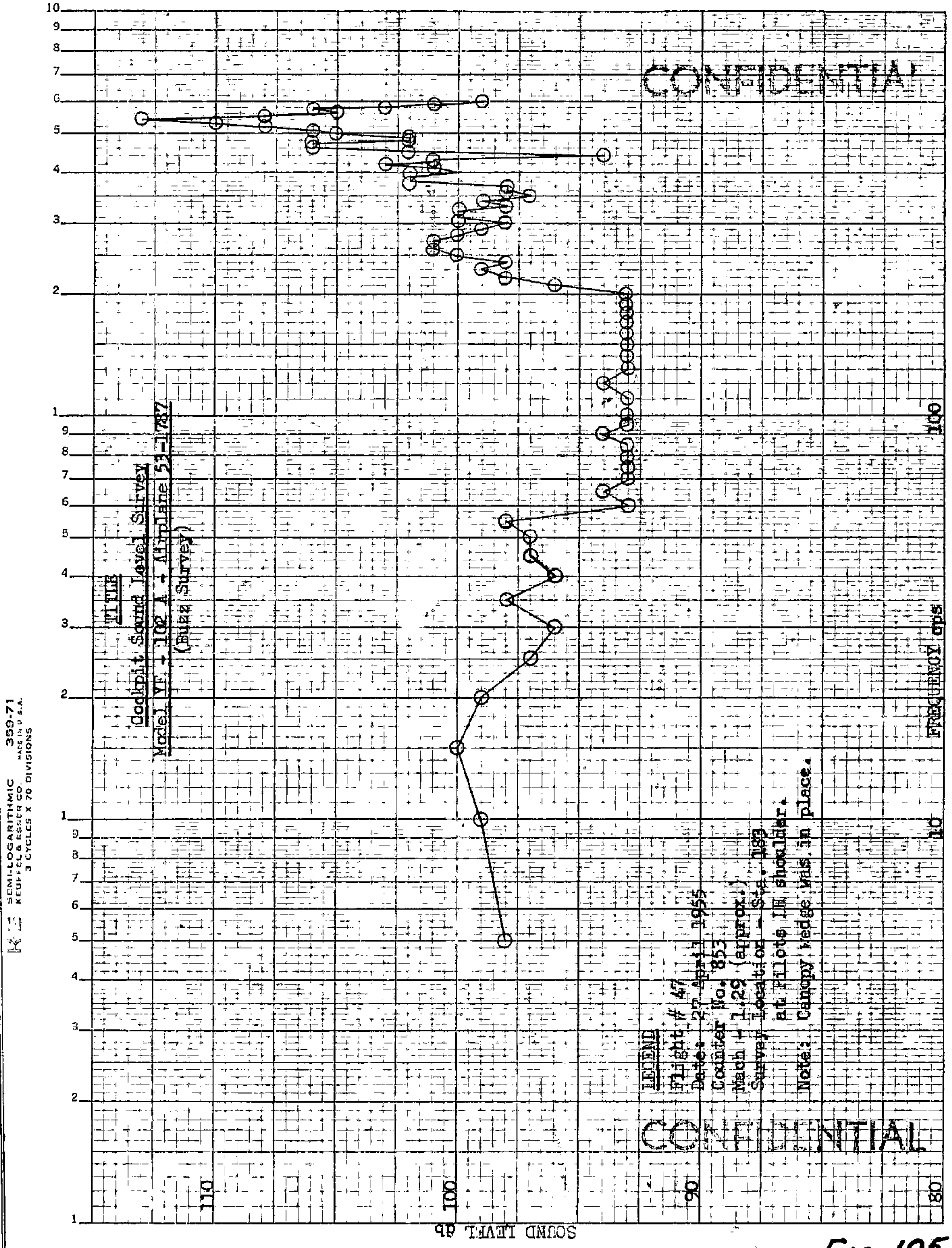
TITLE
Cockpit Sound Level Survey
Model YF - 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 851
Mach - 1.28 (approx.)
Survey Location - Sta. 183
at Pilots LH shoulder.

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 104



TITLE

Cockpit Sound Level Survey
Model YF - 102 A - Airplane 53-1787
(Buzz Survey)

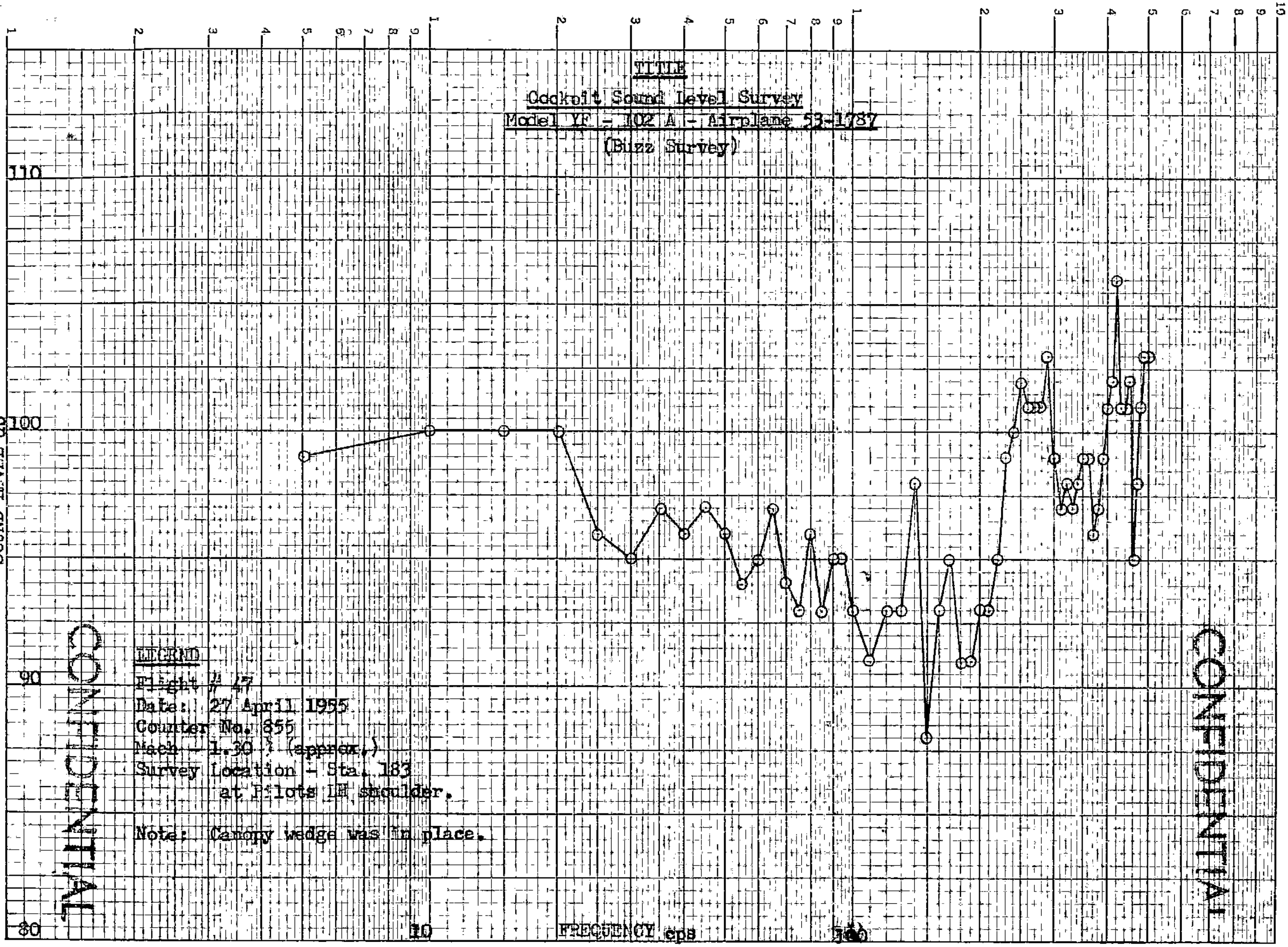
LEGEND

Flight # 47
Date: 27 April 1955
Counter No. 853
Mach - 1.29 (approx.)
Survey Location - Sta. 189
at Pilots LH shoulder.
Note: Canopy wedge was in place.

Fig. 105

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Cockpit Sound Level Survey
Model YF - 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 855
Mach - 1.30 ? (approx.)
Survey Location - Sta. 183
at Pilots LH shoulder.

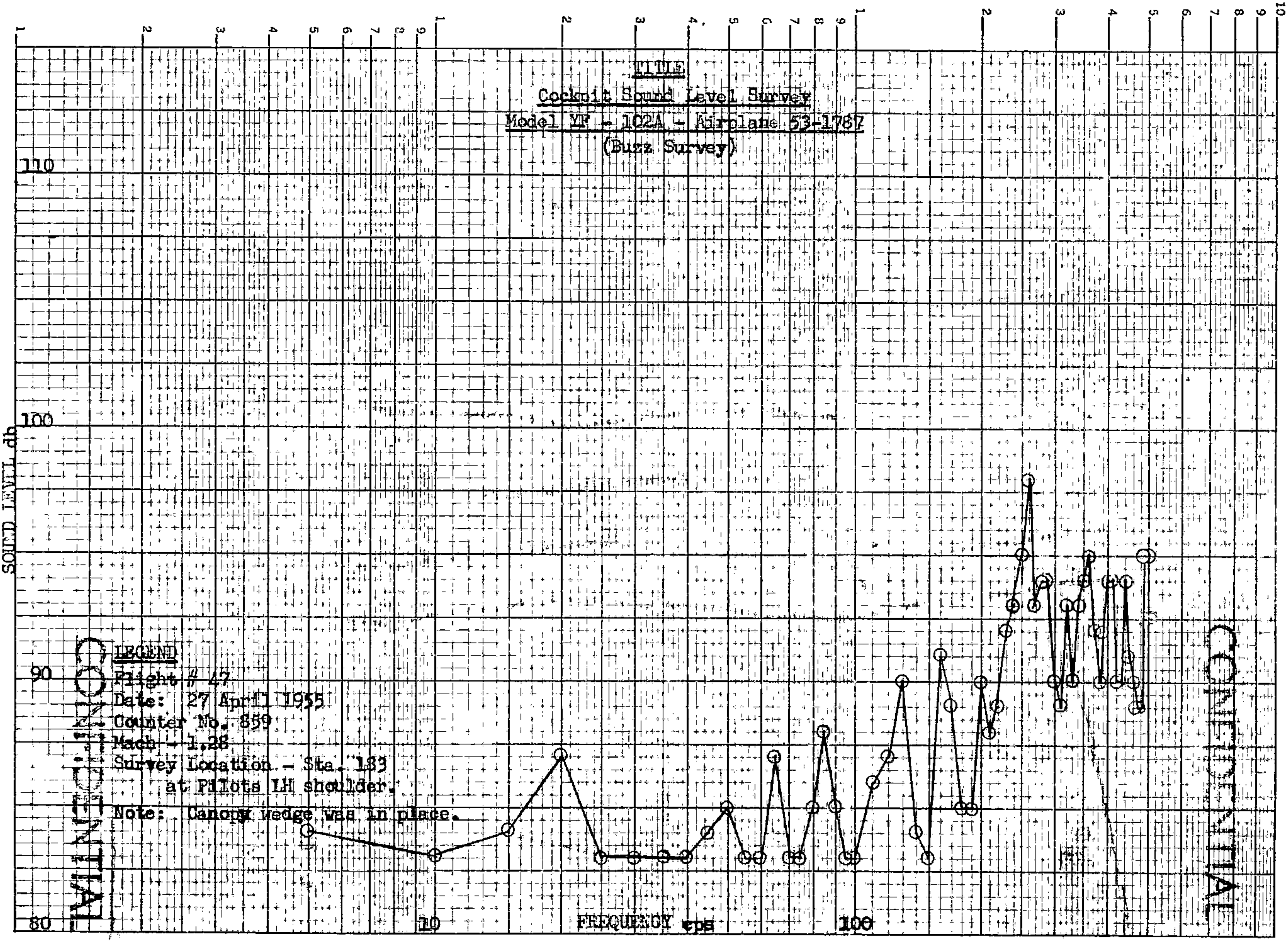
Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 106

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Cockpit Sound Level Survey
Model YF - 102A - Airplane 53-1787
(Buzz Survey)

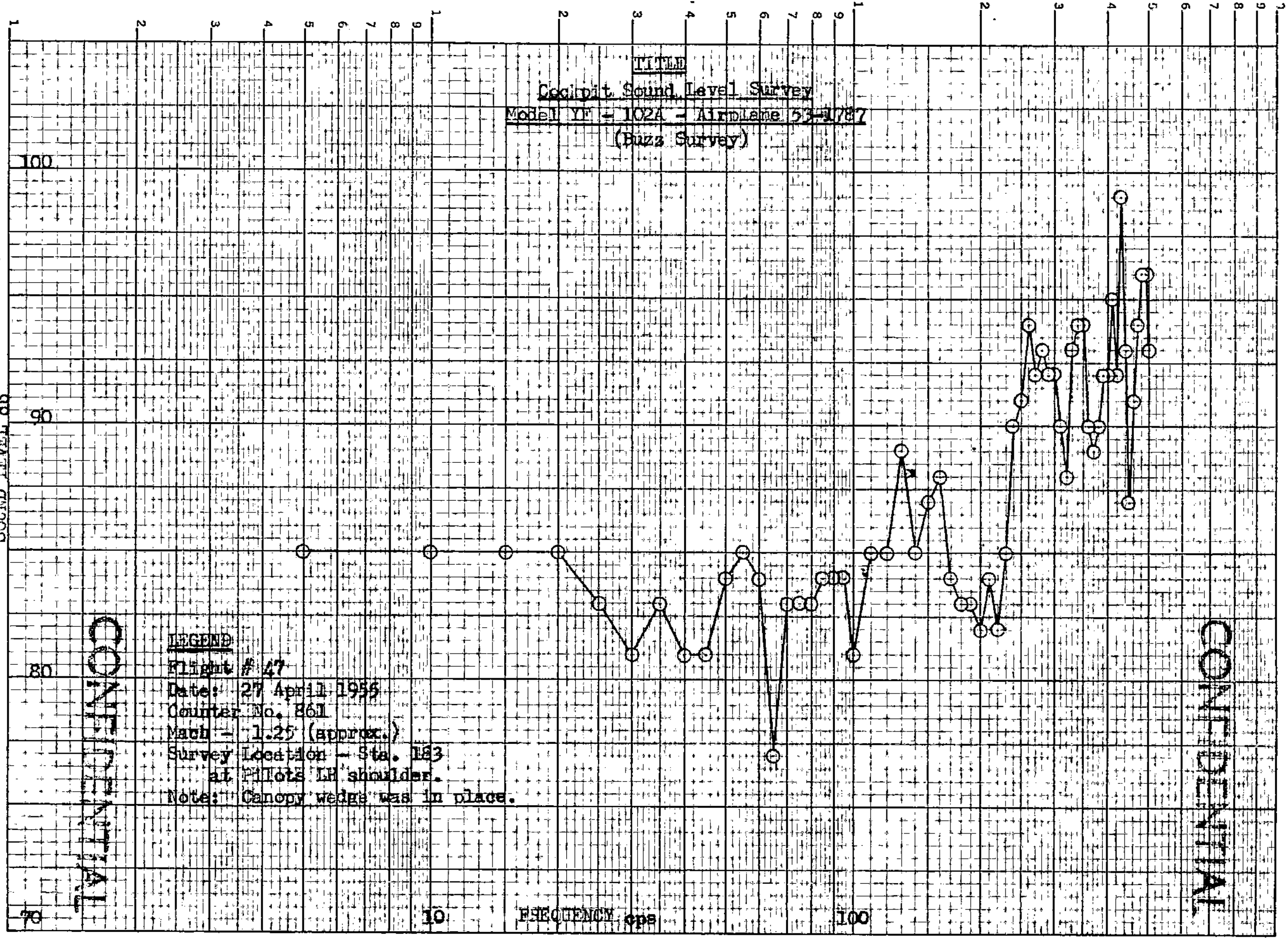
LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 859
Mach - 1.28
Survey Location - Sta. 133
at Pilots LH shoulder.
Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 107

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Cockpit Sound Level Survey
Model YF - 102A - Airplane 53-1787
(Buzz Survey)

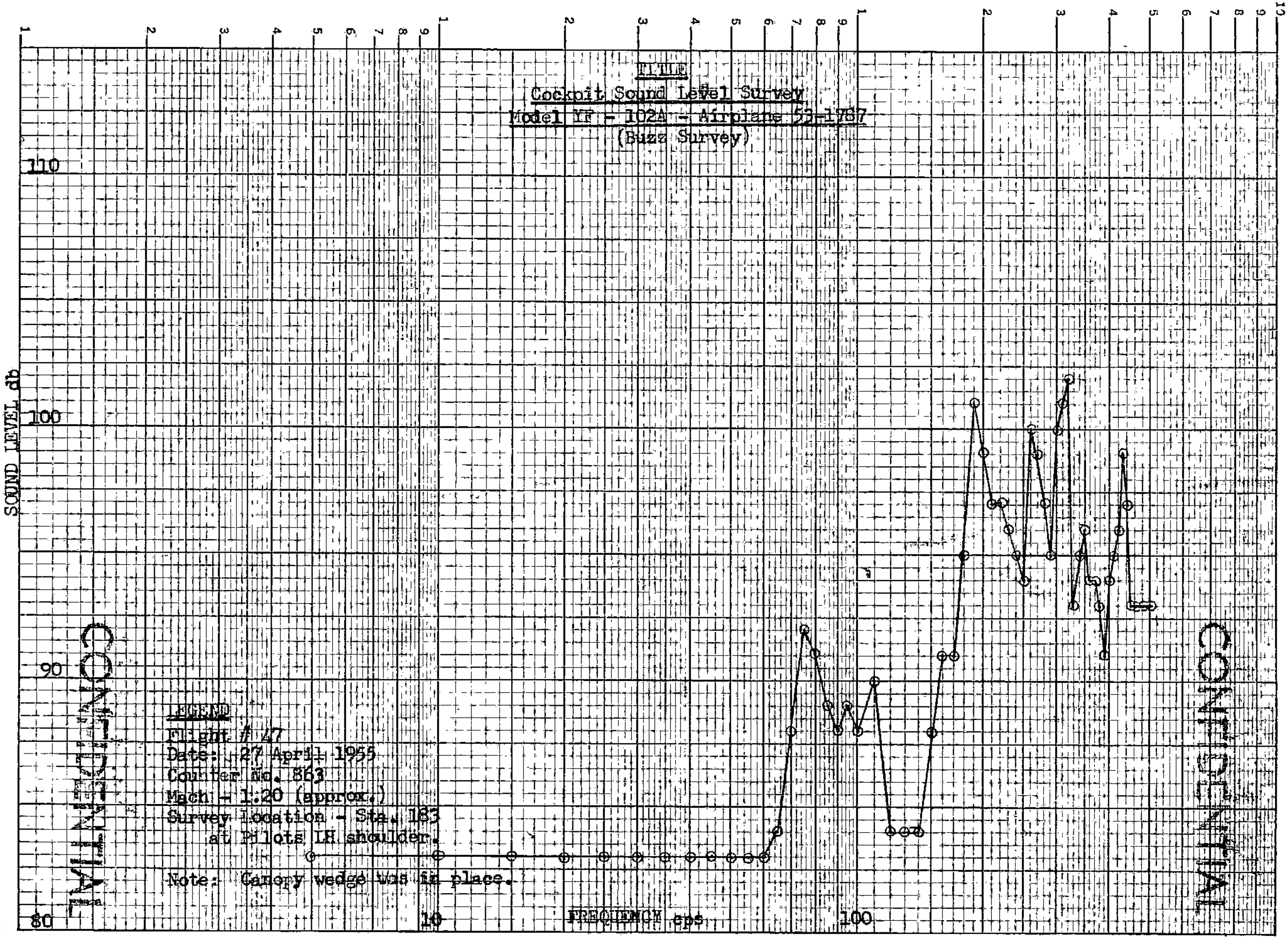
LEGEND
Flight # 47
Date: 27 April 1956
Counter No. 861
Mach - 1.25 (approx.)
Survey Location - Sta. 183
at Pilots LH shoulder.
Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 108

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Cockpit Sound Level Survey
Model YF - 102A - Airplane 53-1787
(Buzz Survey)

LEGEND

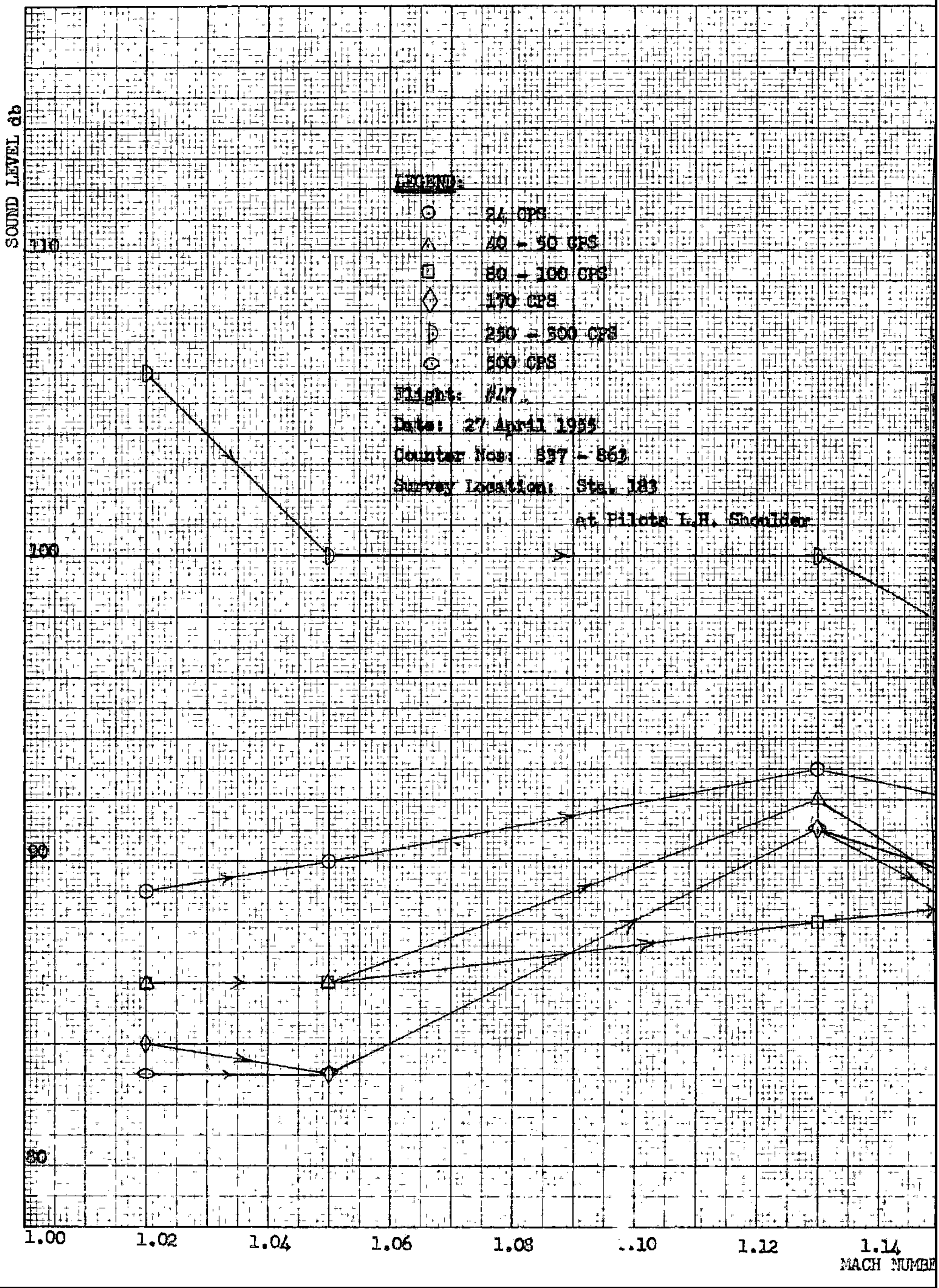
Flight # 47
Date: 27 April 1955
Counter No. 863
Mach - 1.20 (approx.)
Survey Location - Sta. 183
at Pilots LH shoulder.

Note: Canopy wedge was in place.

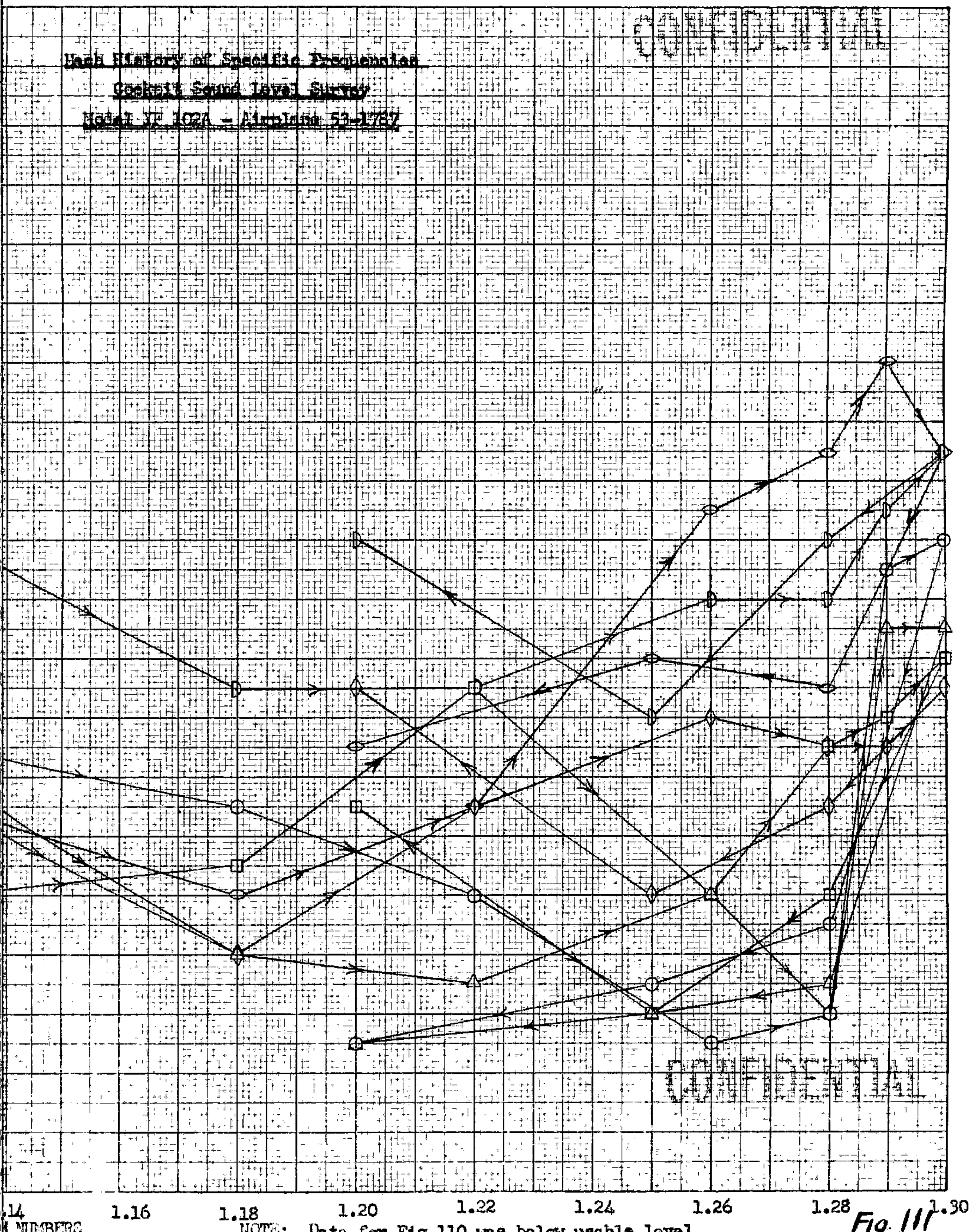
CONFIDENTIAL

NOTE: Data for Fig.110 was below usable level.

FIG. 109



K+E 10 X 10 TO THE 1/2 INCH 359T-11LG
KEUFFEL & ESSER CO. ALBANENE MADE IN U.S.A.

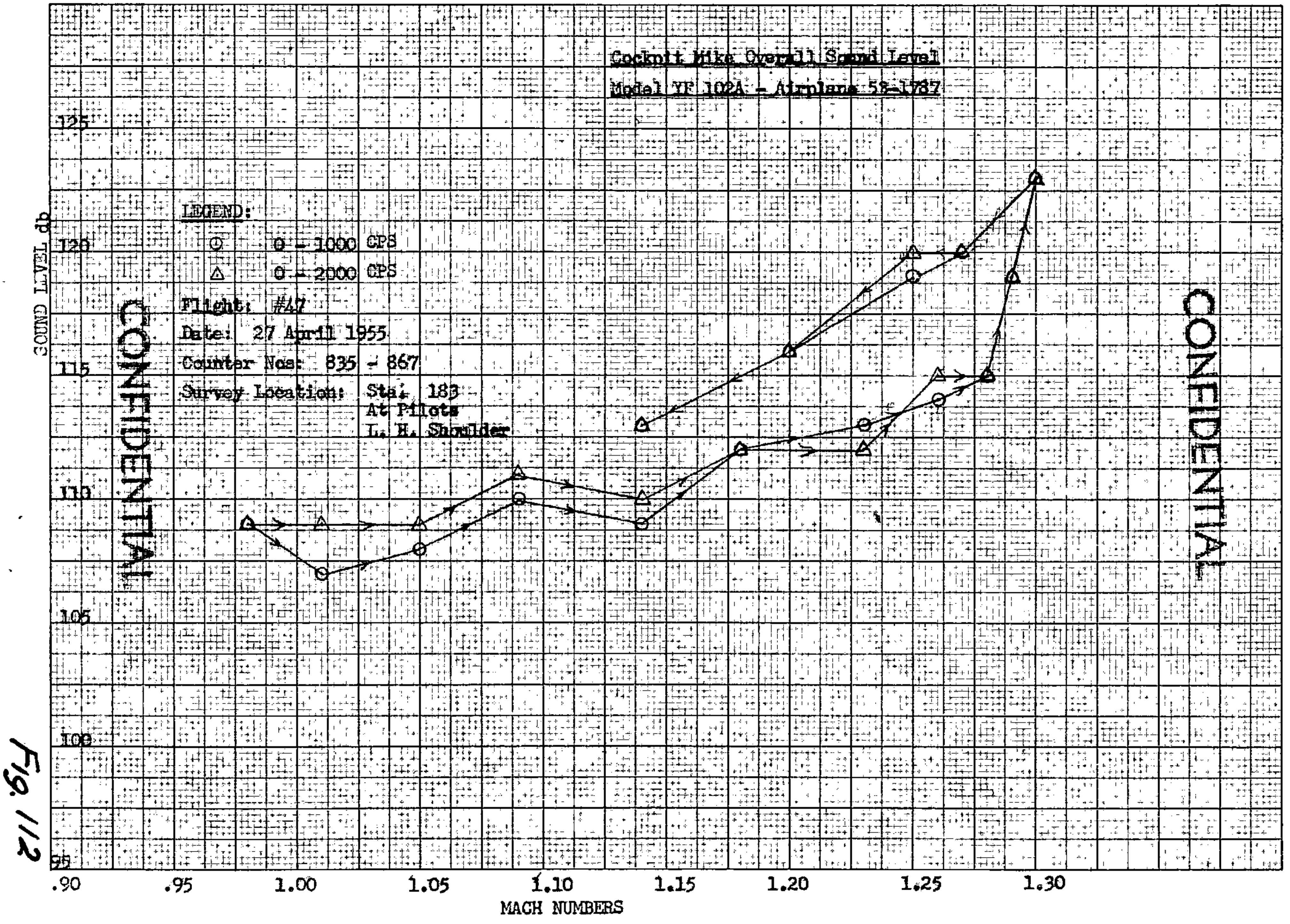

Mach History of Specific Frequencies
Cockpit Sound Level Survey
Model YF 102A - Airplane 53-1787

NOTE: Data for Fig.110 was below usable level.

Fig. 111

K&E 10 X 10 TO THE ½ INCH 359-11
KEUFFEL & ESSER CO. MADE IN U.S.A.

CONFIDENTIAL

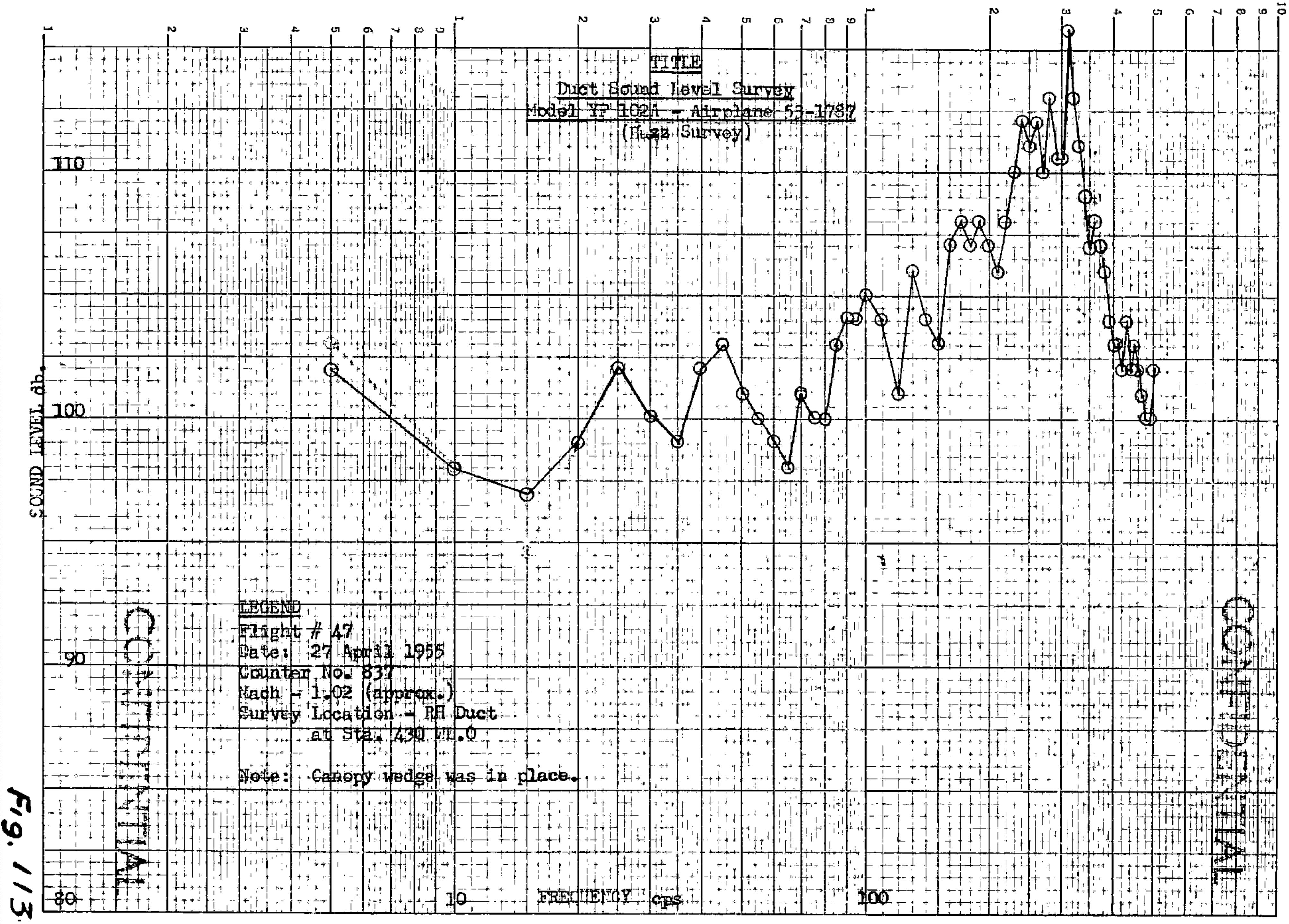


CONFIDENTIAL

Fig. 112

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

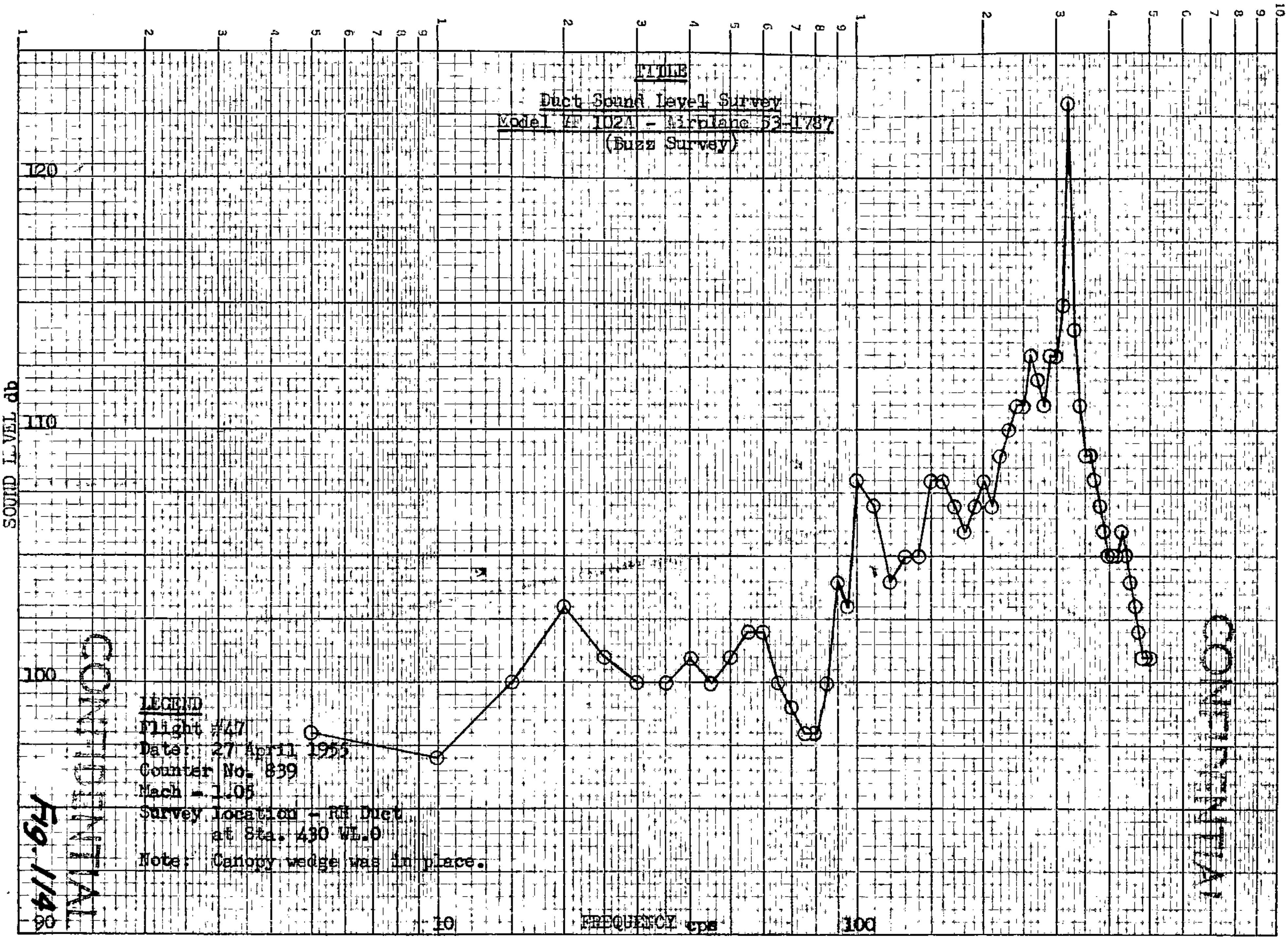
LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 837
Mach - 1.02 (approx.)
Survey Location - RH Duct
at Sta. 430 WL.0

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 113.

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



TITLE

Duct Sound Level Survey

Model F 102A - Airplane 53-1787

(Buzz Survey)

LEGEND

Flight #47

Date: 27 April 1955

Counter No. 839

Mach - 1.05

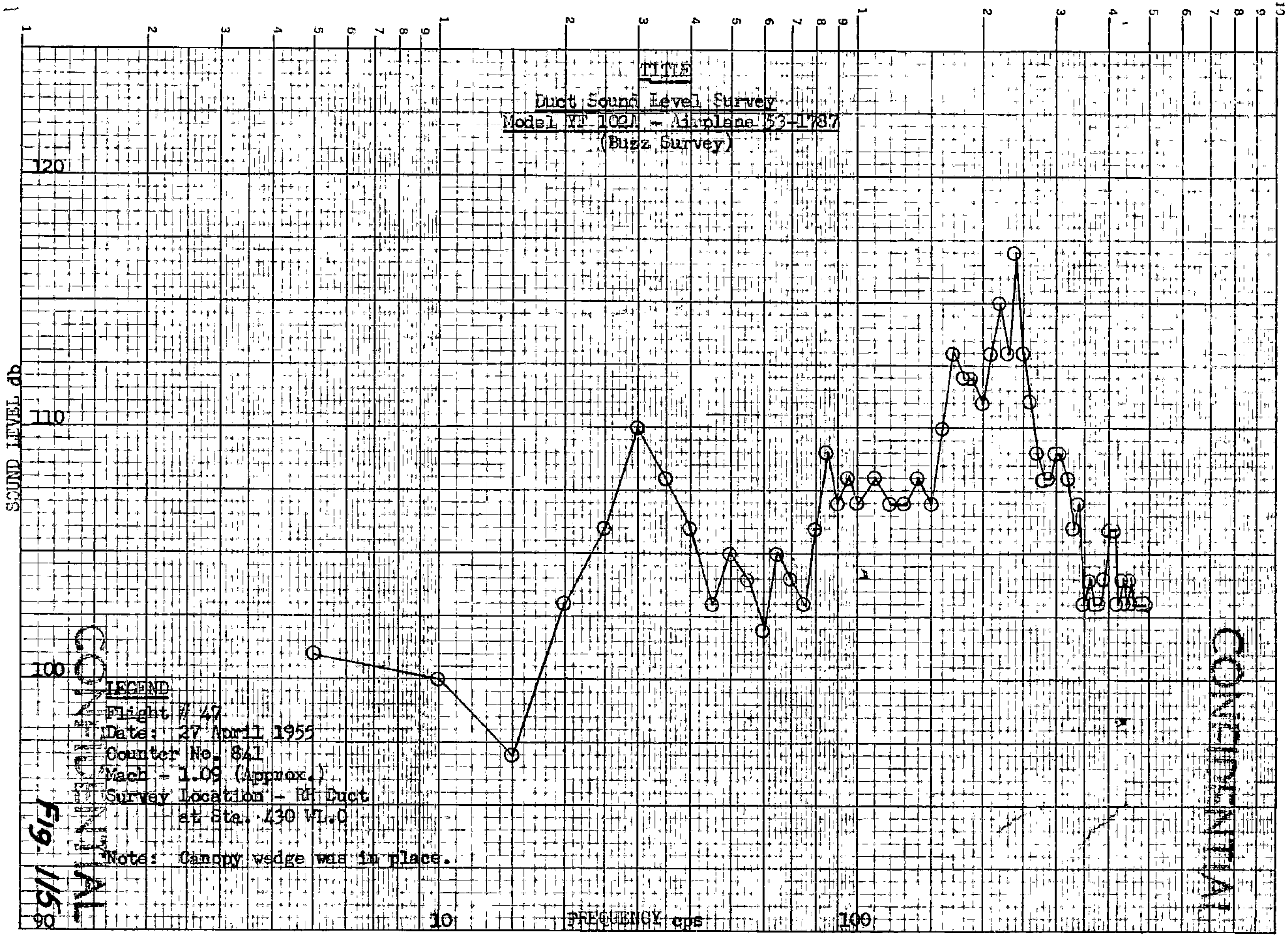
Survey Location - RH Duct
at Sta. 430 WL.0

Note: Canopy wedge was in place.

Fig. 114

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Duct Sound Level Survey
Model YF-102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight # 47
Date: 27 April 1955
Counter No. 841
Mach - 1.09 (Approx.)
Survey Location - Rt Duct
at Sta. 430 WL.0

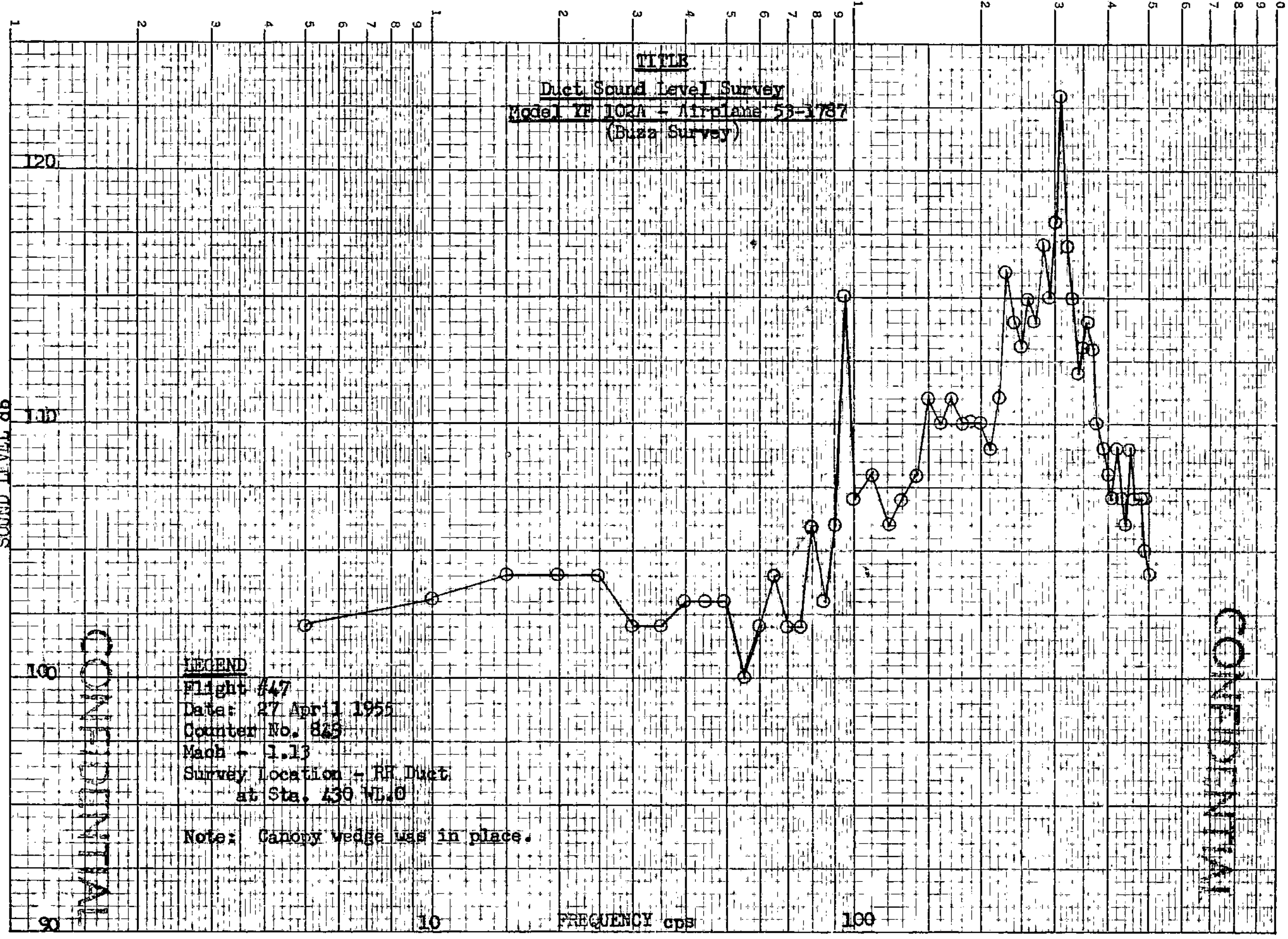
Note: Canopy wedge was in place.

Fig. 115

CO AL

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buza Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 823
Mach - 1.13
Survey Location - RH Duct
at Sta. 430 WL.0

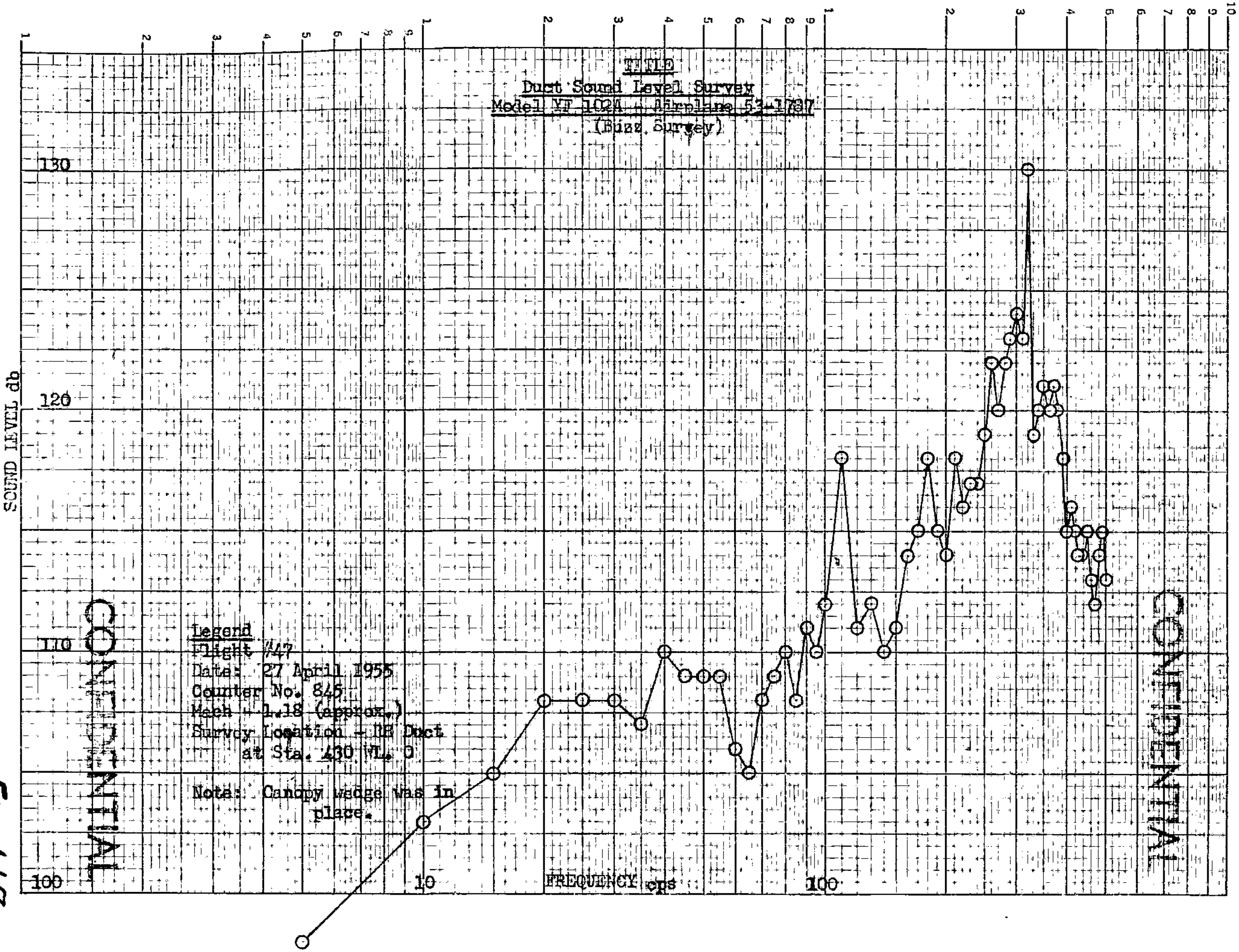
Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 116

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

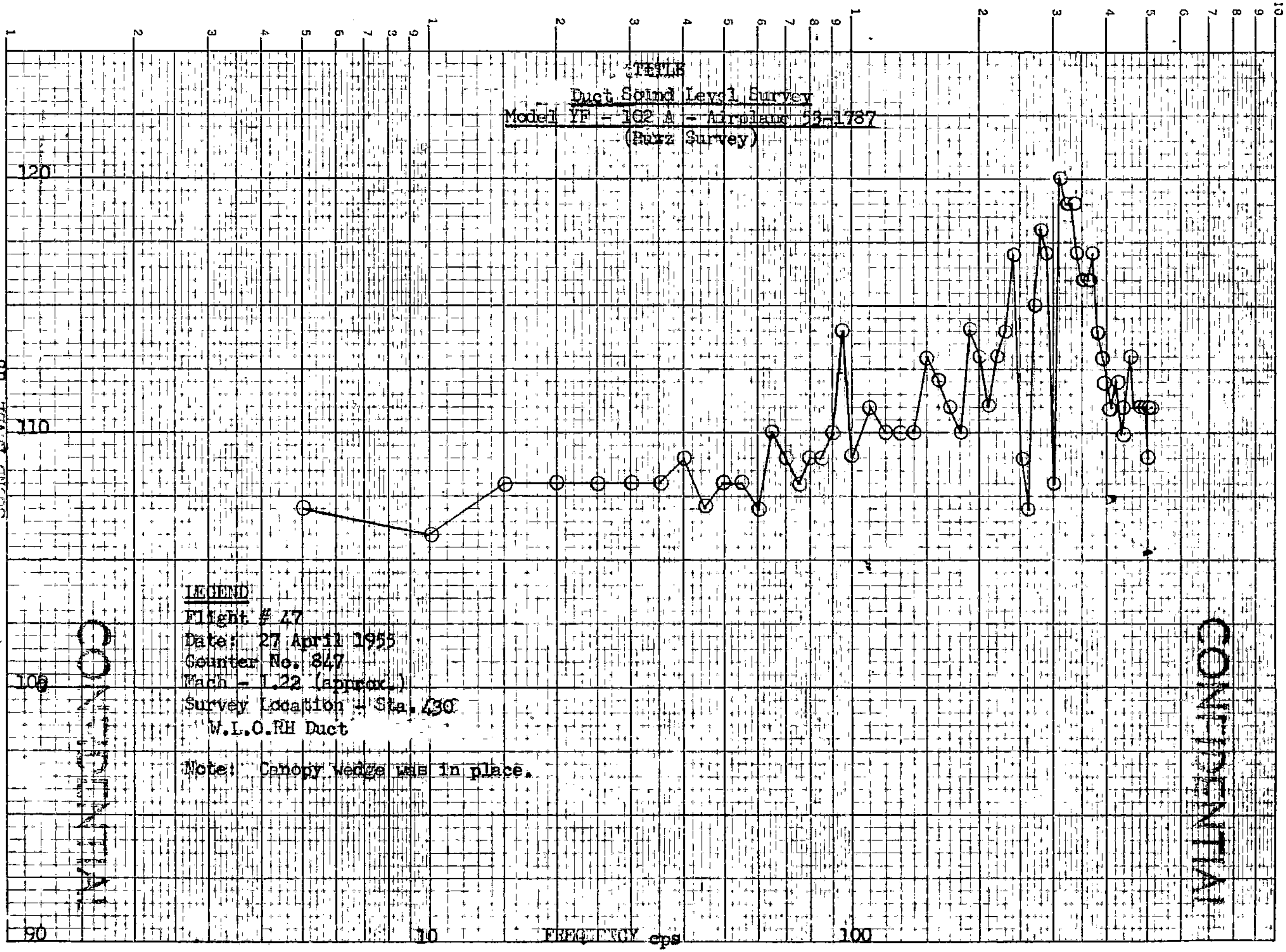
Legend
Flight #47
Date: 27 April 1955
Counter No. 845
Mach - 1.18 (approx.)
Survey Location - RH Duct
at Sta. 430 WL. 0

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 117

CONFIDENTIAL



TITLE

Duct Sound Level Survey

Model YF - 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND

Flight # 47

Date: 27 April 1955

Counter No. 847

Mach - 1.22 (approx.)

Survey Location - Sta. 430

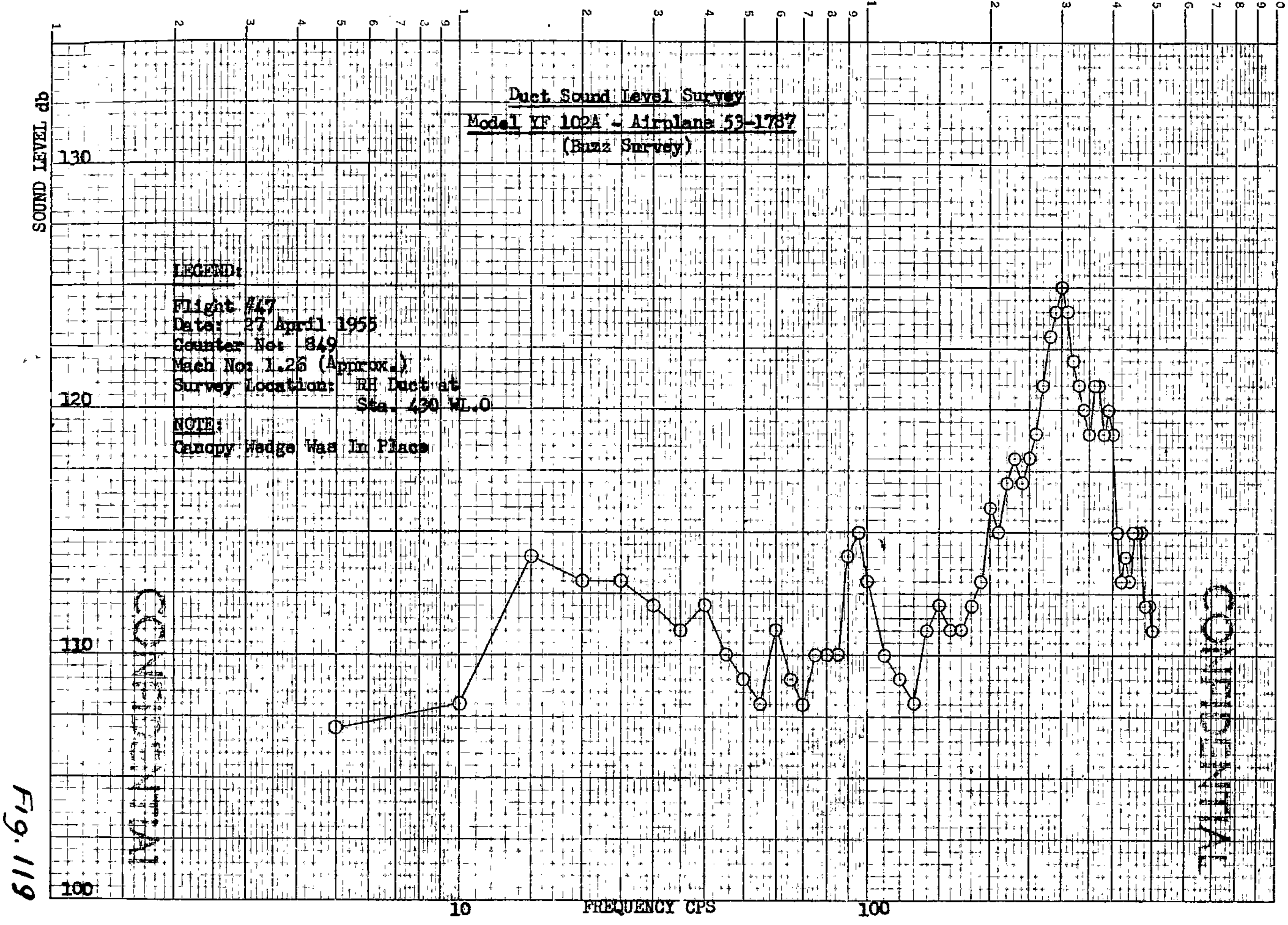
W.L.O. RH Duct

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 118

CONFIDENTIAL



Duct Sound Level Survey
Model XF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:

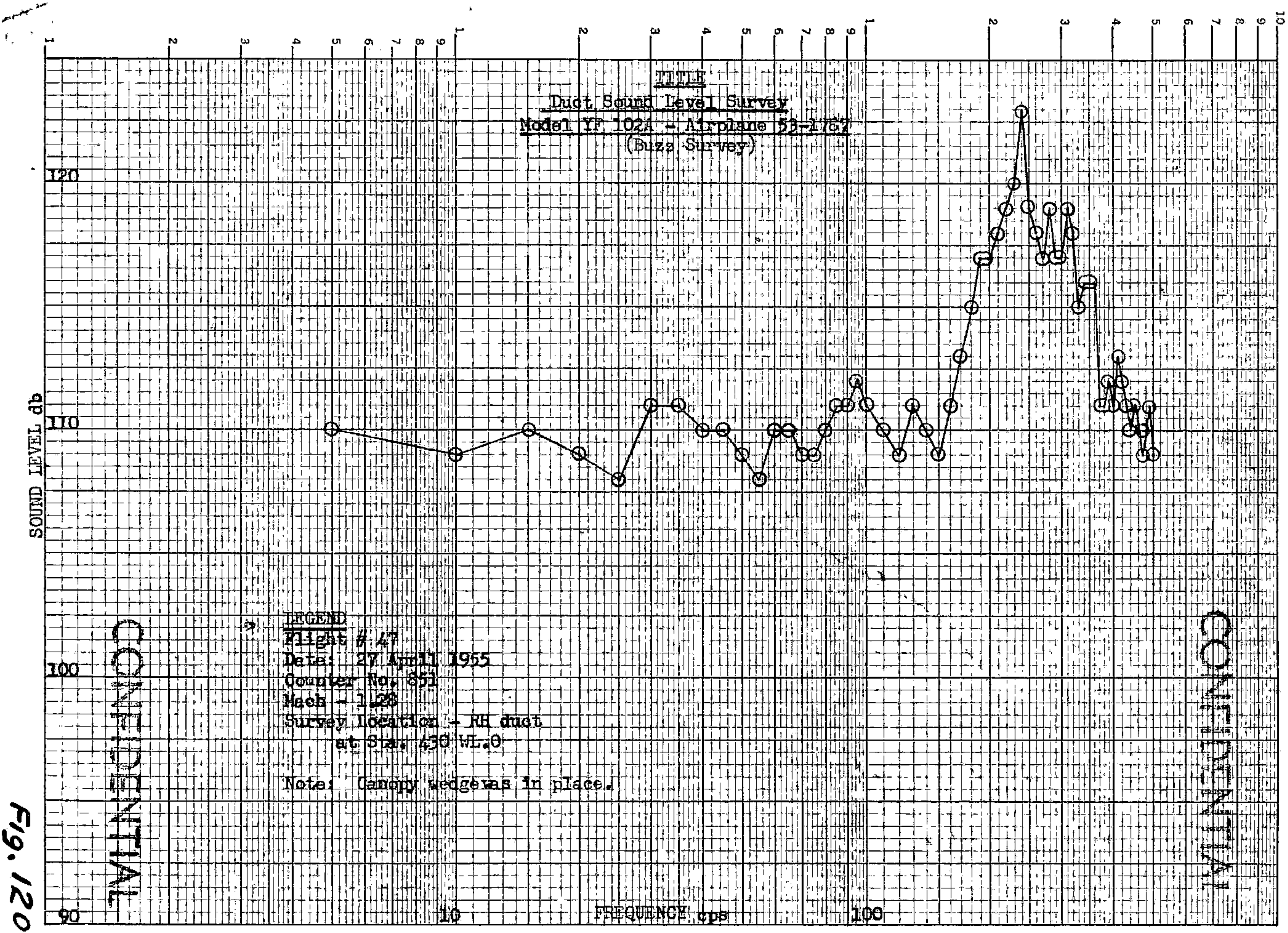
Flight #47
Date: 27 April 1955
Counter No: 849
Mach No: 1.26 (Approx.)
Survey Location: RH Duct at
Sta. 430 WL.0

NOTE:
Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 119

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



TITLE

Duct Sound Level Survey

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND

Flight # 47

Date: 27 April 1955

Counter No. 851

Mach - 1.28

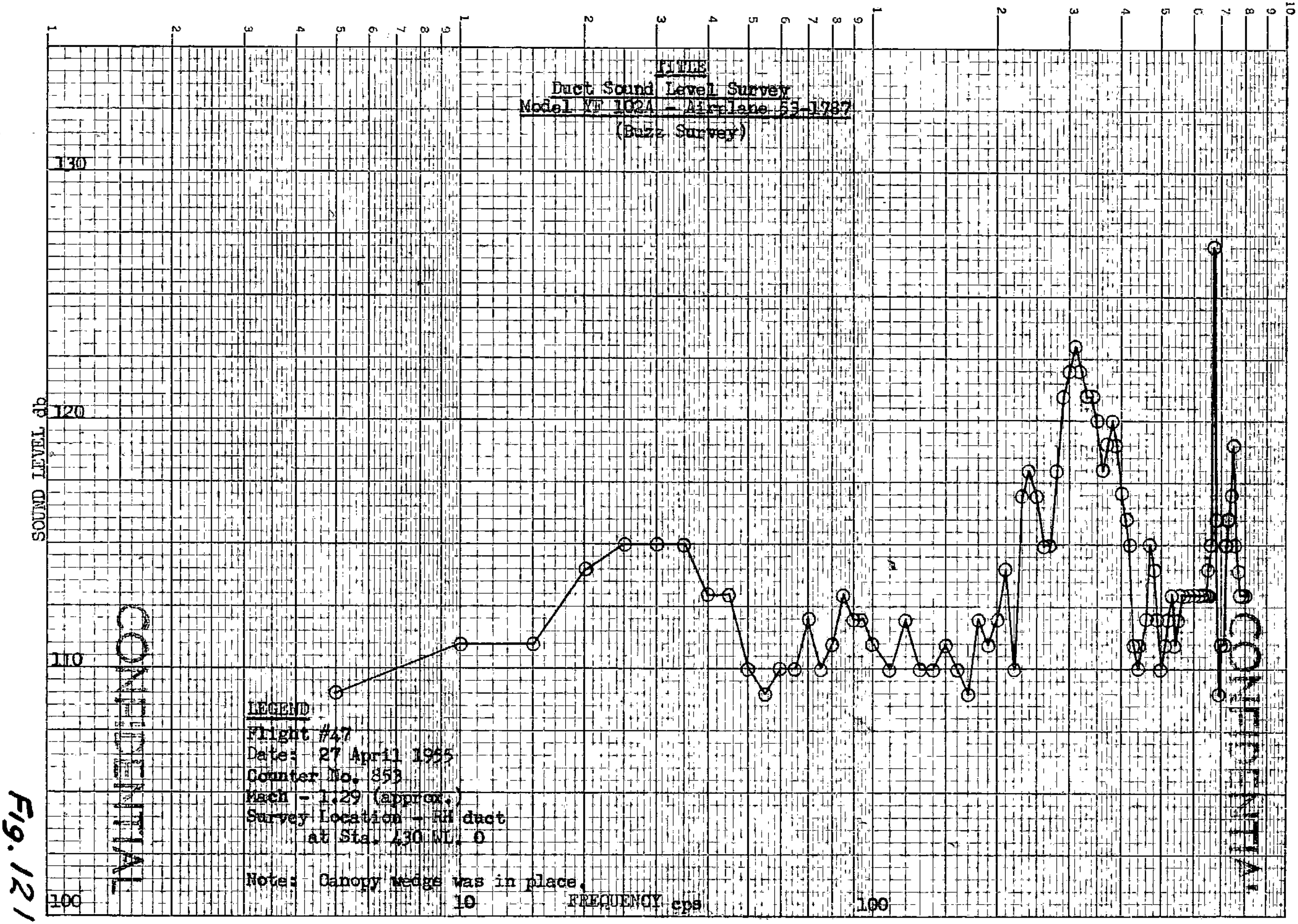
Survey location - RH duct
at Sta. 430 WL.0

Note: Canopy wedgewas in place.

Fig. 120

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 853
Mach - 1.29 (approx.)
Survey Location - RH duct
at Sta. 430 WL. 0

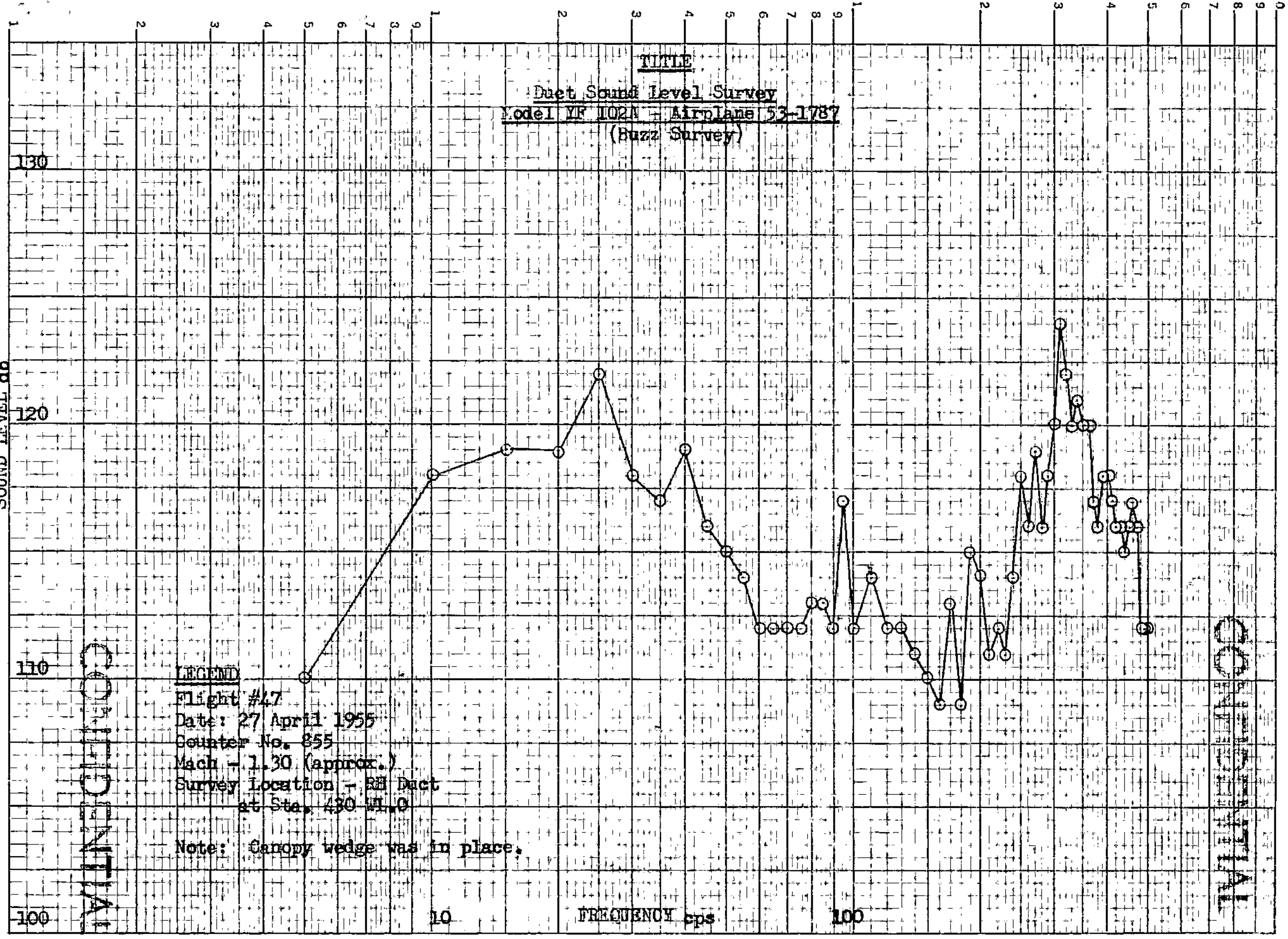
Note: Canopy wedge was in place.

Fig. 121

CONFIDENTIAL

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 855
Mach - 1.30 (approx.)
Survey Location - RH Duct
at Sta. 430 WL 0

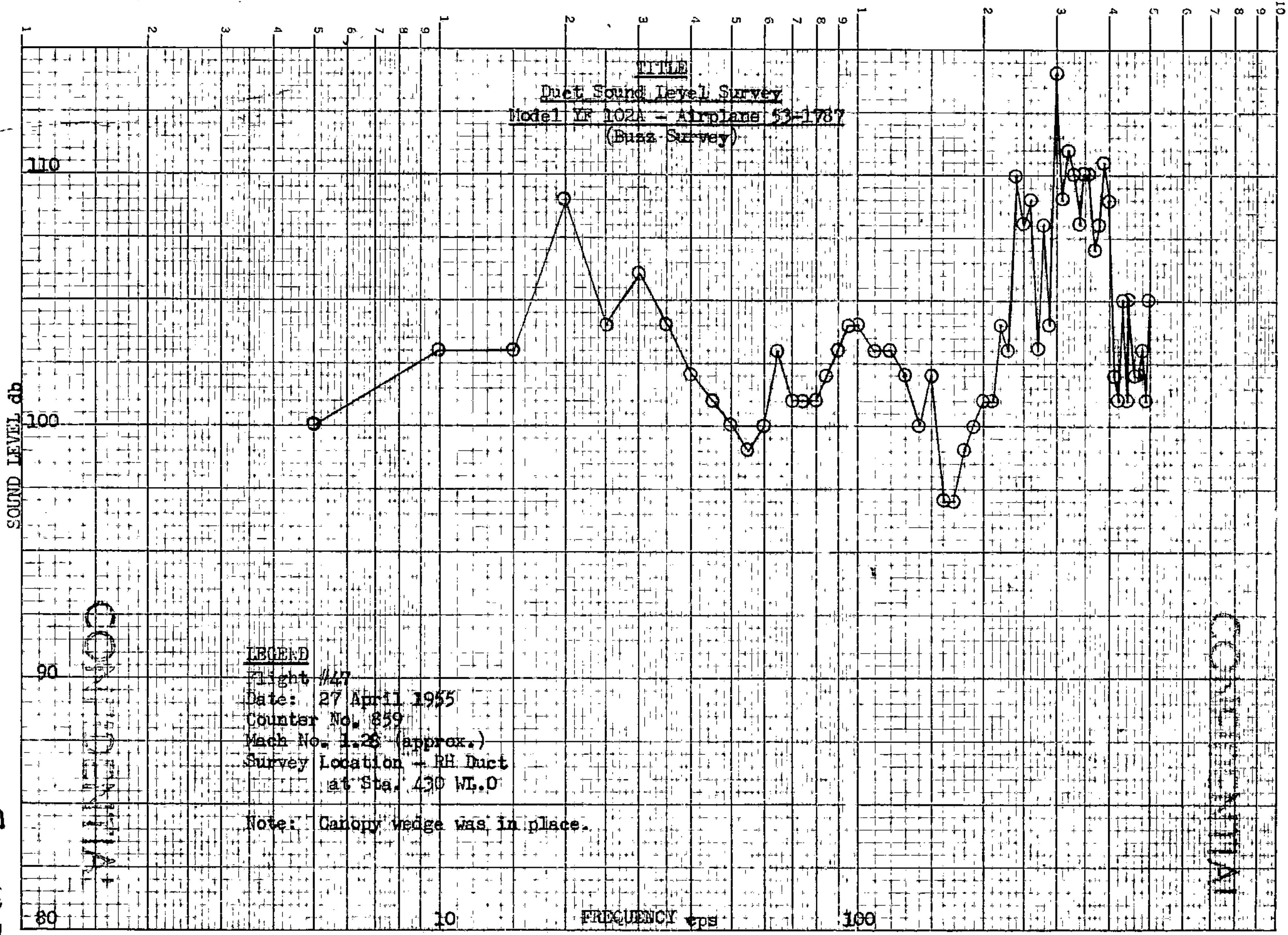
Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 122

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

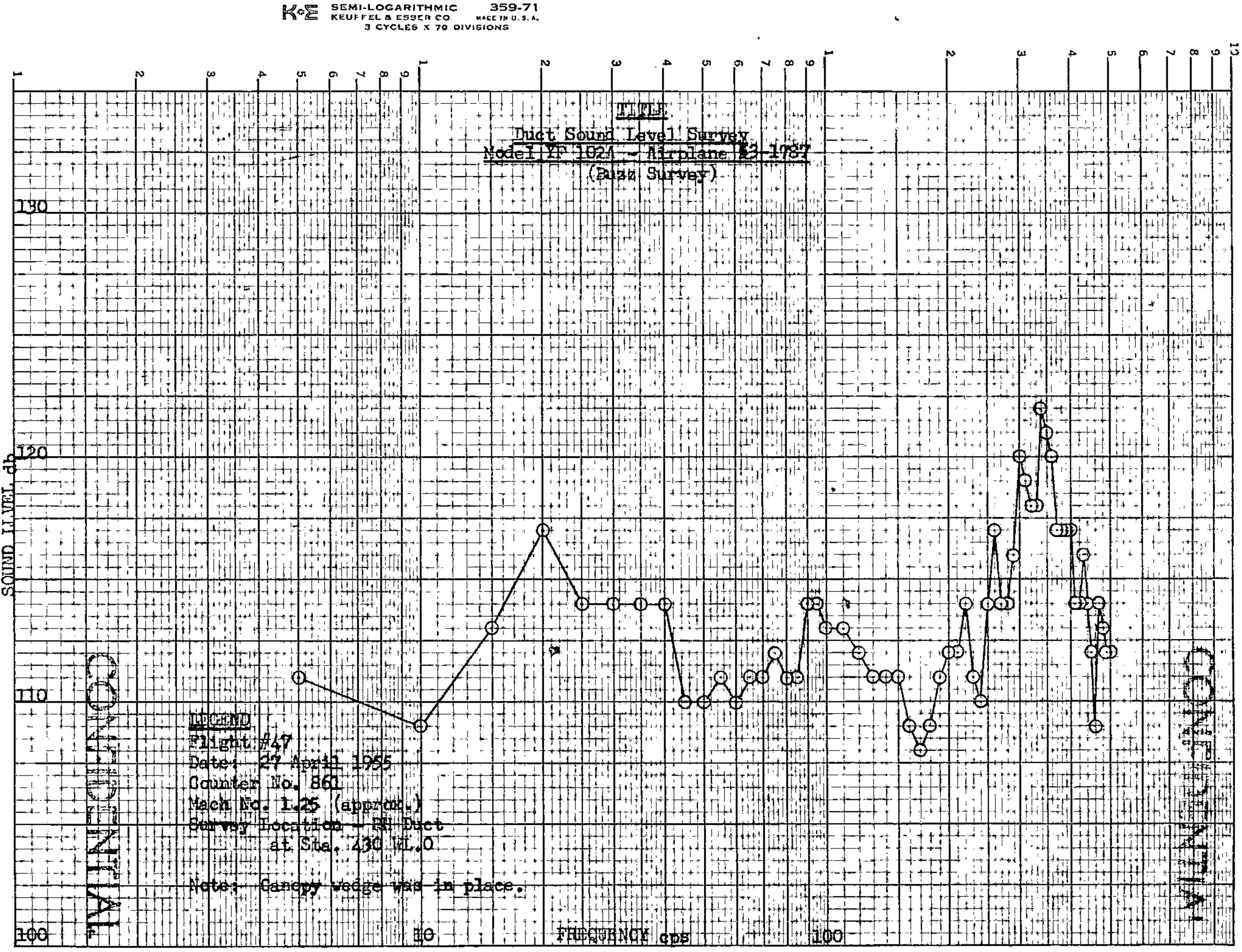
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 859
Mach No. 1.28 (approx.)
Survey Location - RH Duct
at Sta. 430 WL.0

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 123

CONFIDENTIAL



TITLE
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

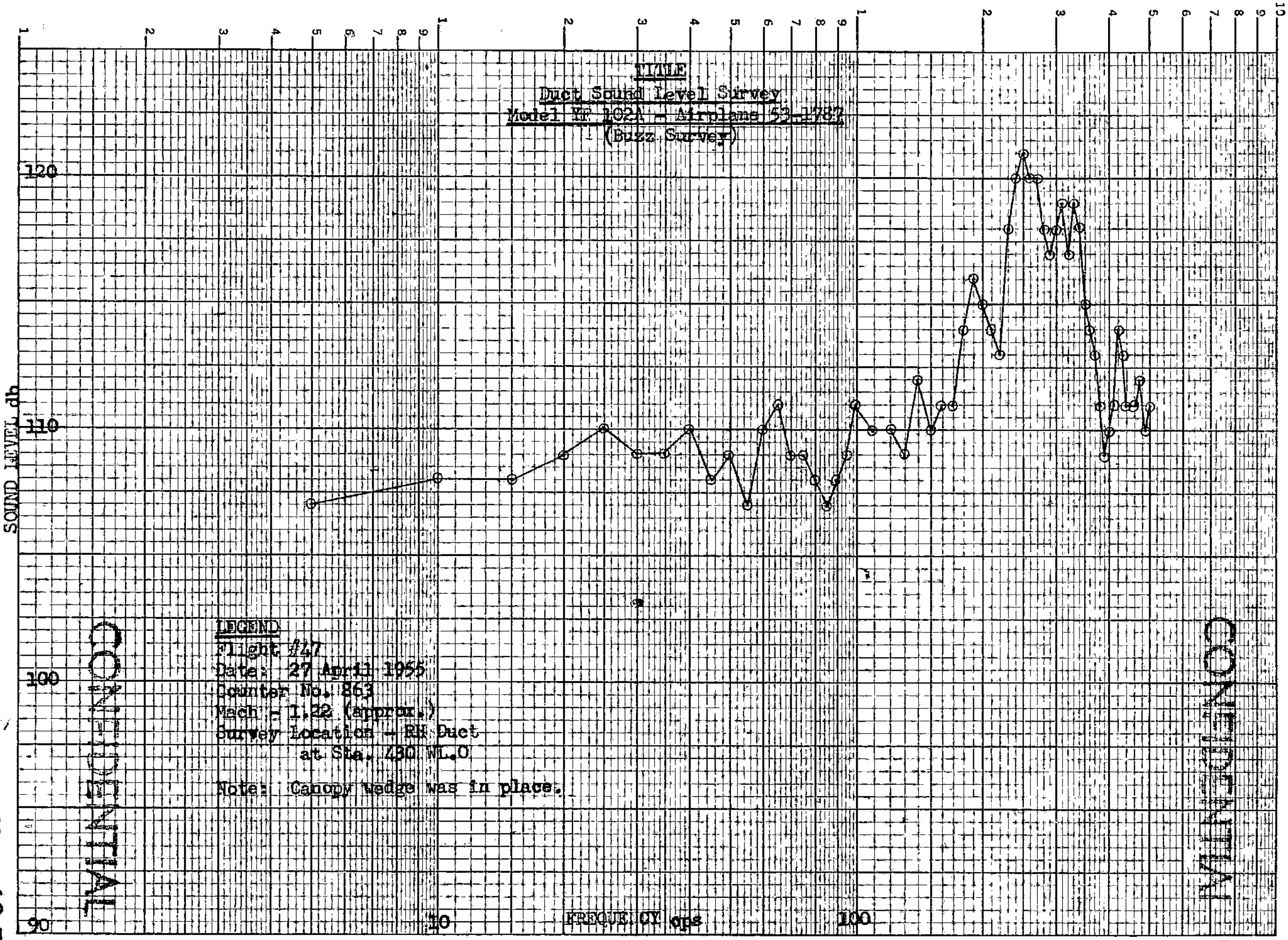
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 861
Mach No. 1.25 (approx.)
Survey Location - RH Duct
at Sta. 430 WL.0

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 124

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



TITLE

Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 863
Mach - 1.22 (approx.)
Survey Location - RH Duct
at Sta. 430 WL.0

Note: Canopy wedge was in place.

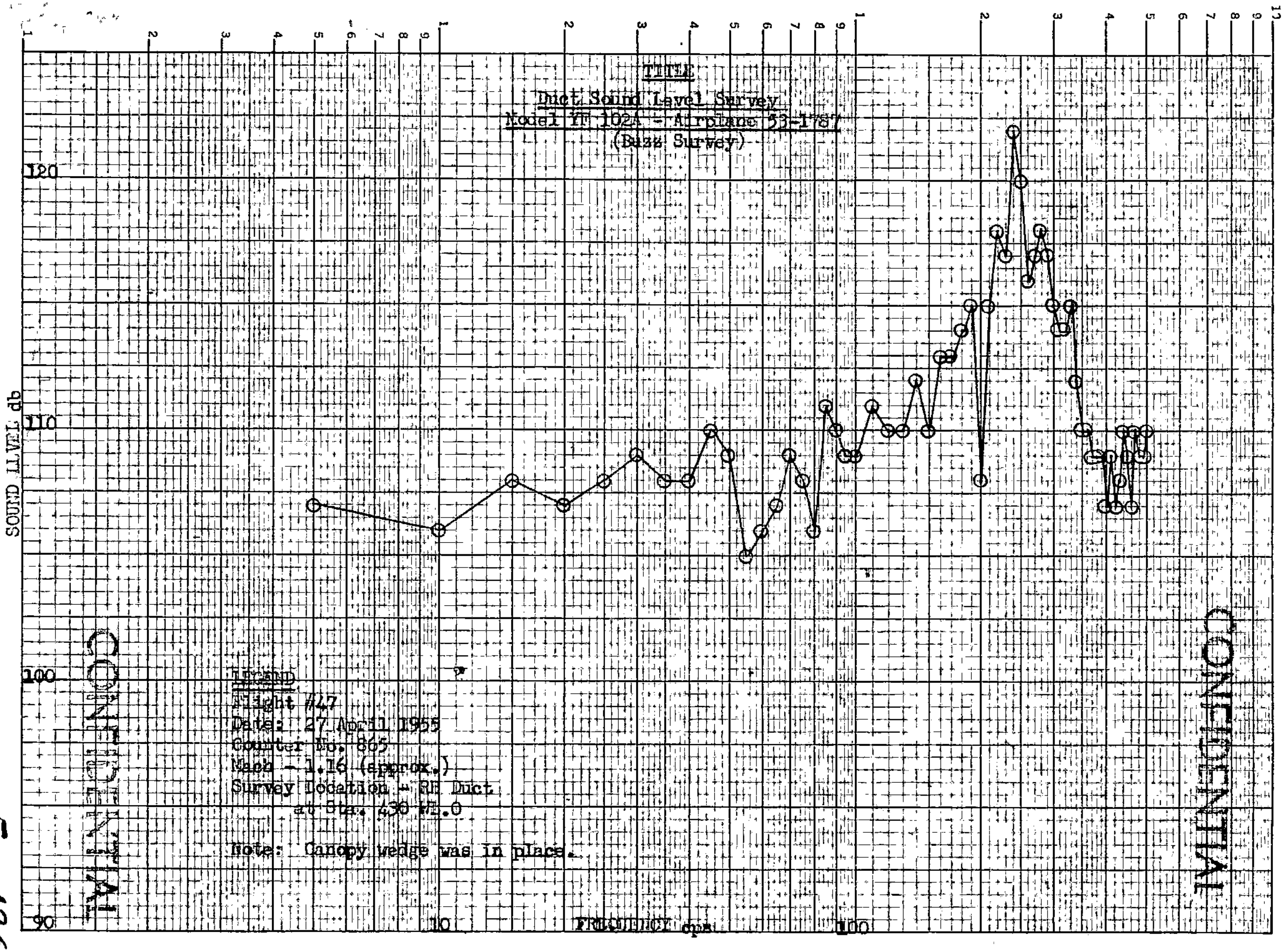
CONFIDENTIAL

CONFIDENTIAL

Fig. 125

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

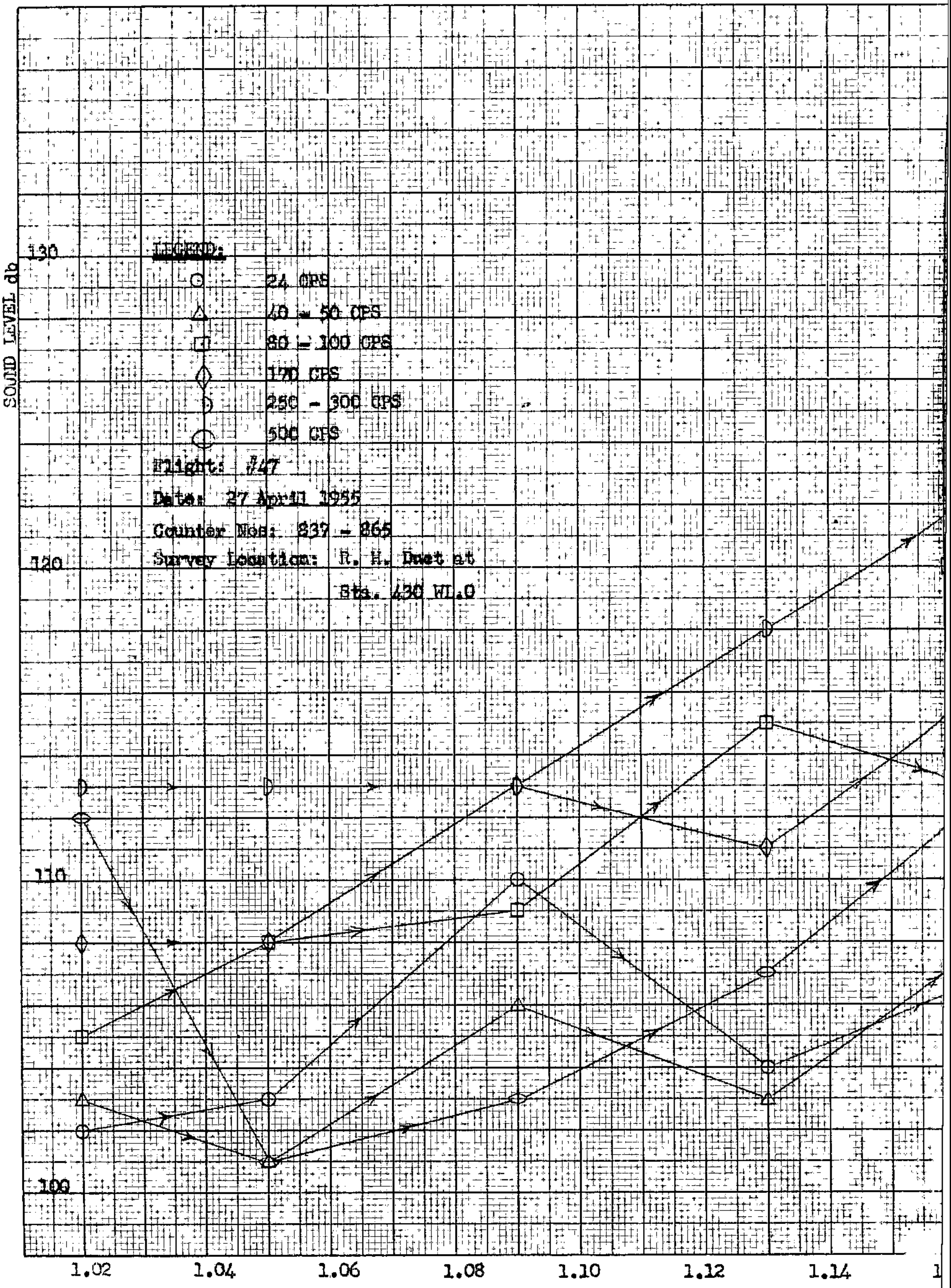
Duct Sound Level Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

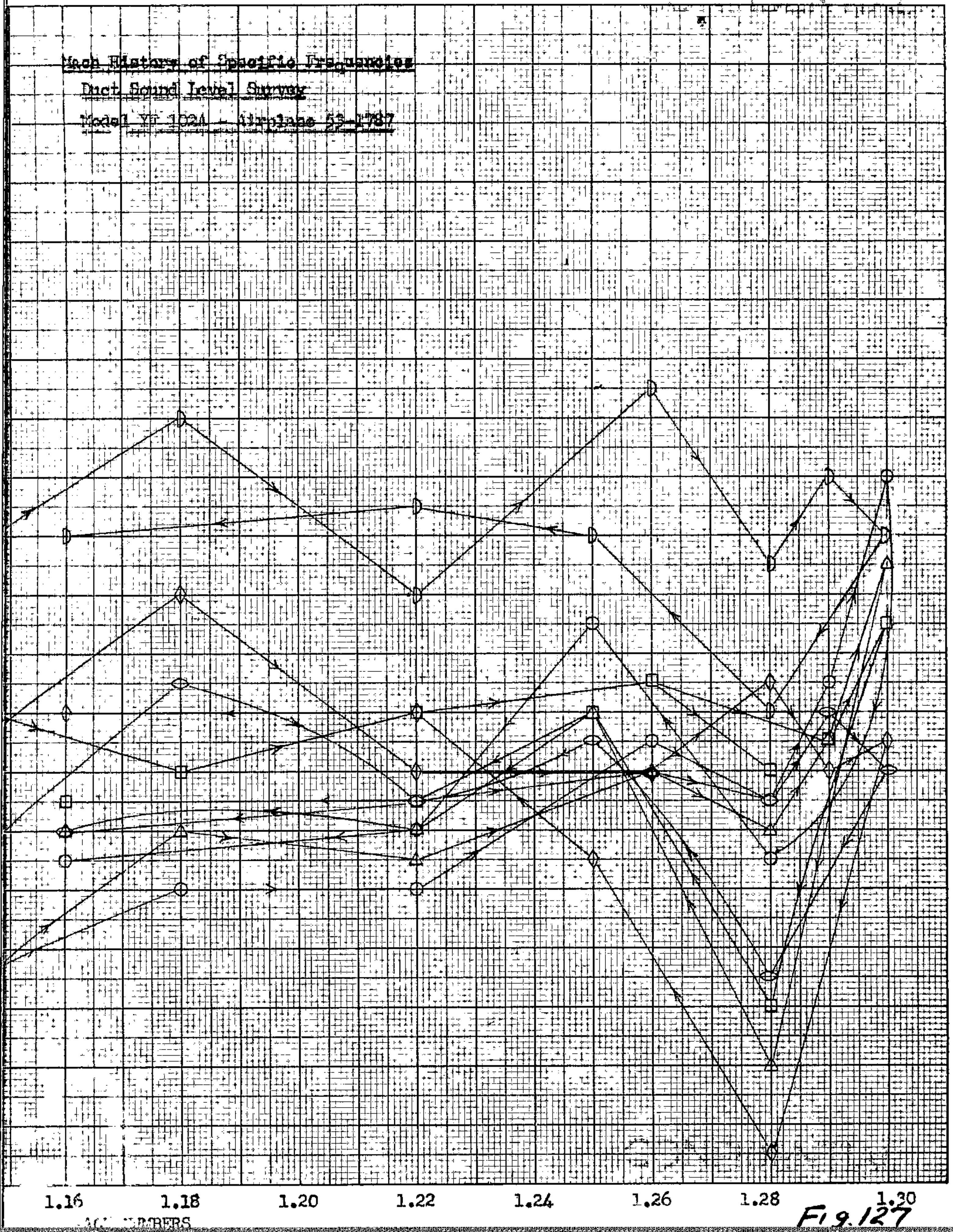
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 865
Mach - 1.16 (approx.)
Survey Location - RH Duct
at Sta. 430 WL.0

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 126


LEGEND:
○ 24 CPS
△ 40 - 50 CPS
□ 80 - 100 CPS
◇ 170 CPS
D 250 - 300 CPS
⬭ 500 CPS
Flight: #47
Date: 27 April 1955
Counter Nos: 837 - 865
Survey Location: R. H. Duct at
Sta. 430 WL.0
SOUND LEVEL db
130
120
110
100
1.02
1.04
1.06
1.08
1.10
1.12
1.14
1



Mach History of Specific Frequencies

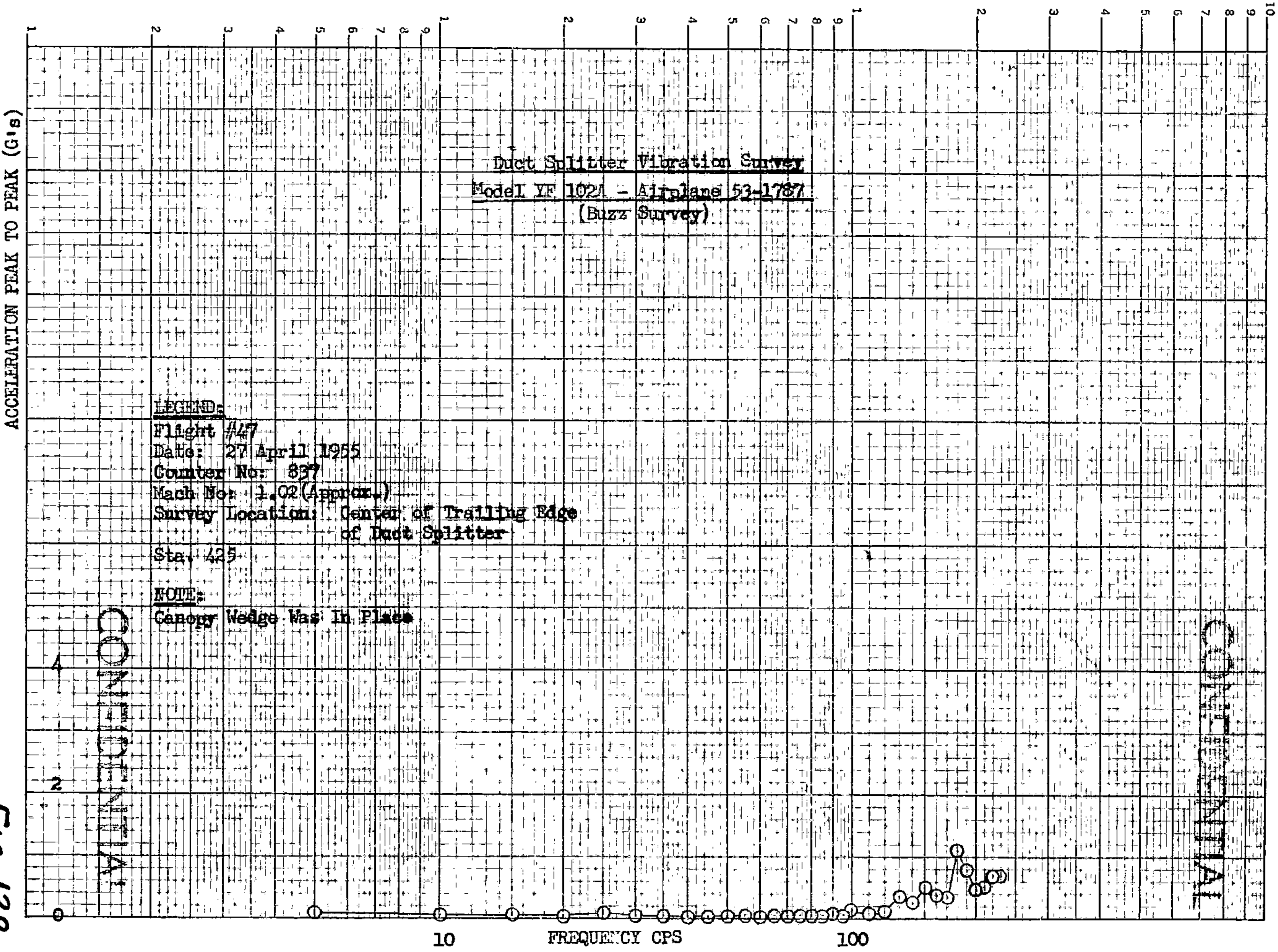
Duct Sound Level Survey

Model YF-102A - Airplane 53-1787

Fig. 127

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



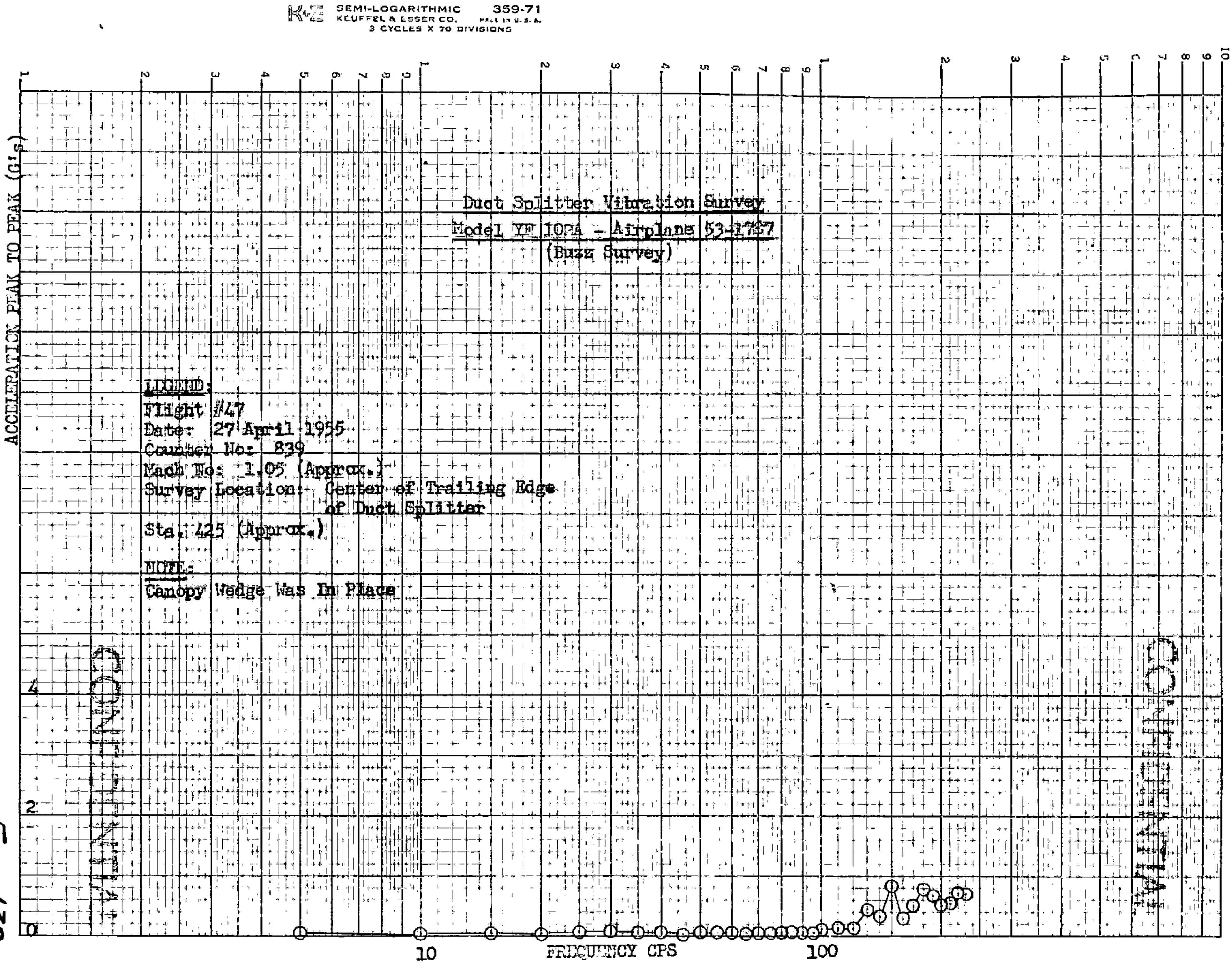
Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 837
Mach No: 1.02 (Approx.)
Survey Location: Center of Trailing Edge of Duct Splitter
Sta. 425

NOTE:
Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 129



Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

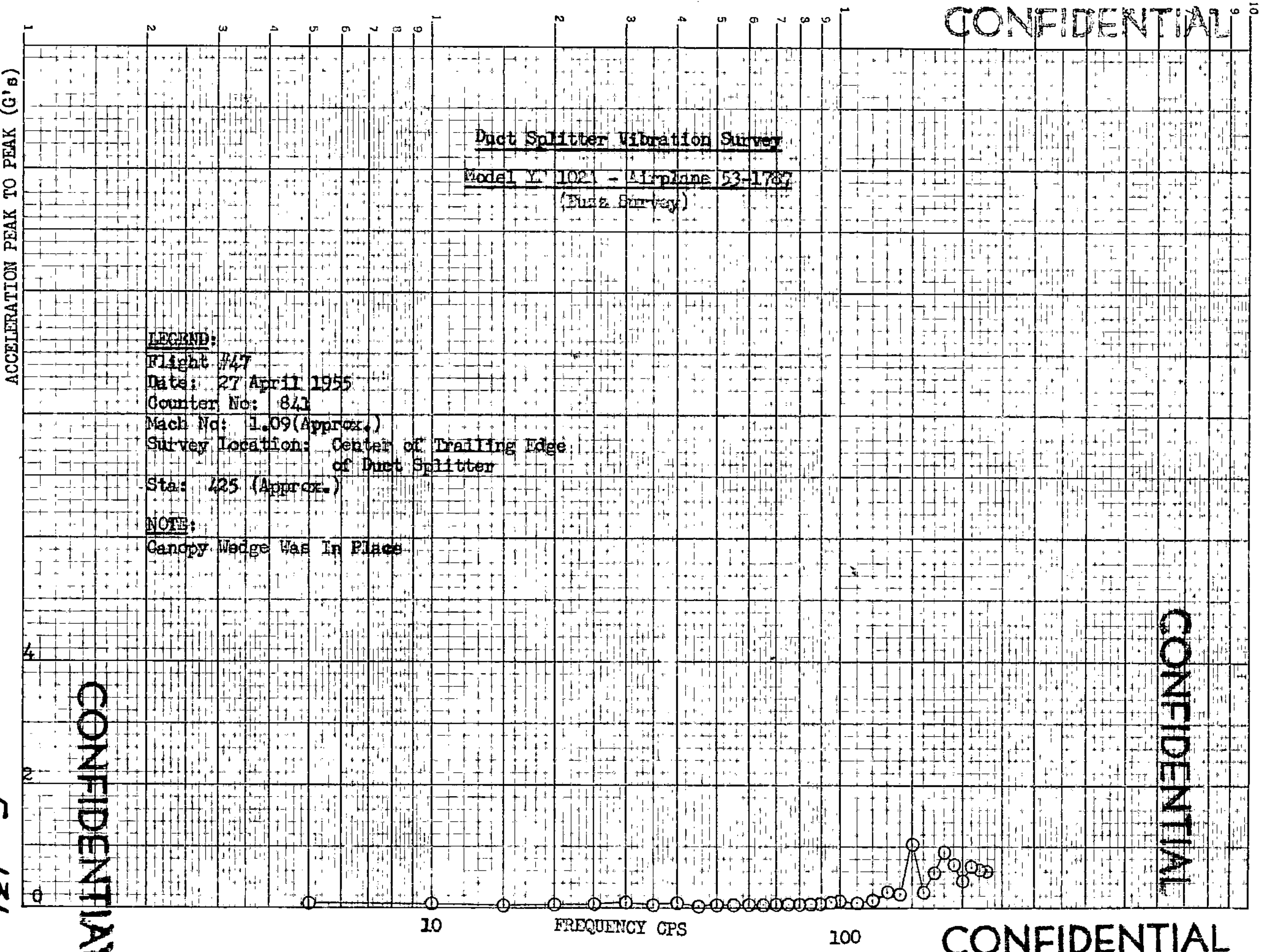
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 839
Mach No: 1.05 (Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta. 425 (Approx.)

NOTE:
Canopy Wedge Was In Place

Fig. 130

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Splitter Vibration Survey

Model Y. 1021 - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 841
Mach No: 1.09(Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta: 425 (Approx.)

NOTE:
Canopy Wedge Was In Place

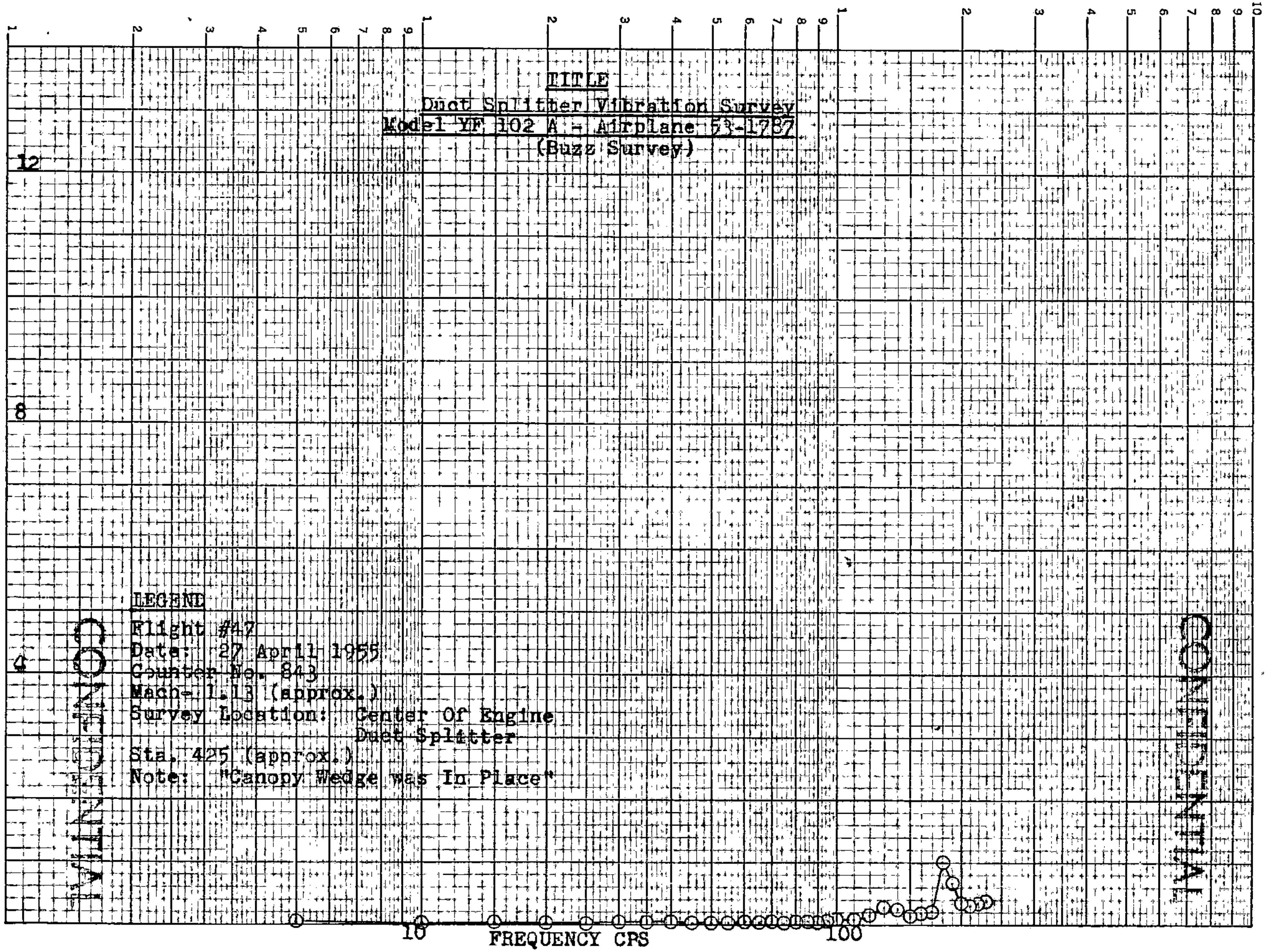
CONFIDENTIAL

CONFIDENTIAL

CONFIDENTIAL

Fig. 131

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



TITLE

Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 843
Mach- 1.13 (approx.)
Survey Location: Center Of Engine
Duct Splitter
Sta. 425 (approx.)
Note: "Canopy Wedge was In Place"

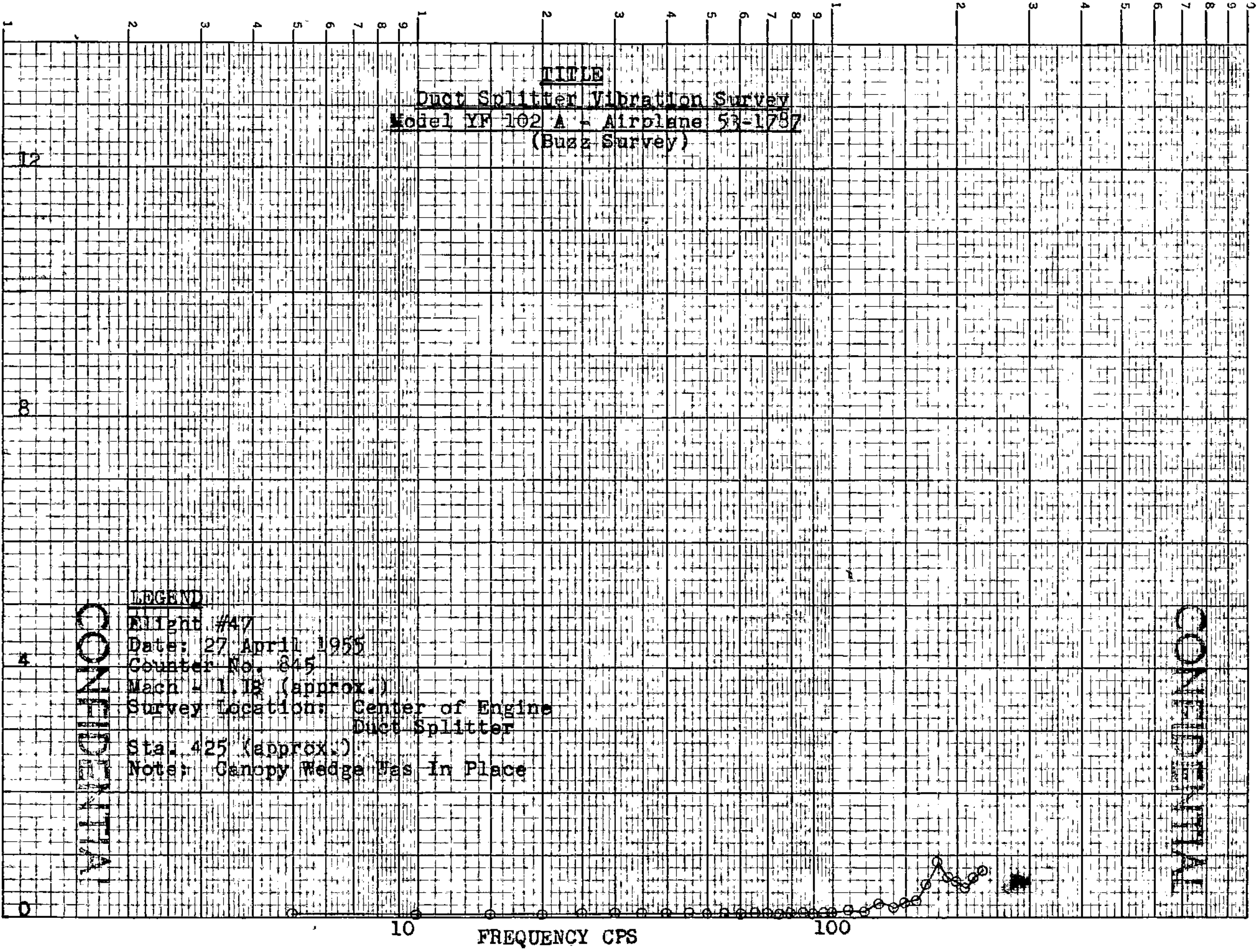
CONFIDENTIAL

CONFIDENTIAL

Fig. 132

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

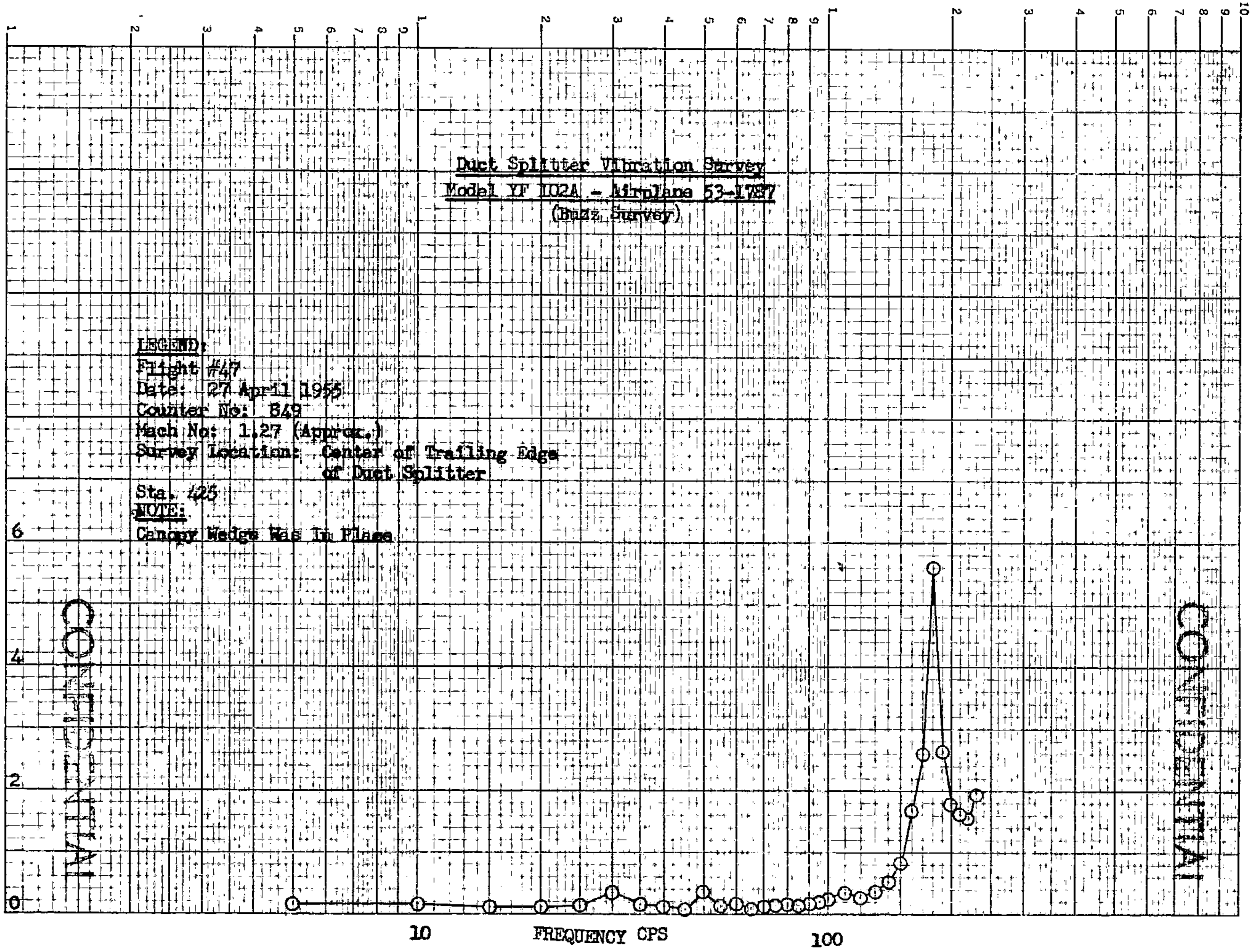
LEGEND

Flight #47
Date: 27 April 1955
Counter No. 845
Mach - 1.18 (approx.)
Survey Location: Center of Engine
Duct Splitter
Sta. 425 (approx.)
Note: Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 133

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



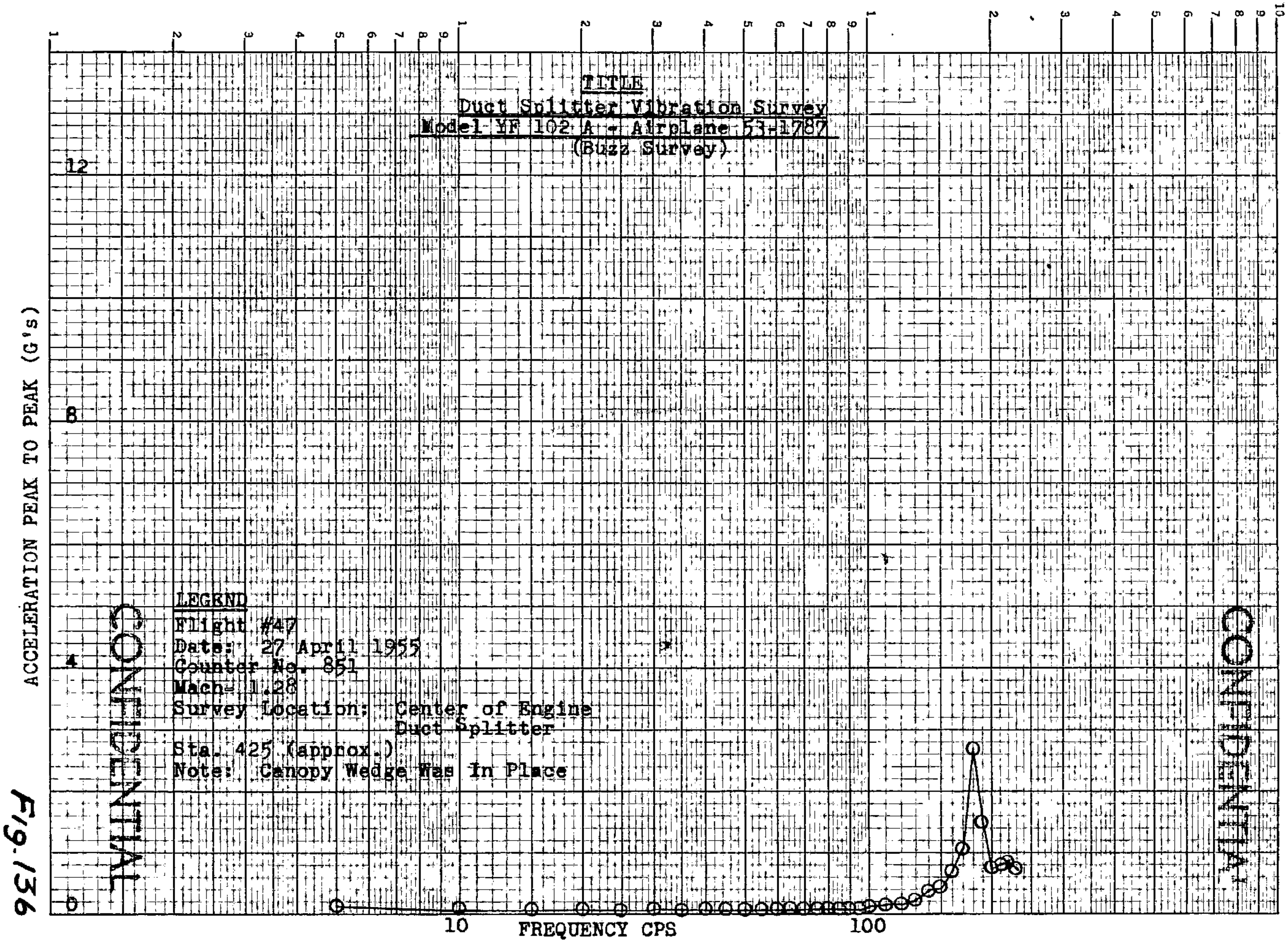
Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 849
Mach No: 1.27 (Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta. 425
NOTE:
Canopy Wedge Was In Place

Fig. 135

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE

Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

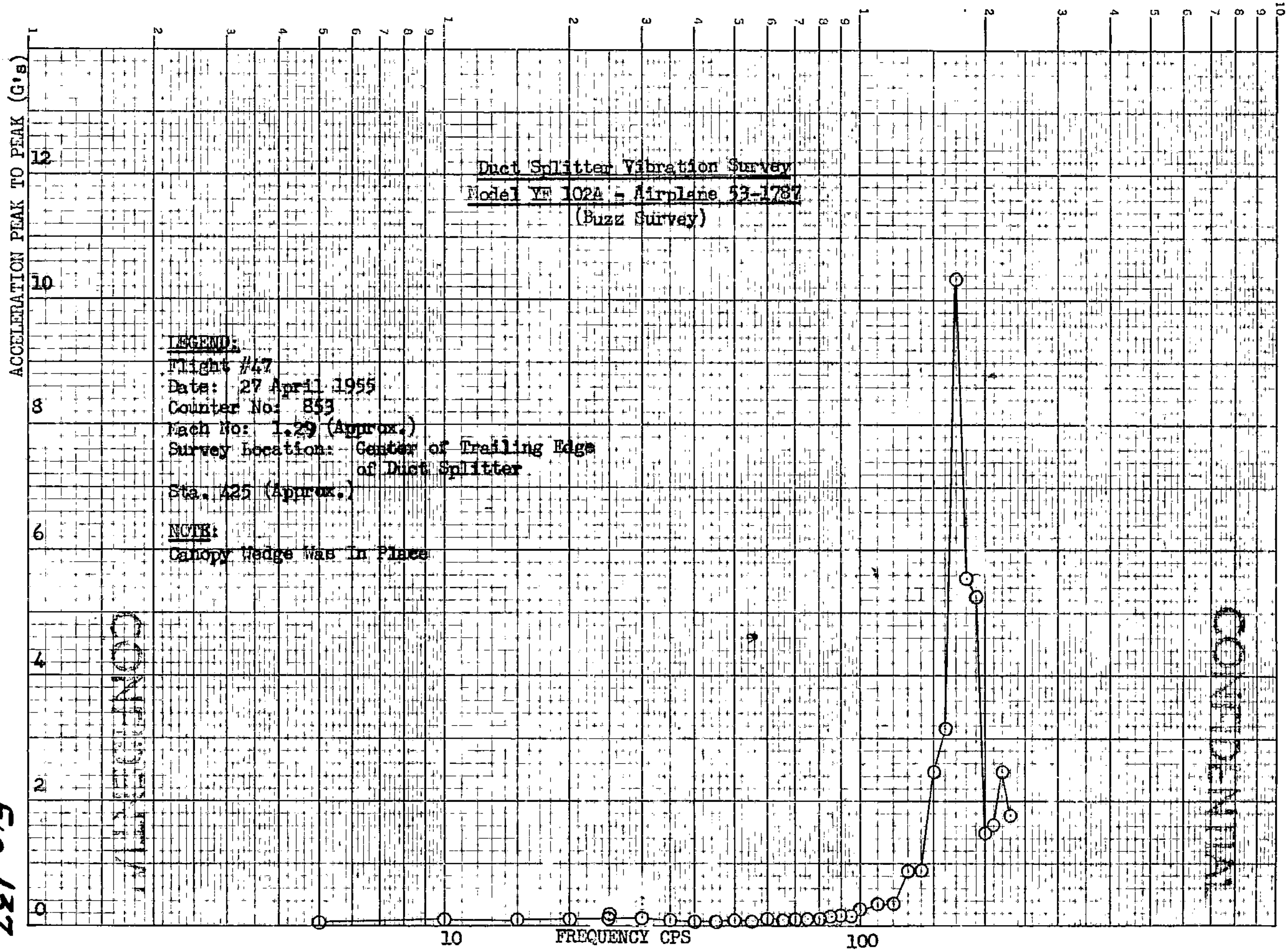
LEGEND

Flight #47
Date: 27 April 1955
Counter No. 851
Mach= 1.28
Survey Location: Center of Engine Duct Splitter
Sta. 425 (approx.)
Note: Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 136

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



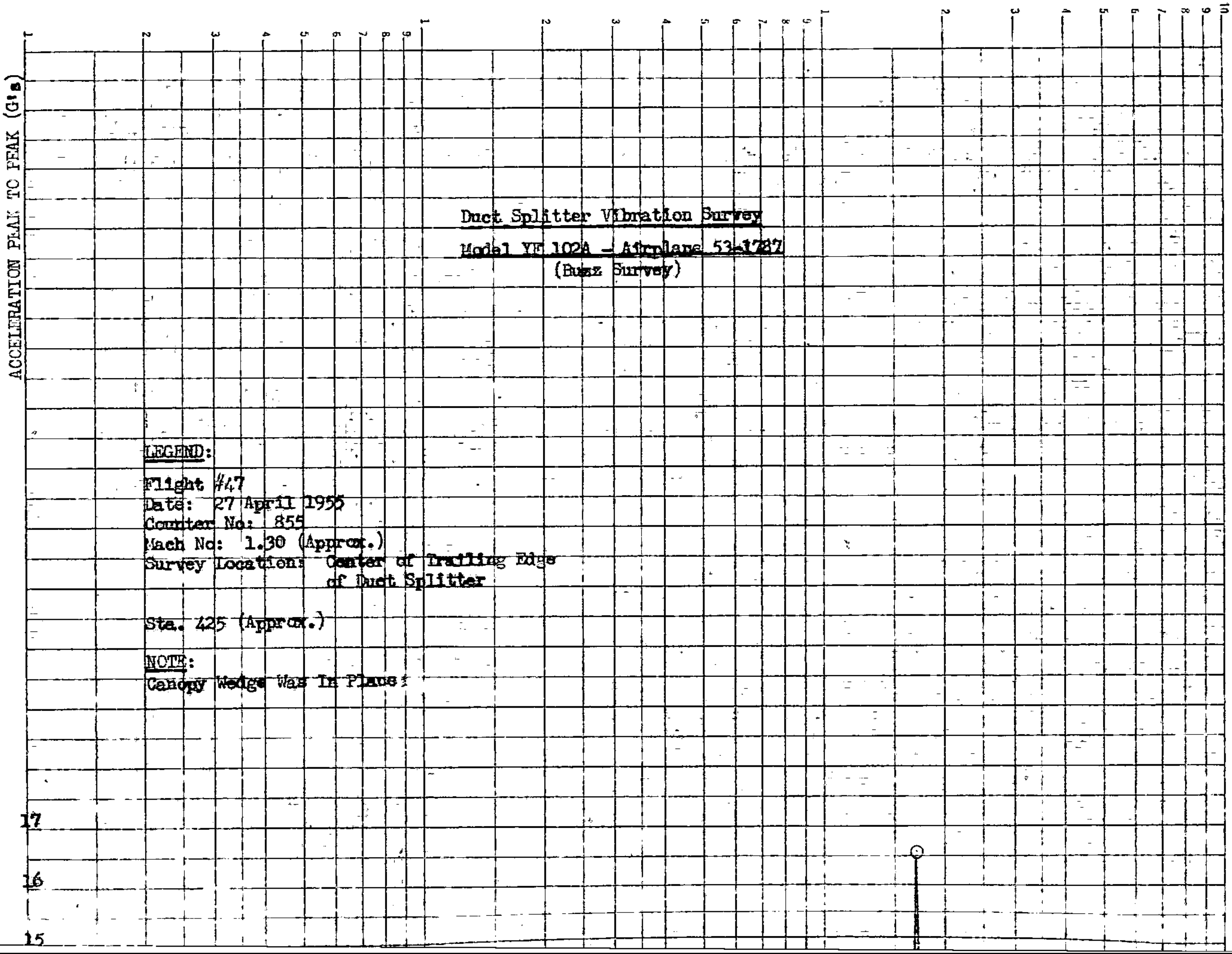
Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 853
Mach No: 1.29 (Approx.)
Survey Location: Center of Trailing Edge of Duct Splitter
Sta. 425 (Approx.)

NOTE:
Canopy Wedge Was In Place

Fig. 137

359-73L KEUFFEL & ESSER CO.
Semi-Logarithmic, 3 Cycles × 10 to the ½ inch.



Duct Splitter Vibration Survey

Model YF 102A - Airplane 53-1787

(Buzz Survey)

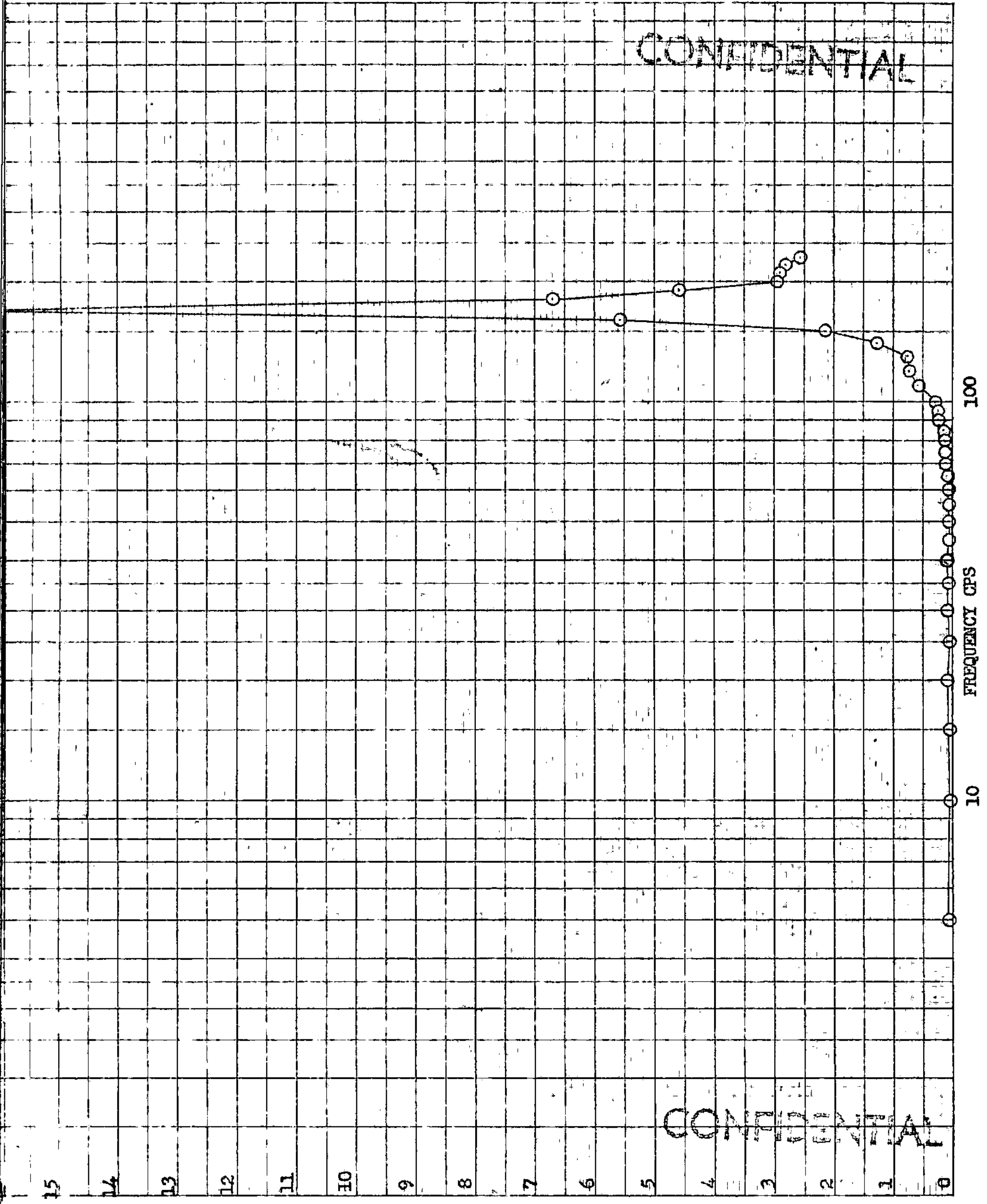
LEGEND:

Flight #47
Date: 27 April 1955
Counter No: 855
Mach No: 1.30 (Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter

Sta. 425 (Approx.)

NOTE:
Canopy Wedge Was In Place.

CONFIDENTIAL

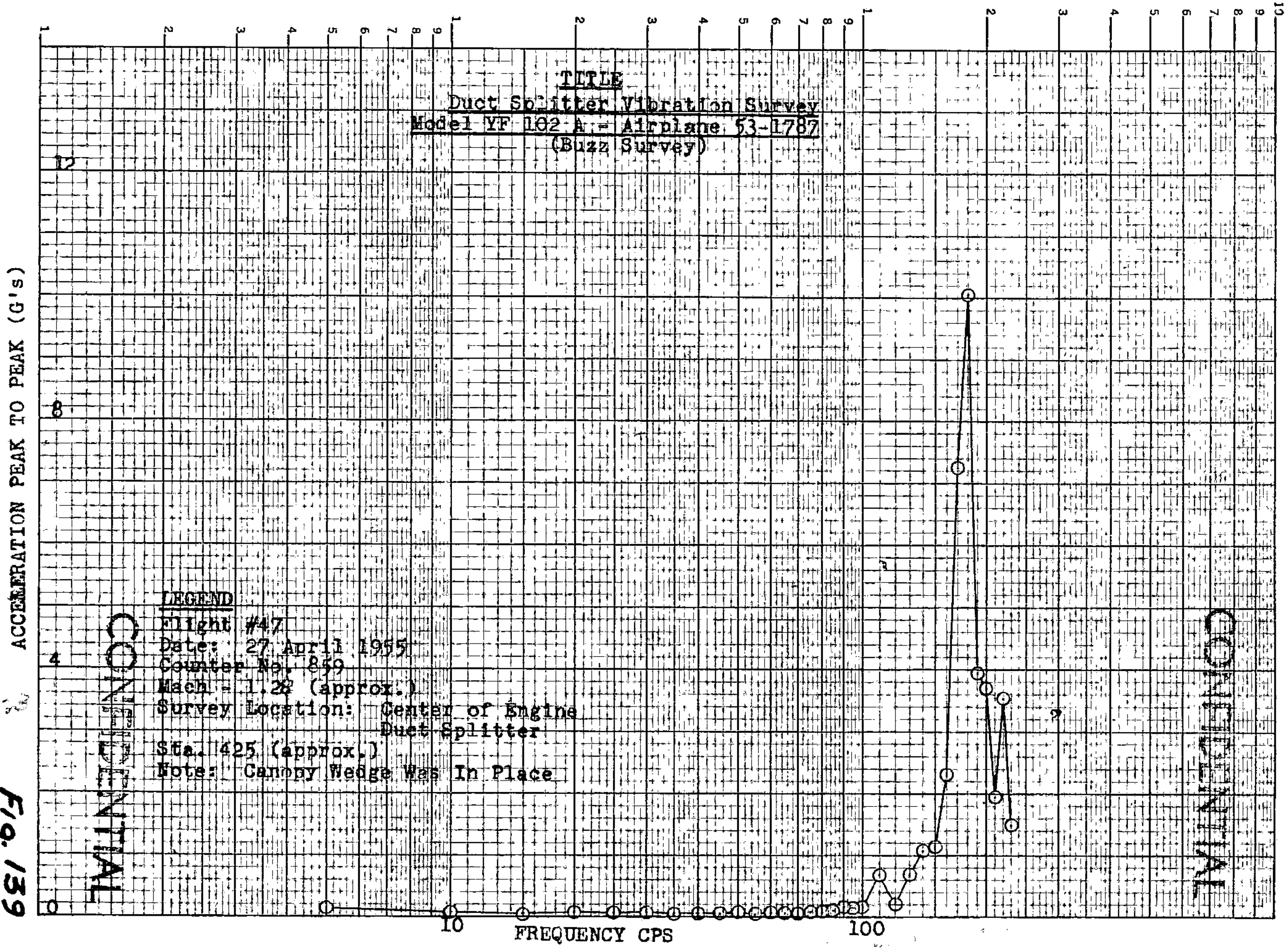


CONFIDENTIAL

Fig. 138

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



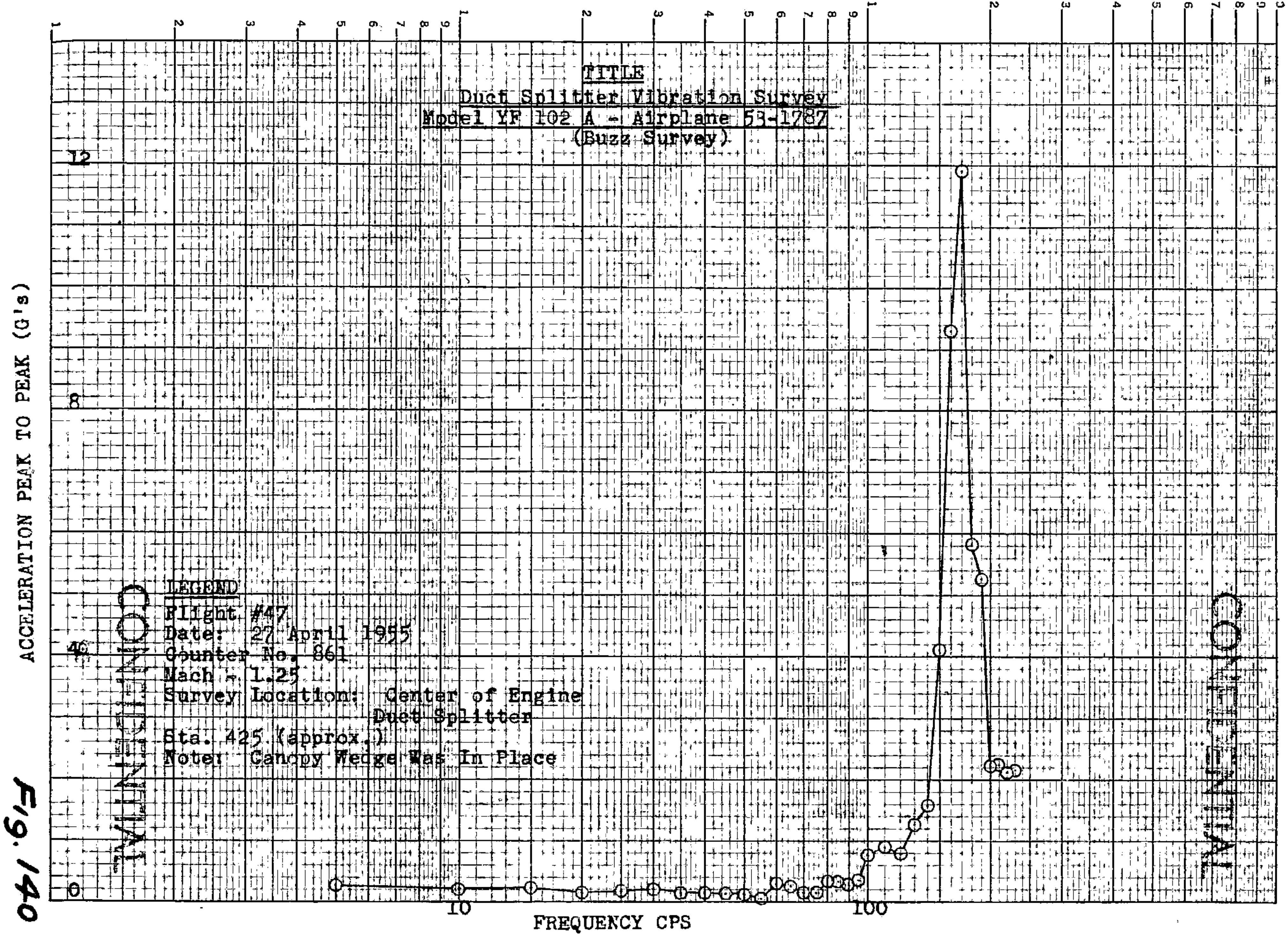
TITLE
Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 859
Mach = 1.22 (approx.)
Survey Location: Center of Engine
Duct Splitter
Sta. 425 (approx.)
Note: Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 139

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

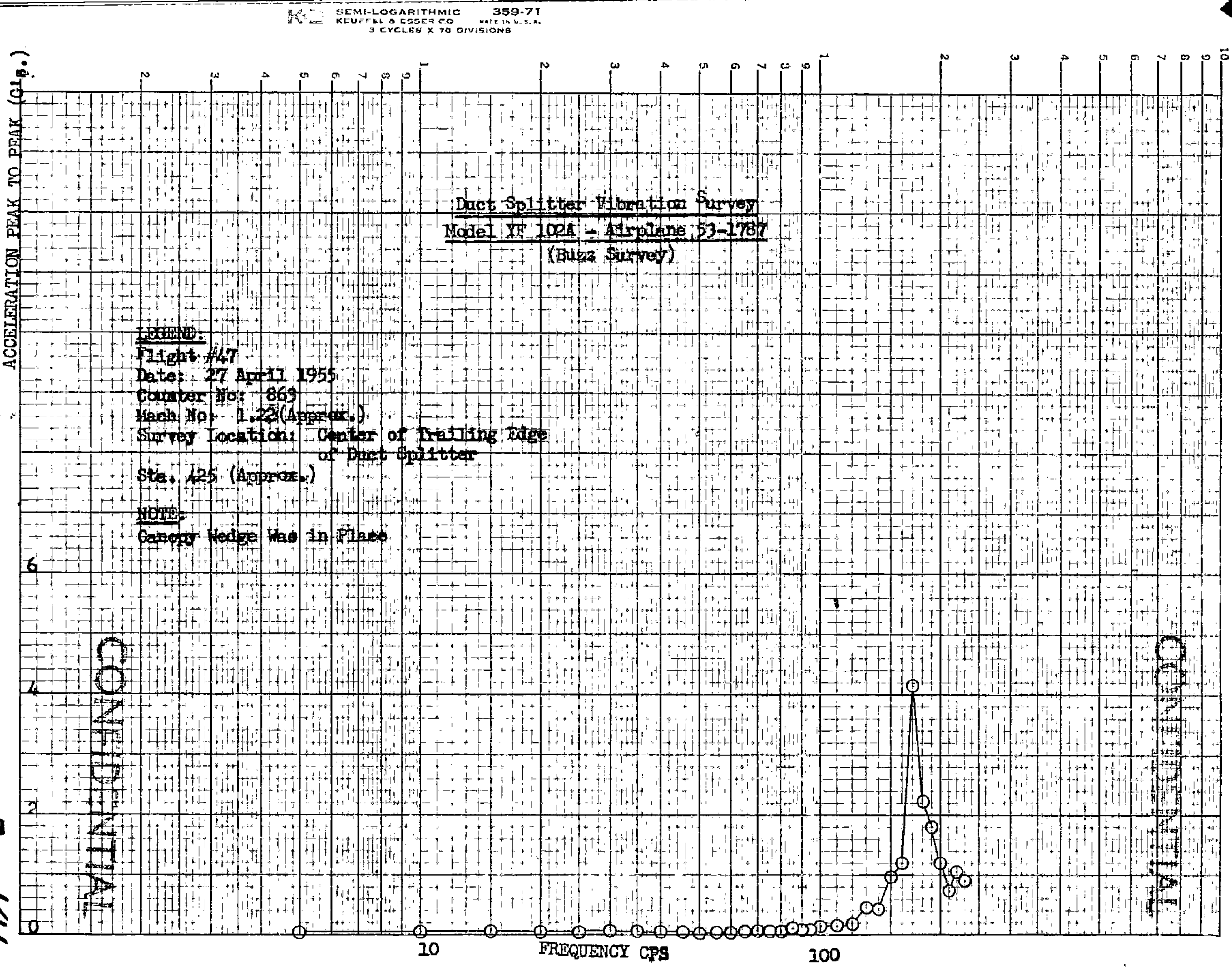


TITLE
Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 861
Mach - 1.25
Survey Location: Center of Engine
Duct Splitter
Sta. 425 (approx.)
Note: Canopy Wedge Was In Place

Fig. 140

CONFIDENTIAL



Duct Splitter Vibration Survey

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:

Flight #47

Date: 27 April 1955

Counter No: 869

Mach No: 1.22 (Approx.)

Survey Location: Center of Trailing Edge of Duct Splitter

Sta. 425 (Approx.)

NOTE:

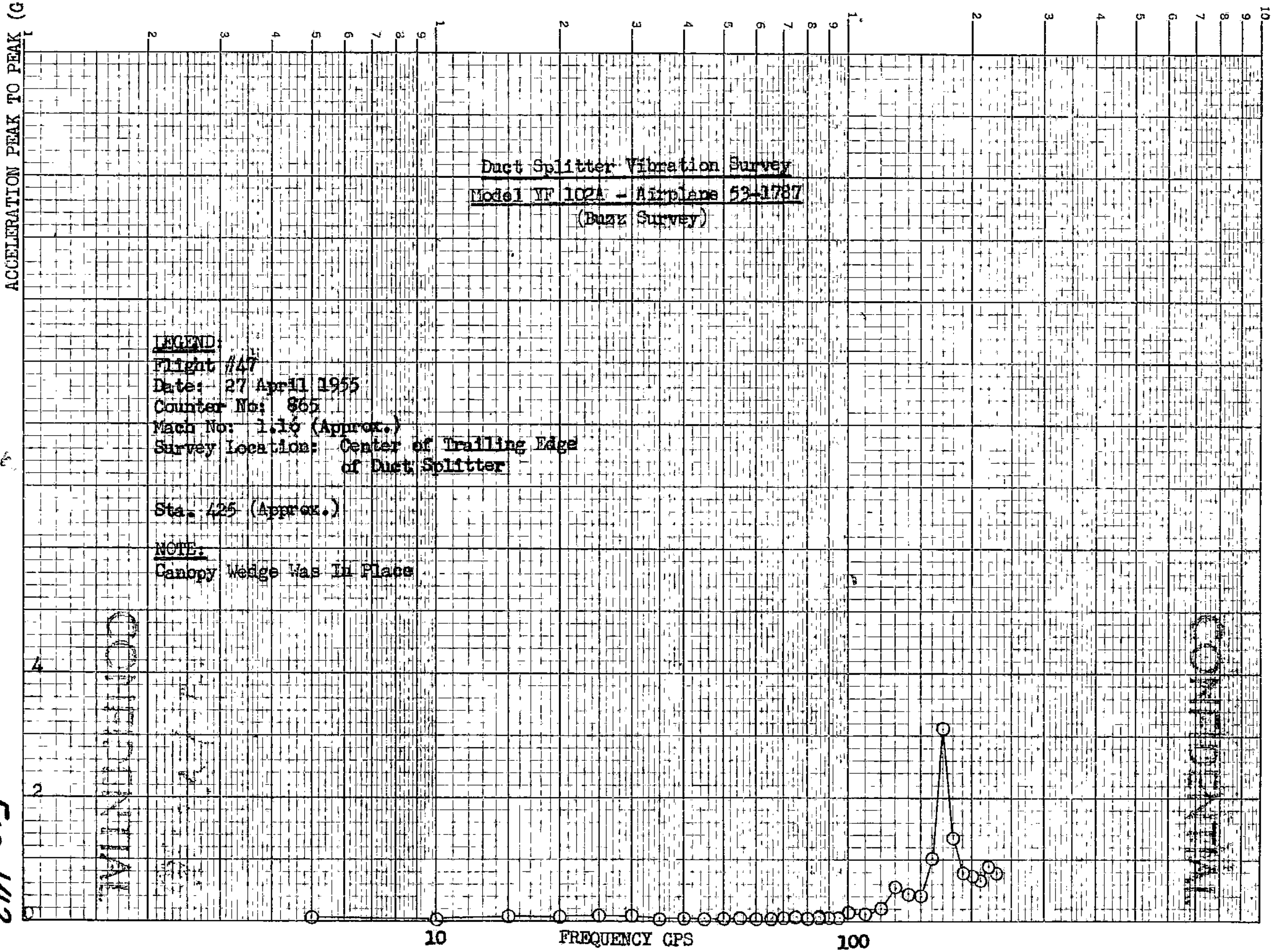
Canopy Wedge Was in Place

CONFIDENTIAL

Fig. 141

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

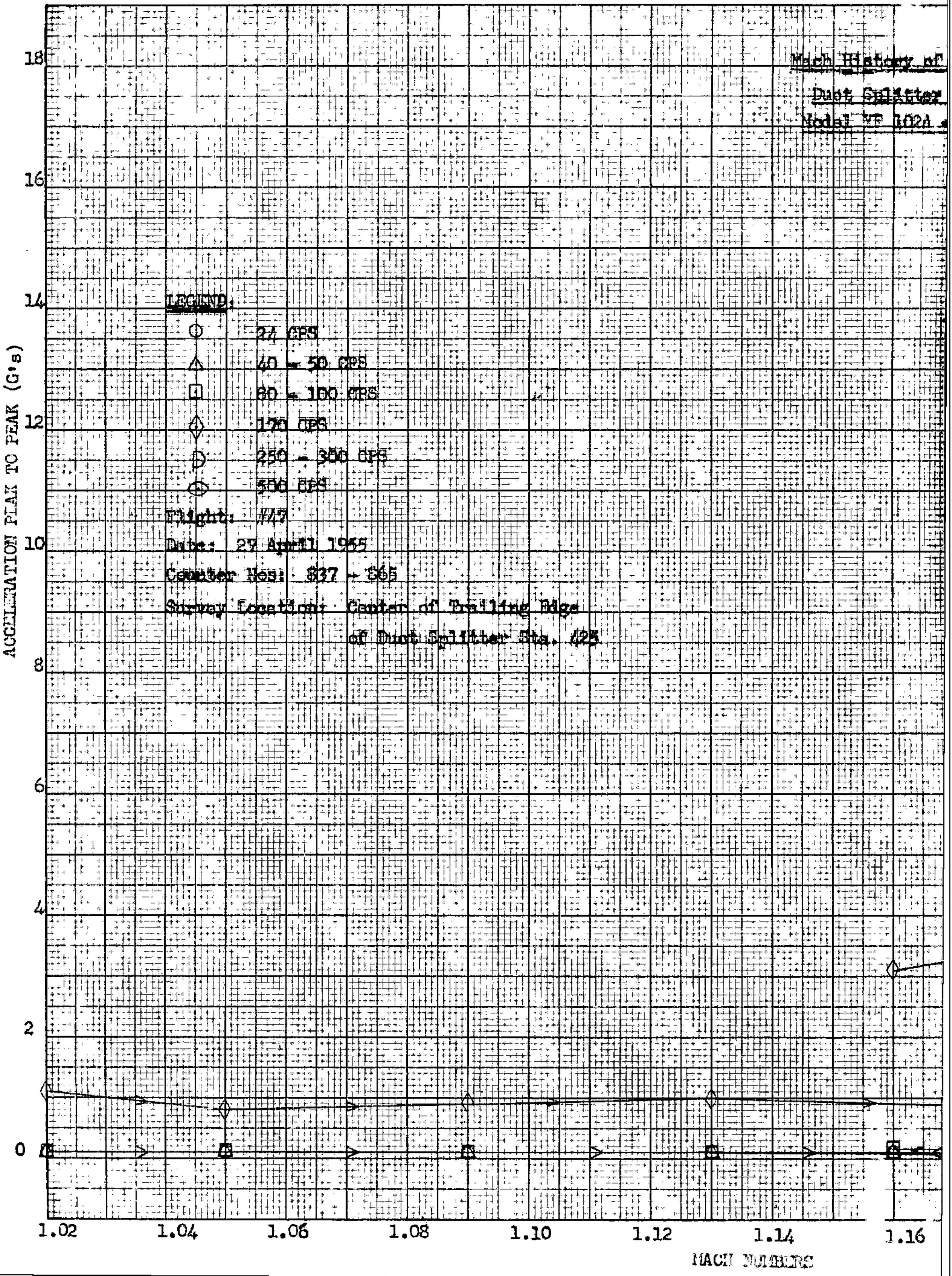
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 865
Mach No: 1.16 (Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter

Sta. 425 (Approx.)

NOTE:
Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 142



Mach History of
Duct Splitter
Model YF 102A

LEGEND:
○ 24 CPS
△ 40 - 50 CPS
□ 80 - 100 CPS
◇ 170 CPS
D 250 - 350 CPS
⊖ 500 CPS

Flight: #47
Date: 27 April 1955
Counter Nos: 337 - 365
Survey Location: Center of Trailing Edge
of Duct Splitter Sta. 425

K+E 10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO. MADE IN U.S.A.
ALBANENE ®

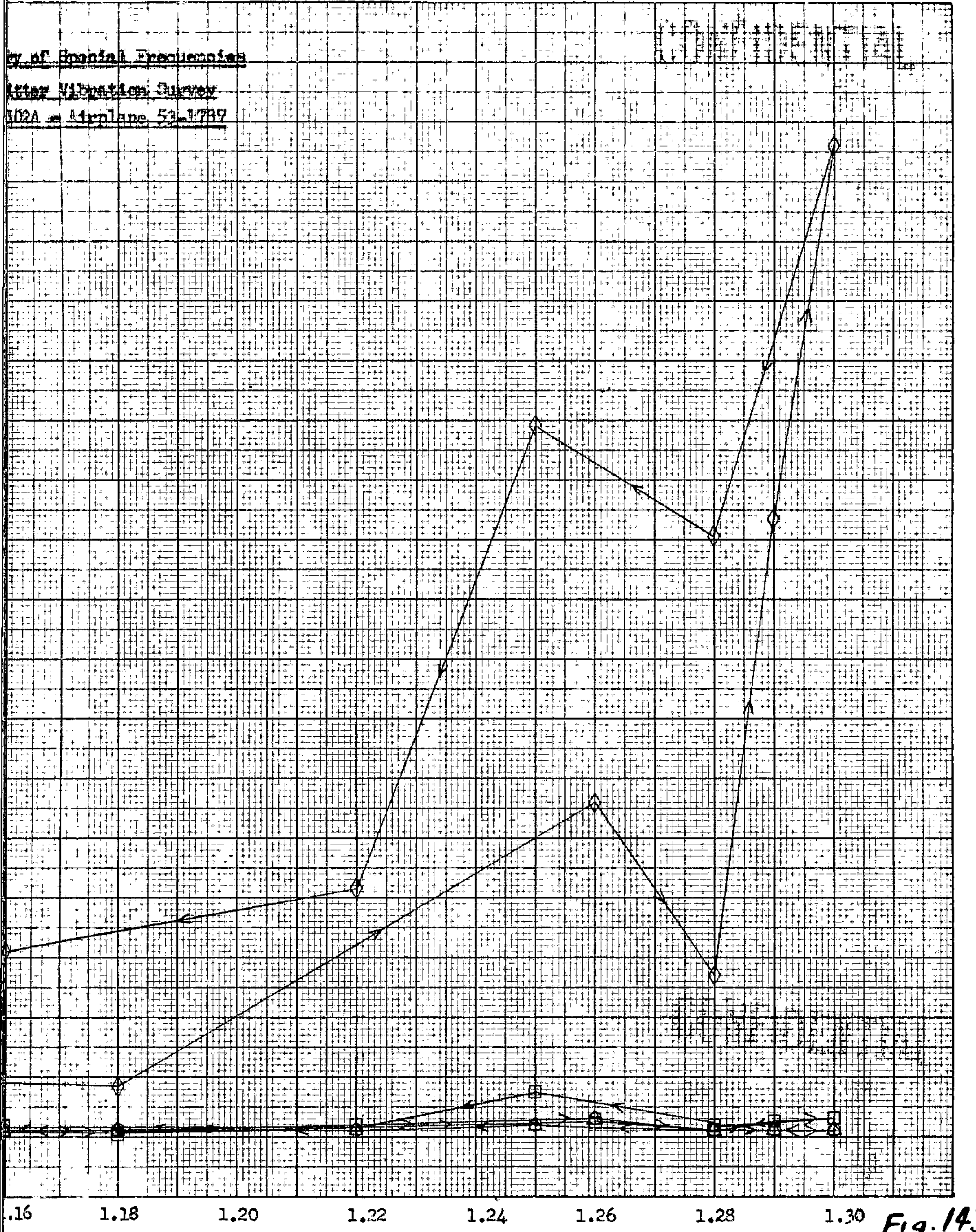
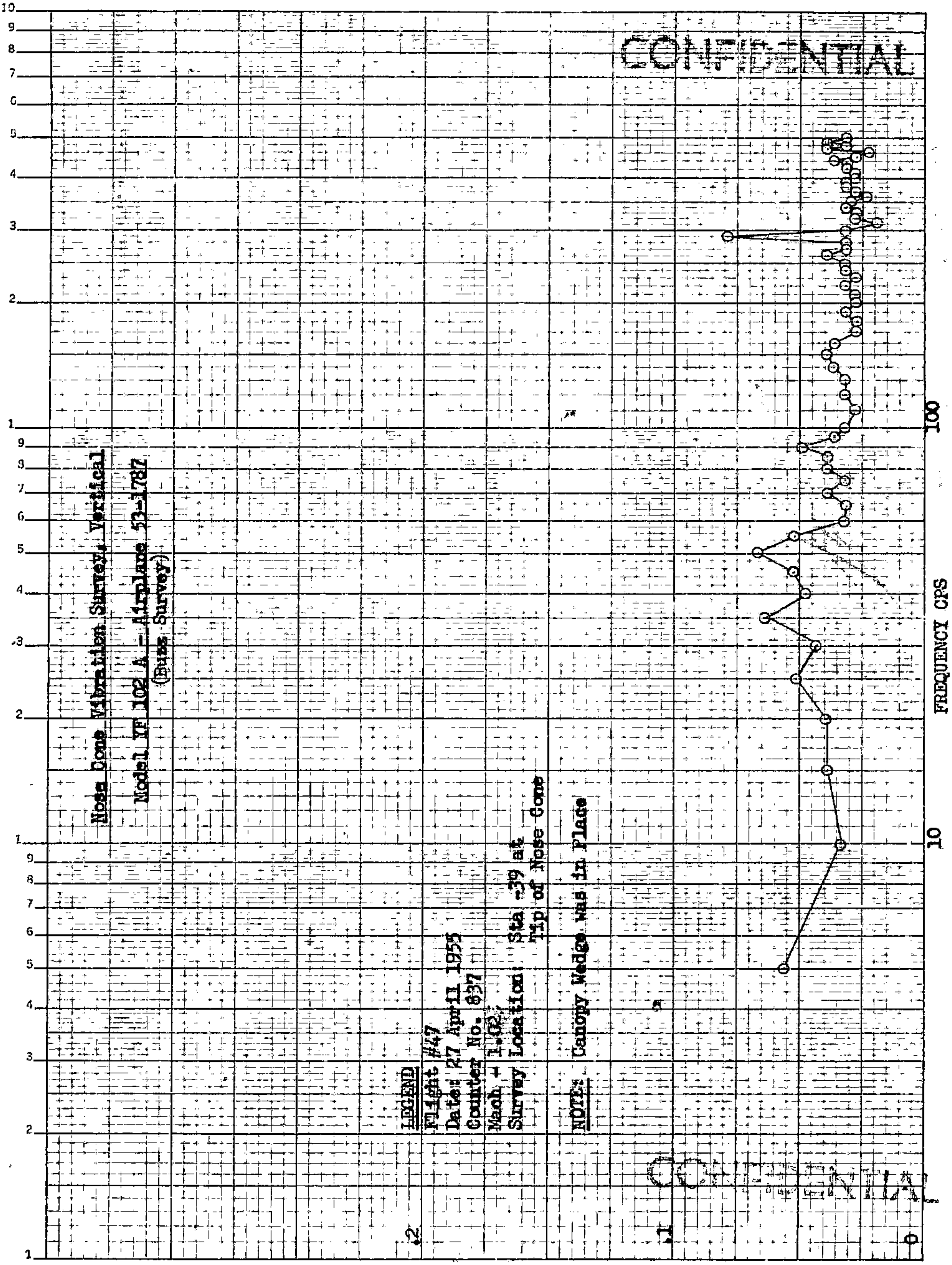

y of Special Frequencies
tter Vibration Survey
102A - Airplane 53-1787
.16
1.18
1.20
1.22
1.24
1.26
1.28
1.30

Fig. 143

CONFIDENTIAL

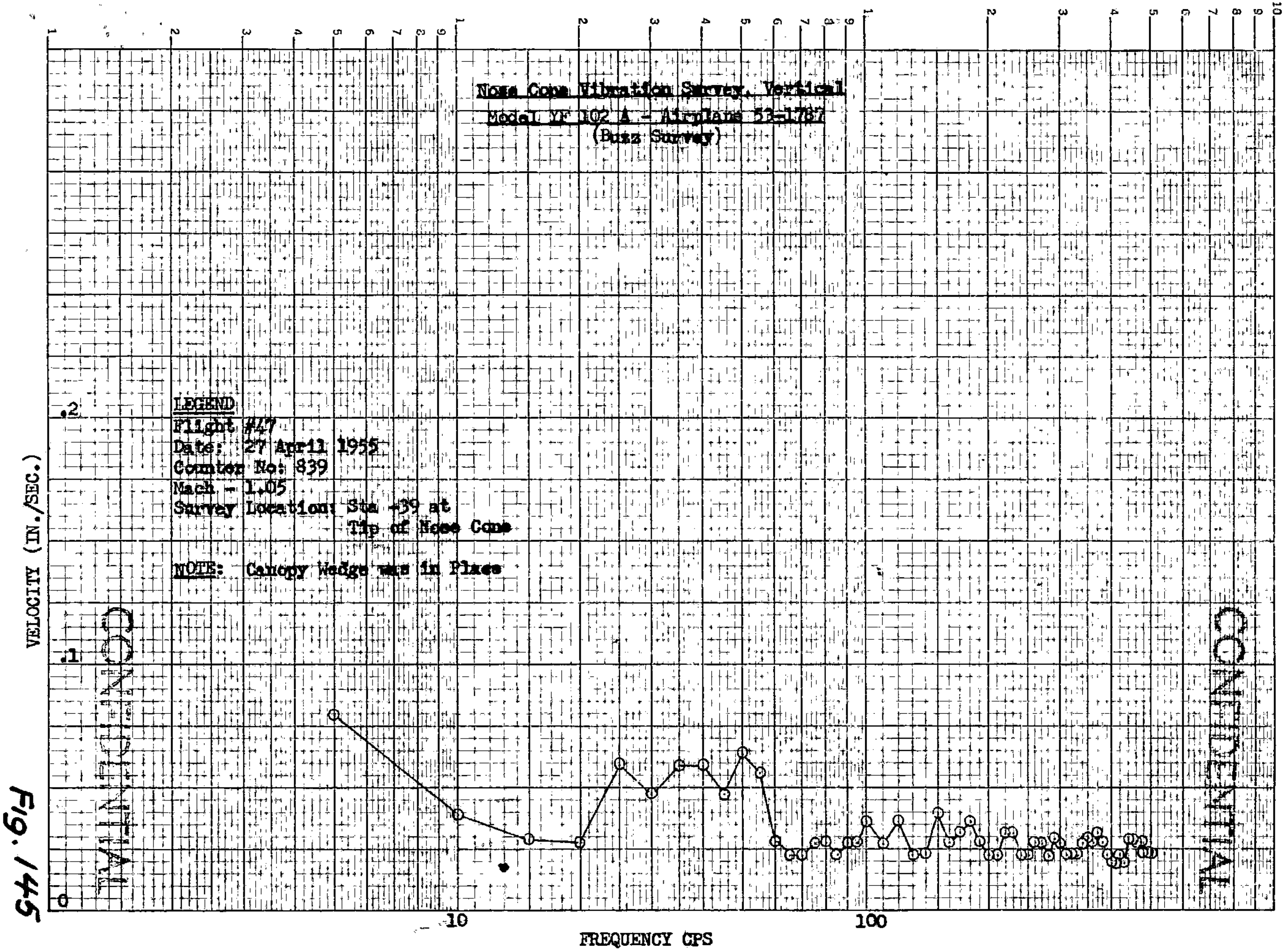


CONFIDENTIAL

Fig. 144

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No: 839
Mach - 1.05
Survey Location: Sta -39 at
Tip of Nose Cone

NOTE: Canopy Wedge was in Place

CONFIDENTIAL

Fig. 145

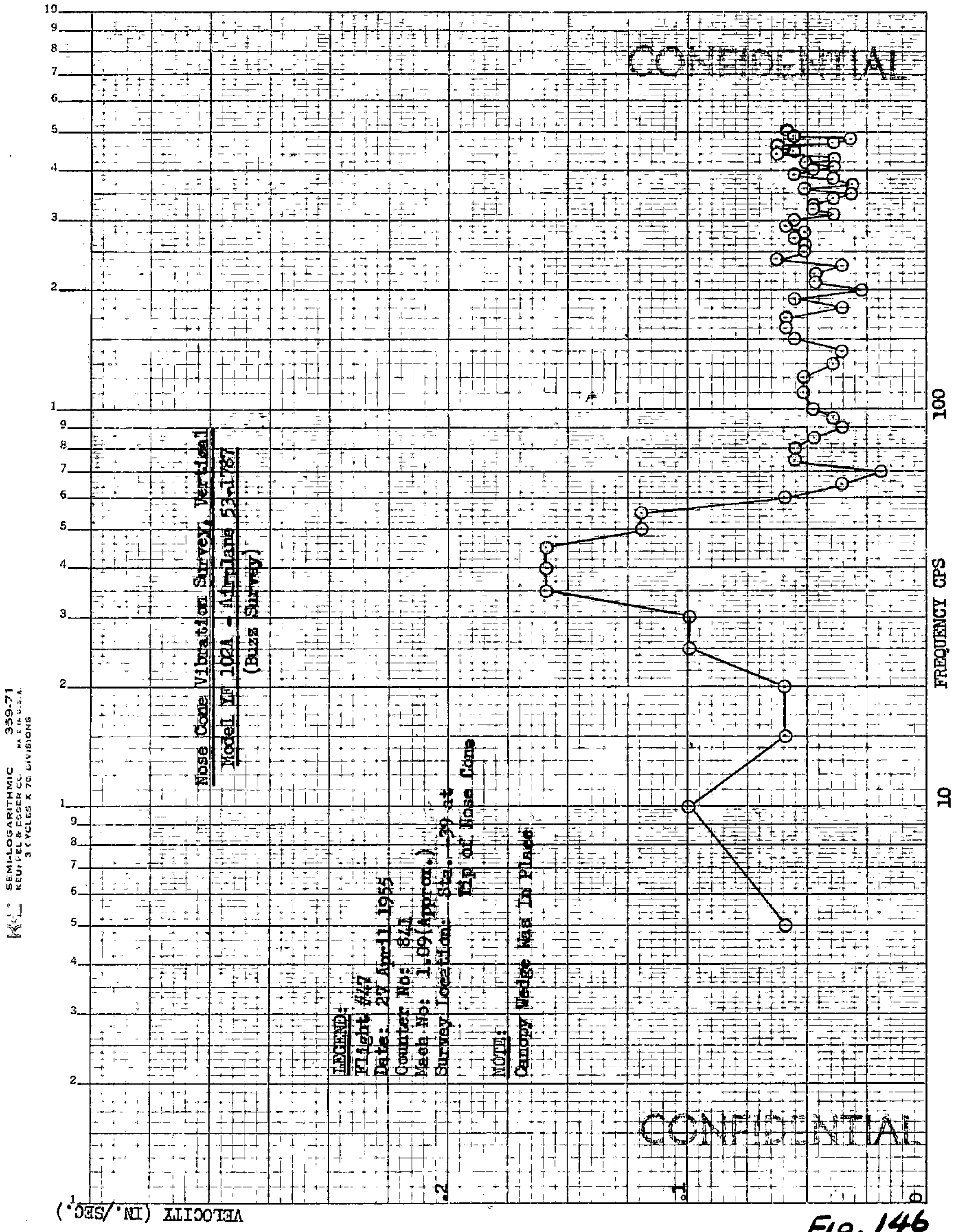
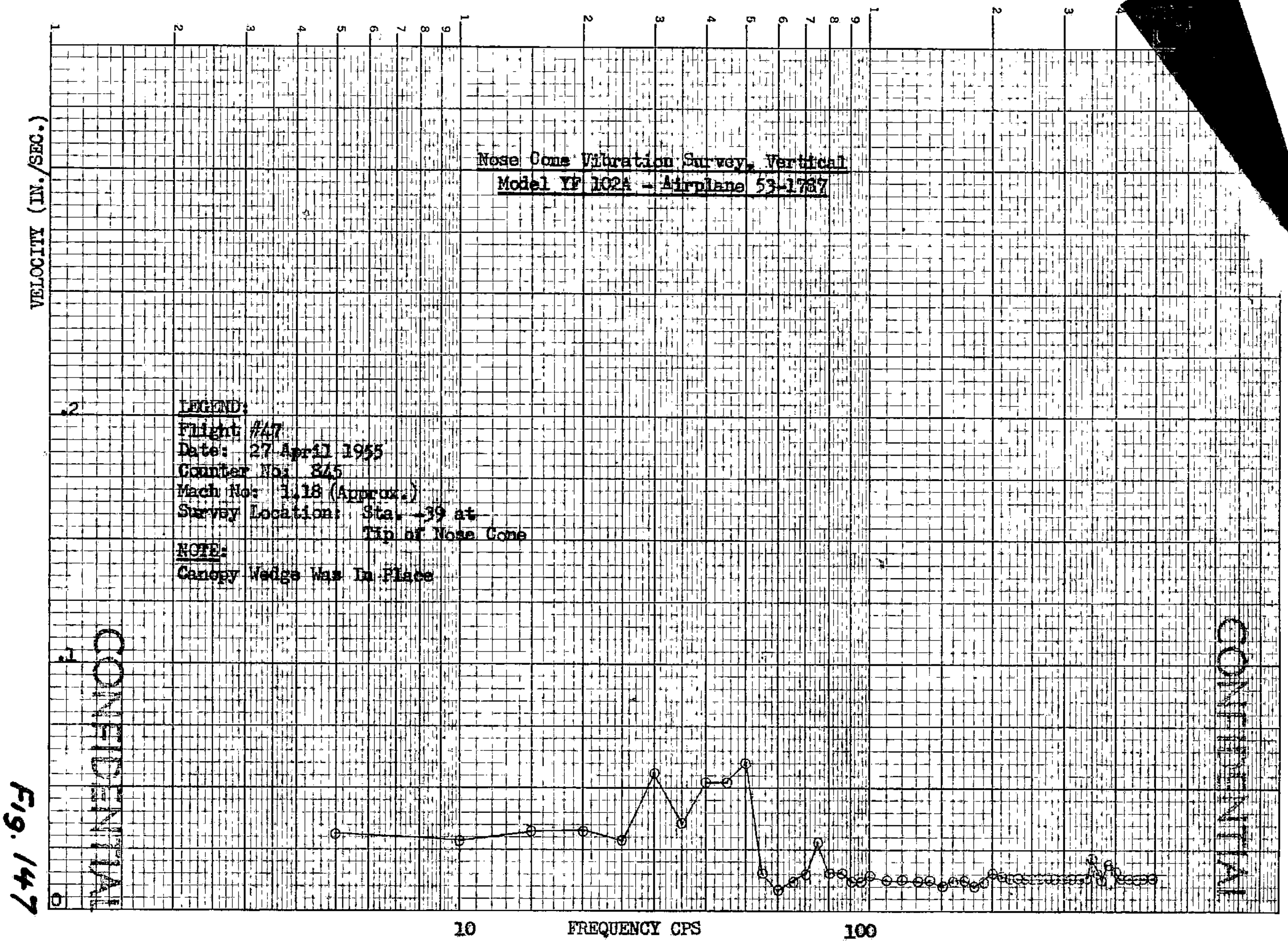

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS
CONFIDENTIAL
Nose Cone Vibration Survey, Vertical
Model XF 102A - Airplane 53-1787
(Buzz Survey)
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 841
Mach No: 1.09(Approx.)
Survey Location: Sta. 29 at Tip of Nose Cone
NOTE:
Canopy Wedge Was In Place
CONFIDENTIAL
VELOCITY (IN./SEC.)
FREQUENCY CPS
10
100
.1
.2
1
10

Fig. 146

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical
Model YF 102A - Airplane 53-1787

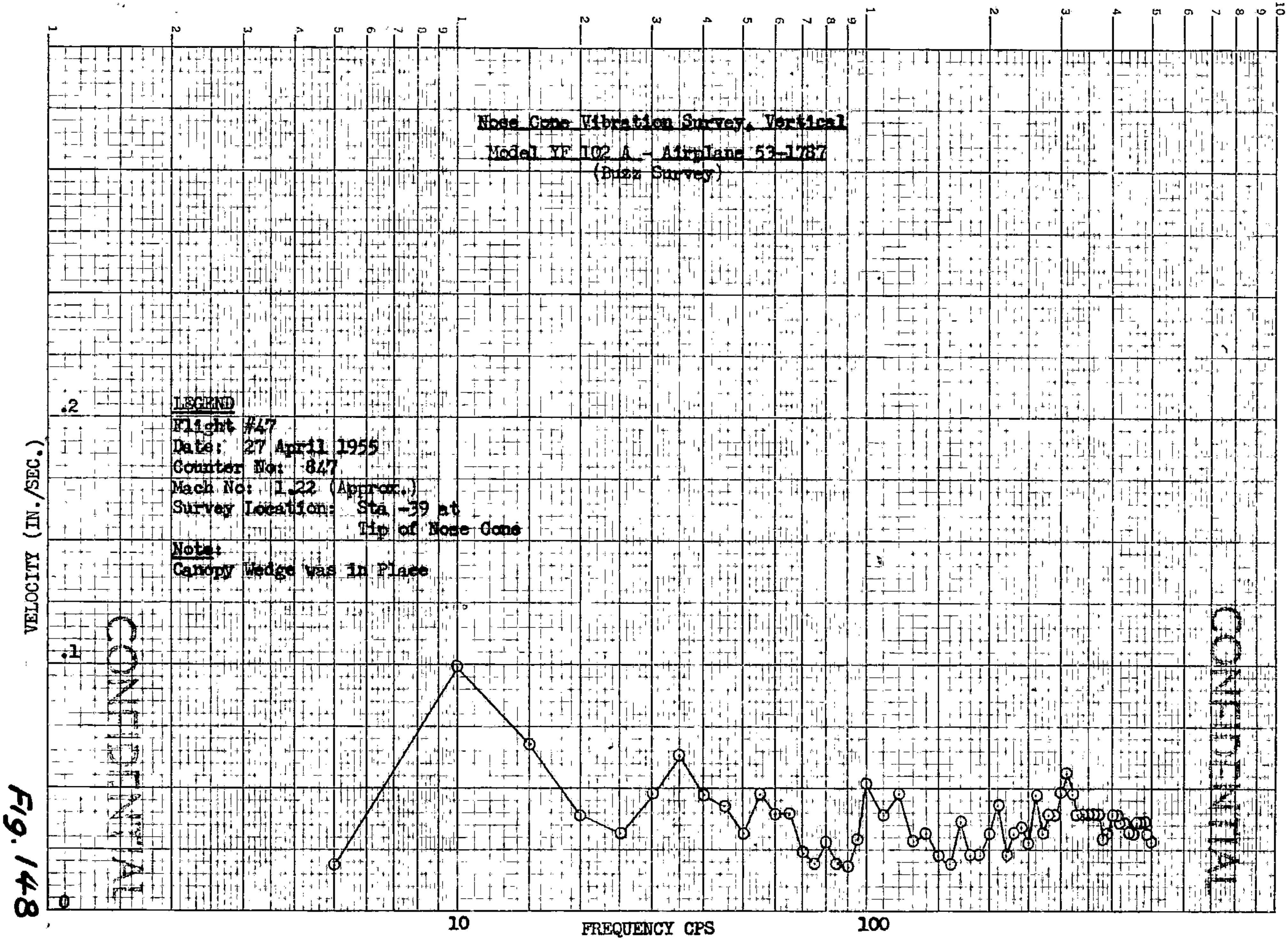
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 845
Mach No: 1.18 (Approx.)
Survey Location: Sta. -39 at
Tip of Nose Cone

NOTE:
Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 147

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical

Model YF 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND

Flight #47

Date: 27 April 1955

Counter No: 847

Mach No: 1.22 (Approx.)

Survey Location: Sta -39 at Tip of Nose Cone

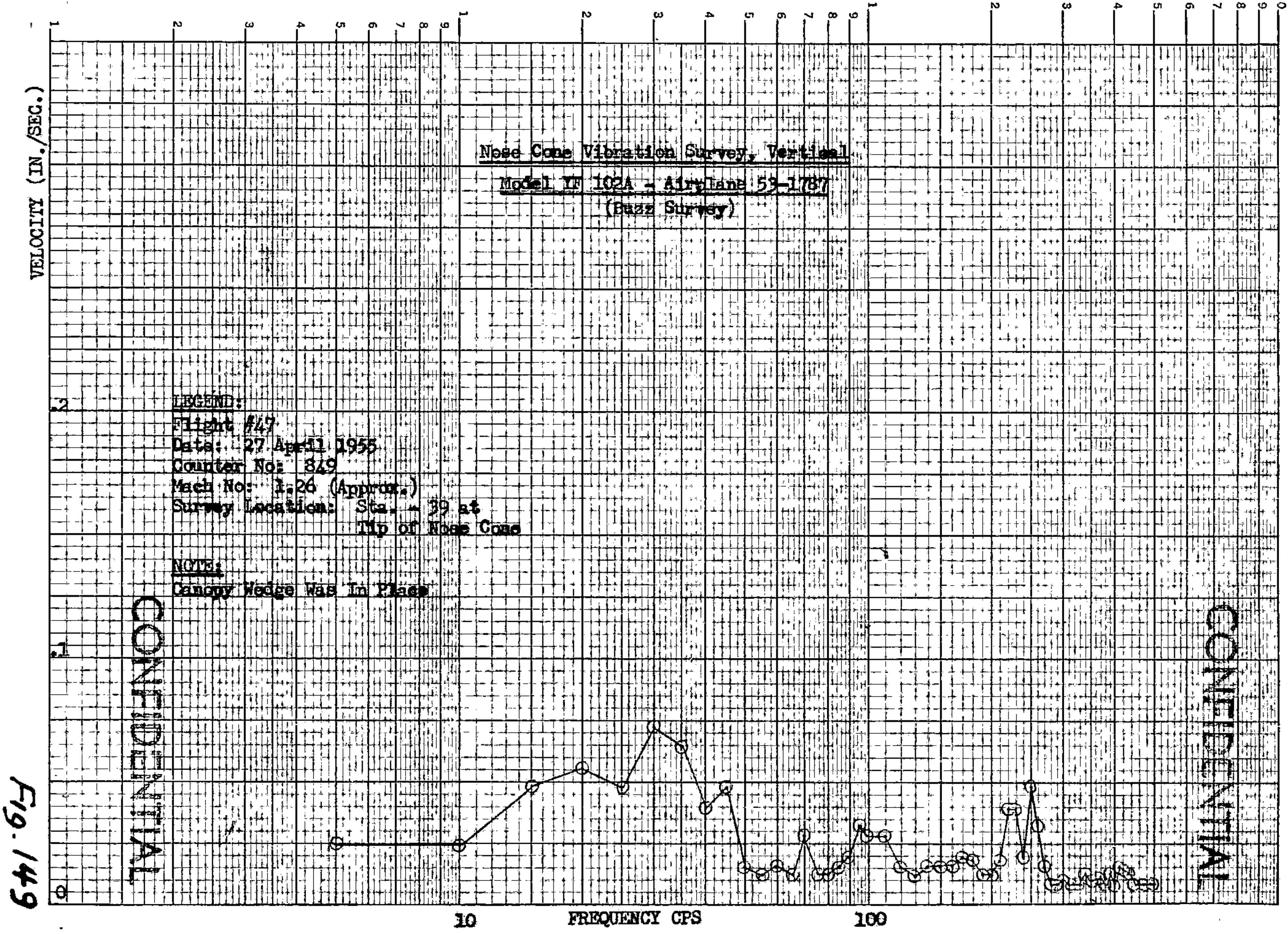
Note:

Canopy Wedge was in Place

CONFIDENTIAL

Fig. 148

K•E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



Nose Cone Vibration Survey, Vertical

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 849
Mach No: 1.26 (Approx.)
Survey Location: Sta. - 39 at
Tip of Nose Cone

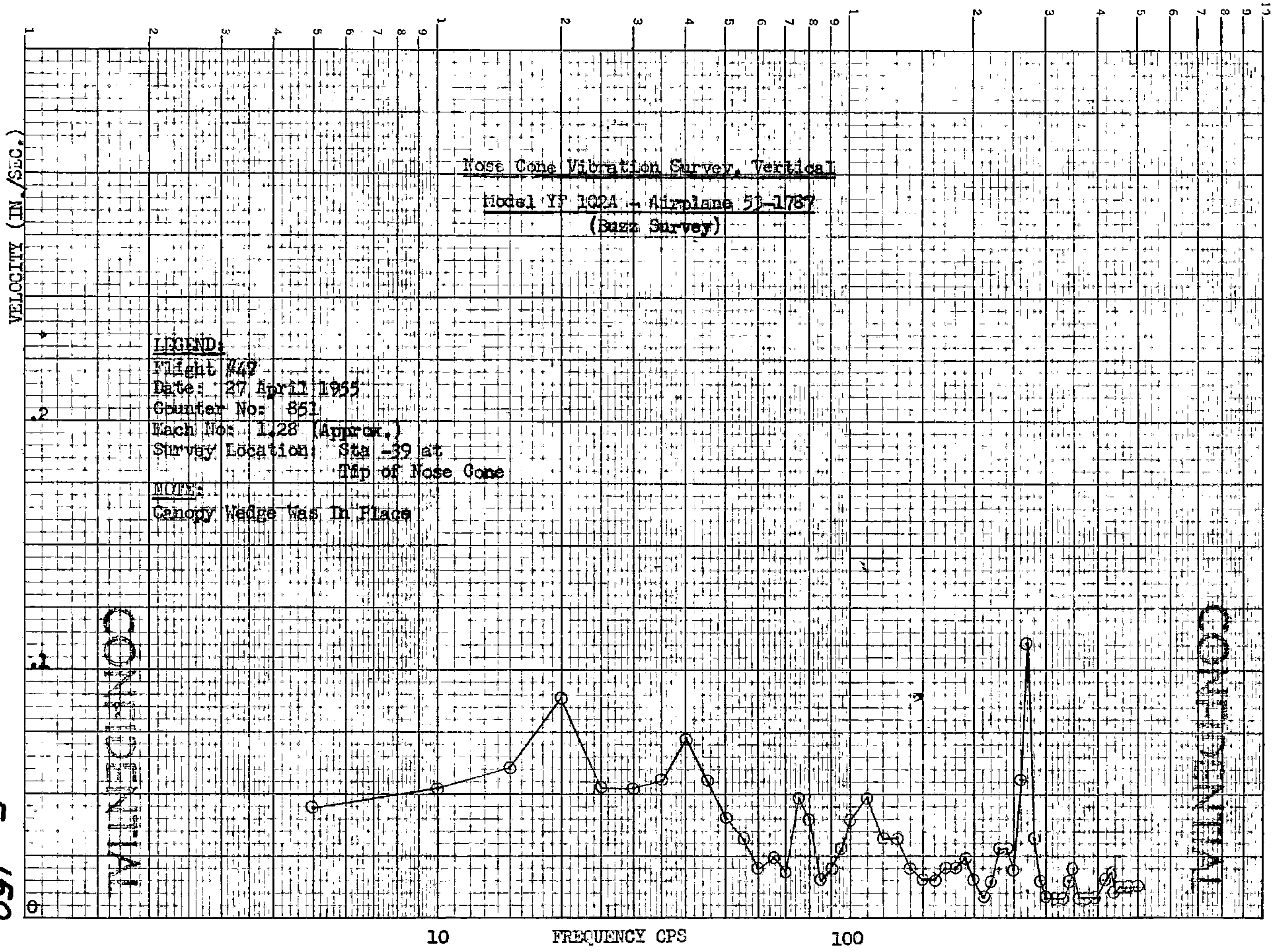
NOTE:
Canopy Wedge Was In Place

CONFIDENTIAL

CONFIDENTIAL

Fig. 149

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



Nose Cone Vibration Survey, Vertical

Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 851
Mach No: 1.28 (Approx.)
Survey Location: Sta -39 at
Tip of Nose Cone

NOTE:
Canopy Wedge Was In Place

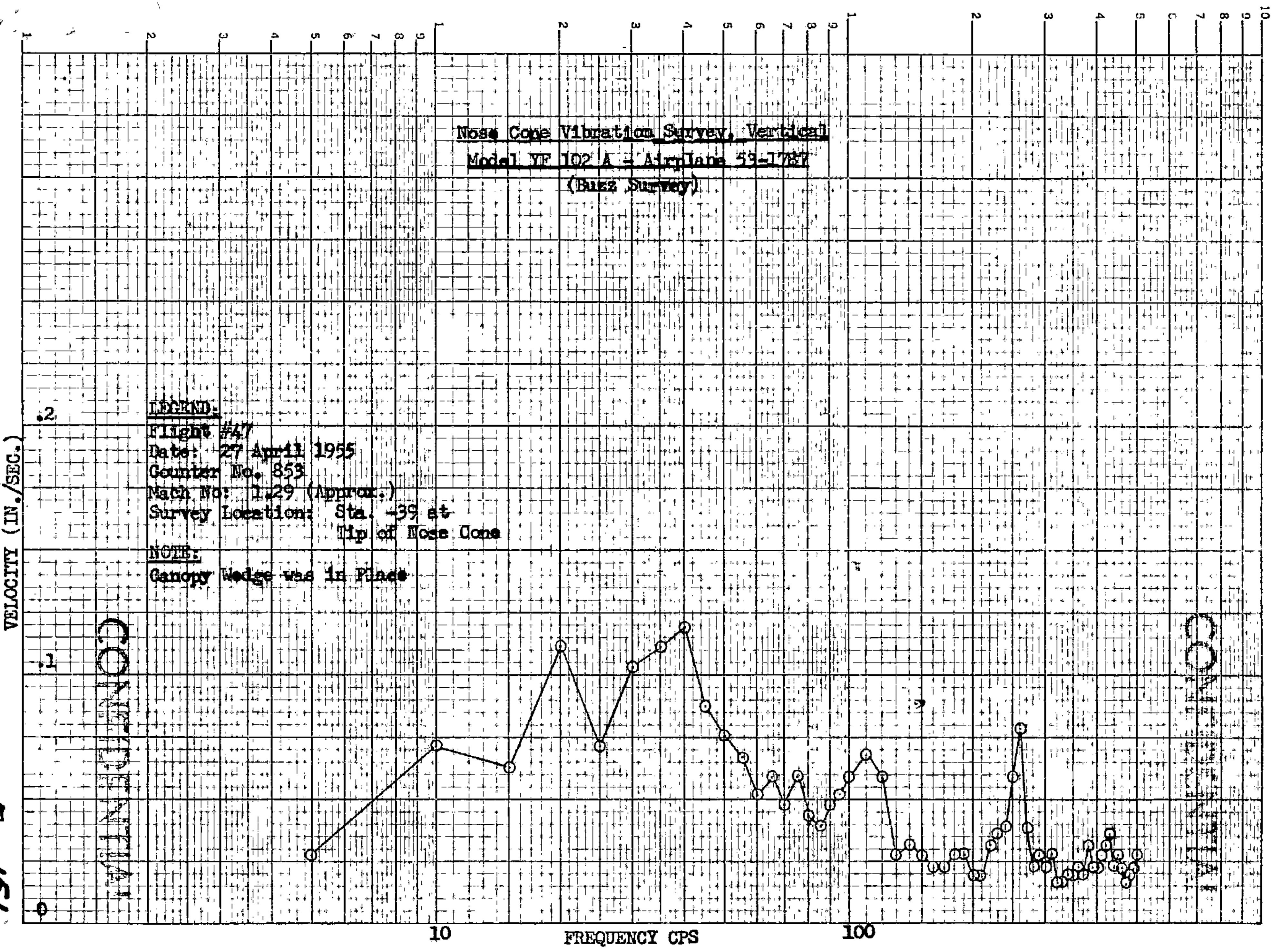
CONFIDENTIAL

CONFIDENTIAL

Fig. 150

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical

Model YF 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No. 853
Mach No: 1.29 (Approx.)
Survey Location: Sta. -39 at
Tip of Nose Cone

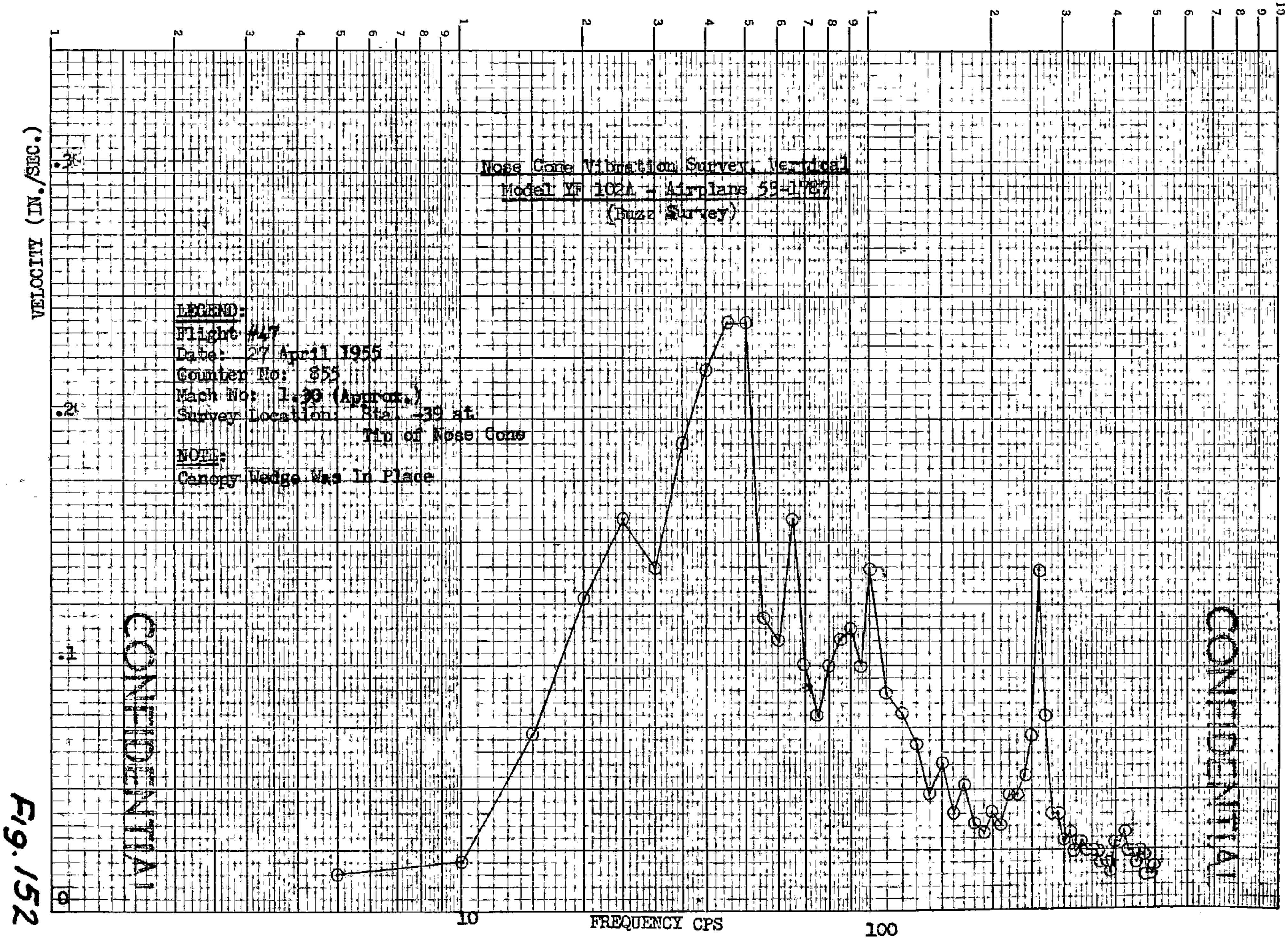
NOTE:
Canopy Wedge was in Place

CONFIDENTIAL

Fig. 151

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical
Model YF 102A - Airplane 53-1787
(Buzz Survey)

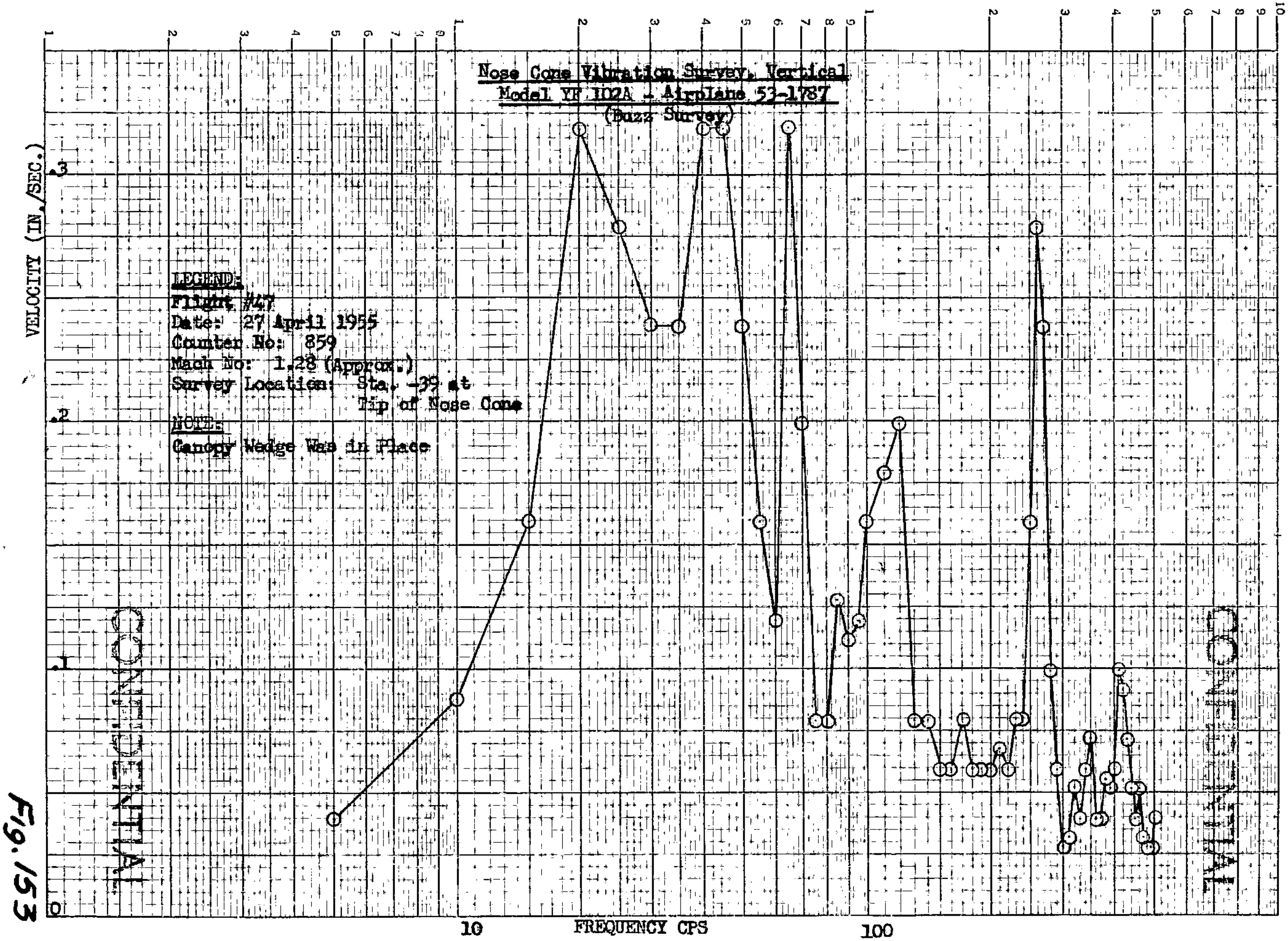
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 855
Mach No: 1.30 (Approx.)
Survey Location: Sta. -39 at
Tip of Nose Cone

NOTE:
Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 152

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



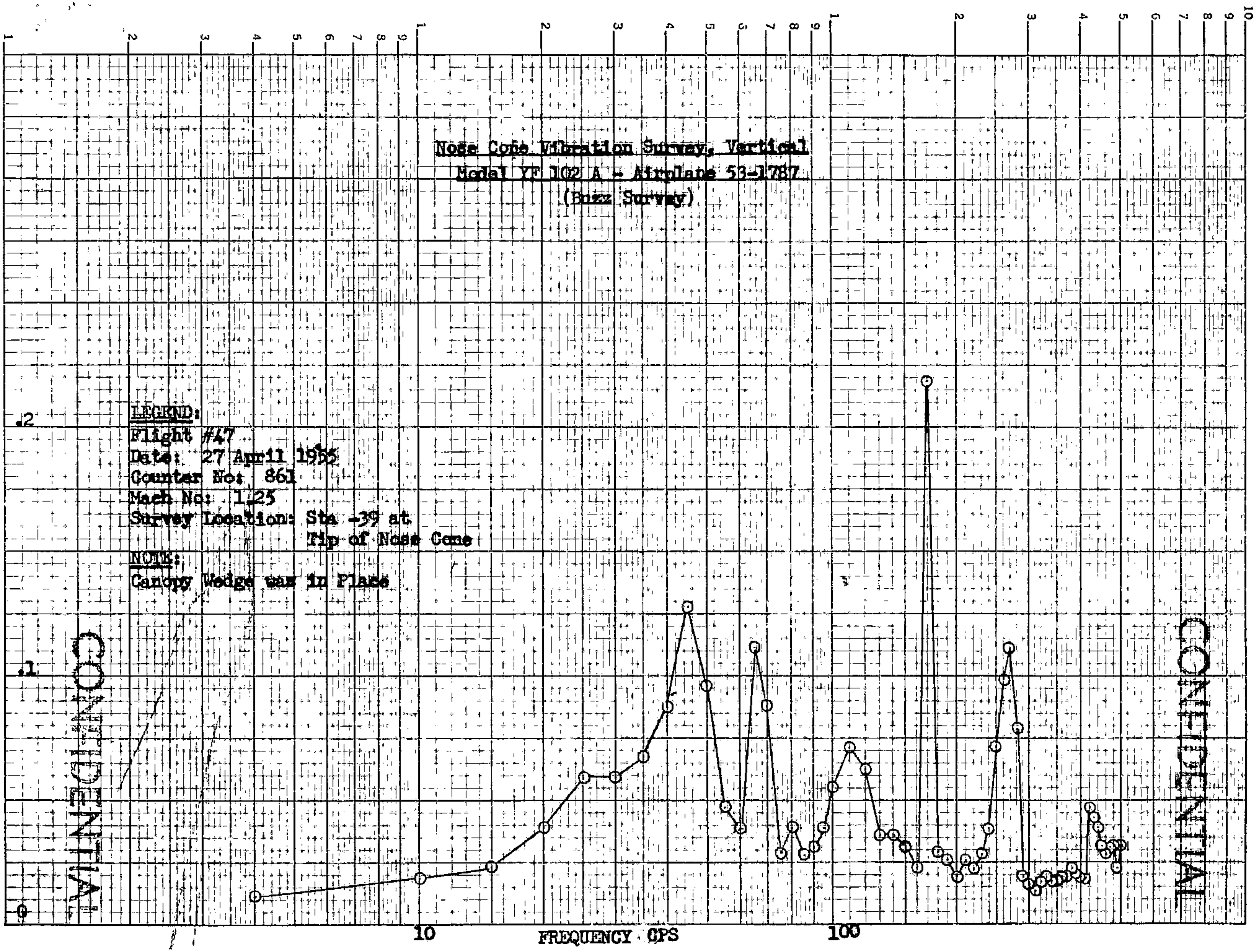
Nose Cone Vibration Survey, Vertical
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 859
Mach No: 1.28 (Approx.)
Survey Location: Sta. -39 at Tip of Nose Cone

NOTE:
Canopy Wedge Was in Place

Fig. 153

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 861
Mach No: 1.25
Survey Location: Sta -39 at
Tip of Nose Cone

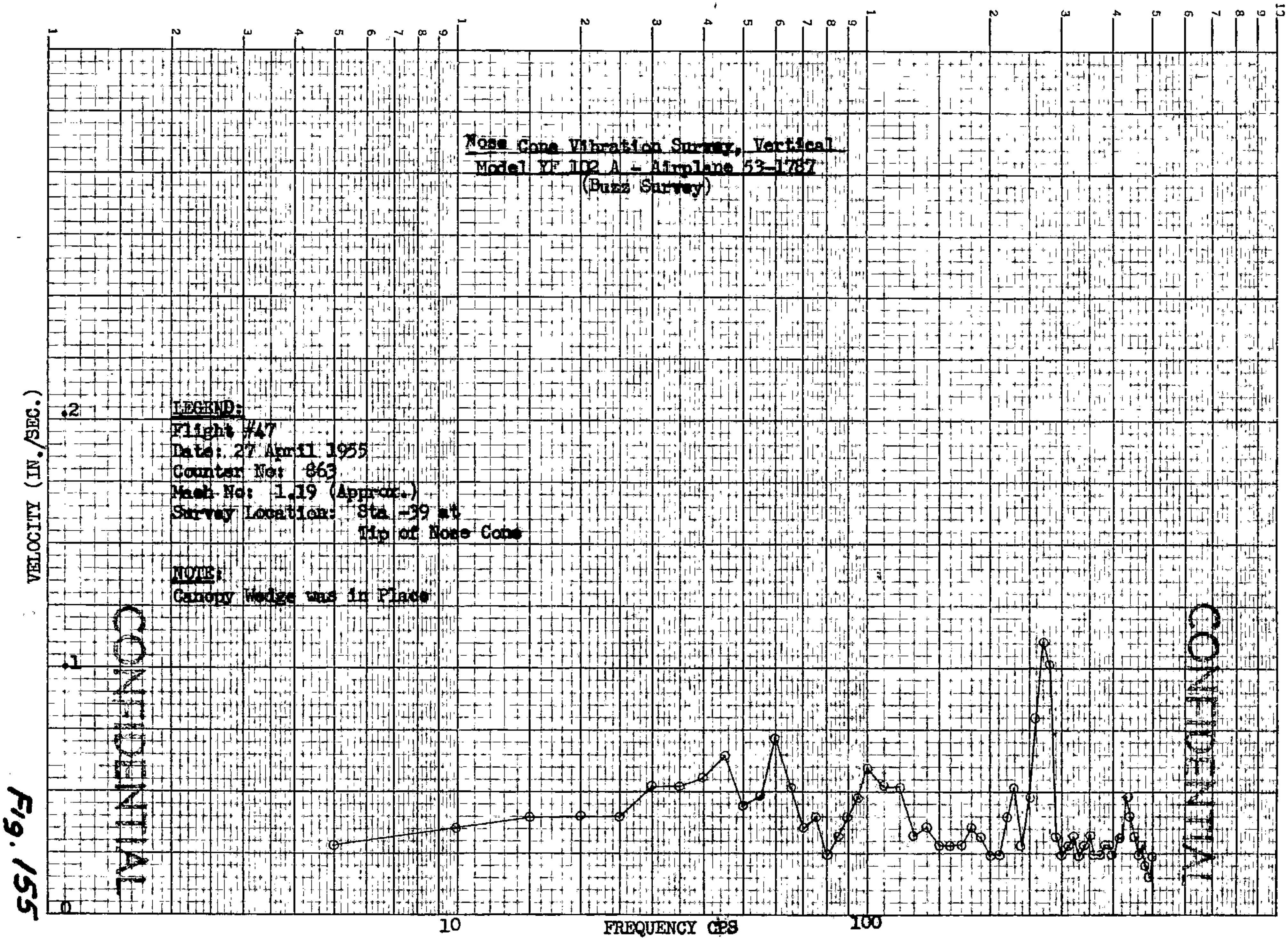
NOTE:
Canopy Wedge was in Place

CONFIDENTIAL

Fig. 154

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 863
Mach No: 1.19 (Approx.)
Survey Location: Sta -39 at
Tip of Nose Cone

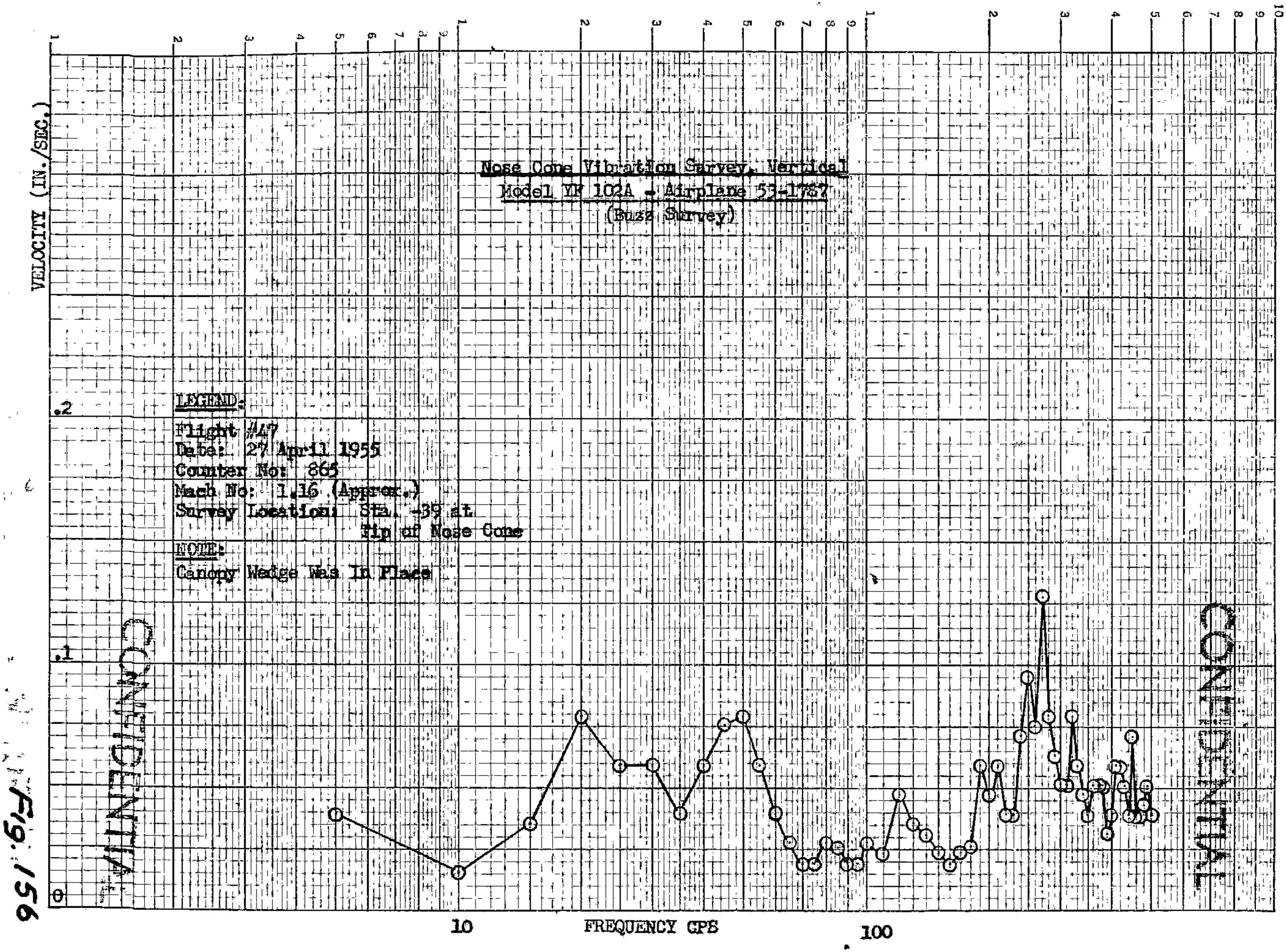
NOTE:
Canopy Wedge was in Place

CONFIDENTIAL

Fig. 155

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



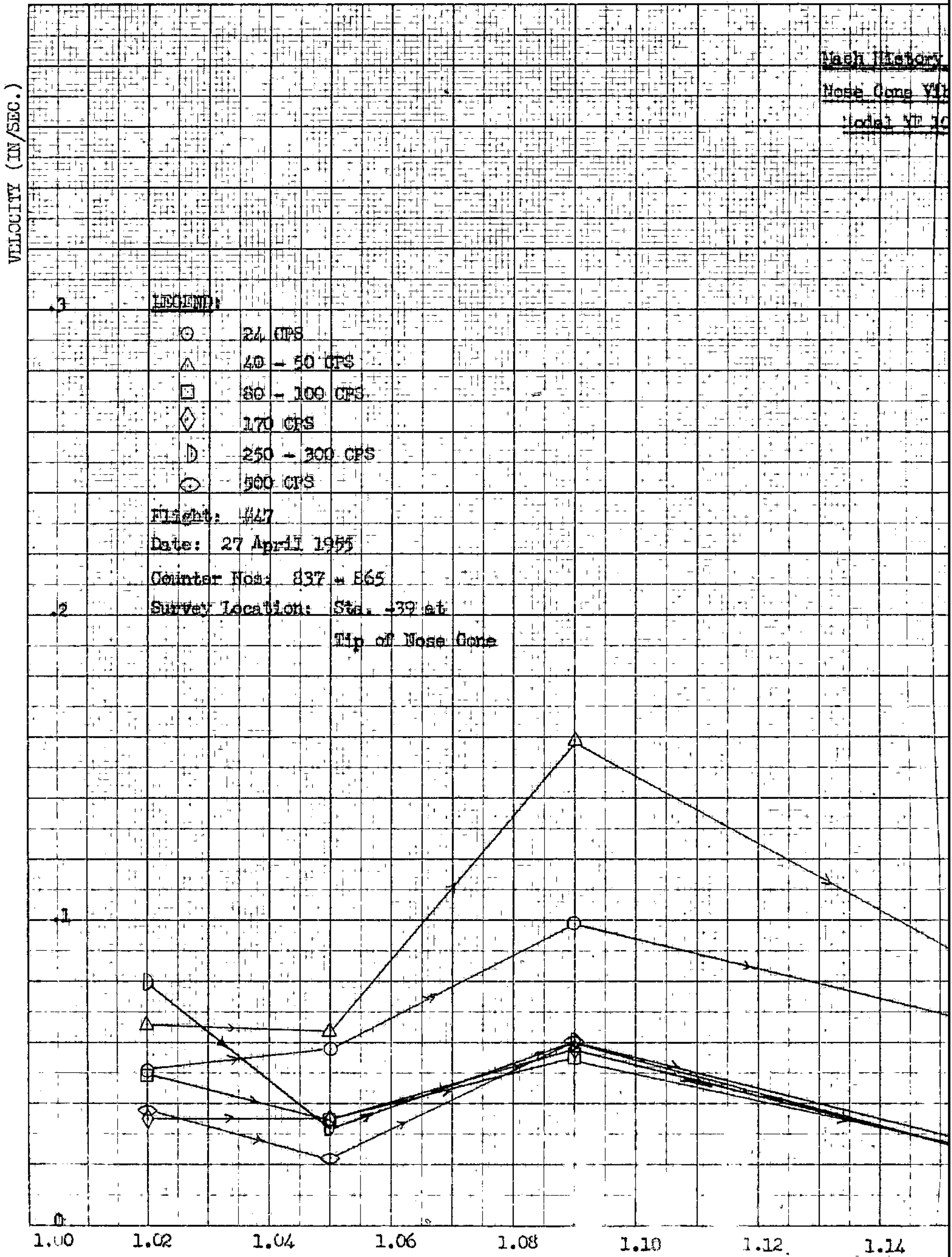
Nose Cone Vibration Survey, Vertical
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 865
Mach No: 1.16 (Approx.)
Survey Location: Sta. -39 at
Tip of Nose Cone

NOTE:
Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 156


Mach History
Nose Cone Vib
Model XF 10
VELOCITY (IN/SEC.)
LEGEND:
⊙ 24 CPS
△ 40 - 50 CPS
☐ 80 - 100 CPS
◊ 170 CPS
D 250 - 300 CPS
⊖ 500 CPS
Flight: #47
Date: 27 April 1955
Counter Nos: 837 - 865
Survey Location: Sta. -39 at
Tip of Nose Cone
.3
.2
.1
0
1.00
1.02
1.04
1.06
1.08
1.10
1.12
1.14

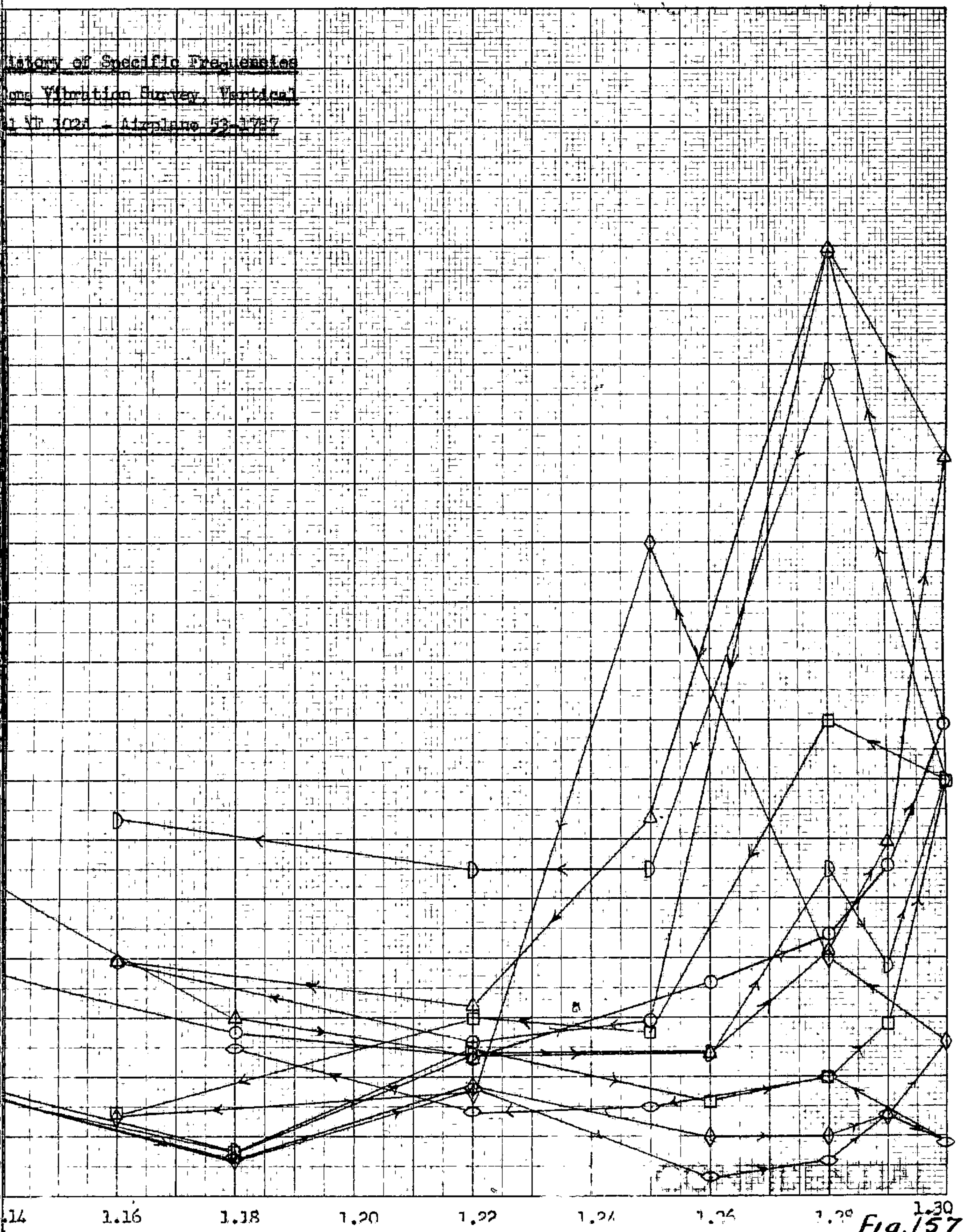
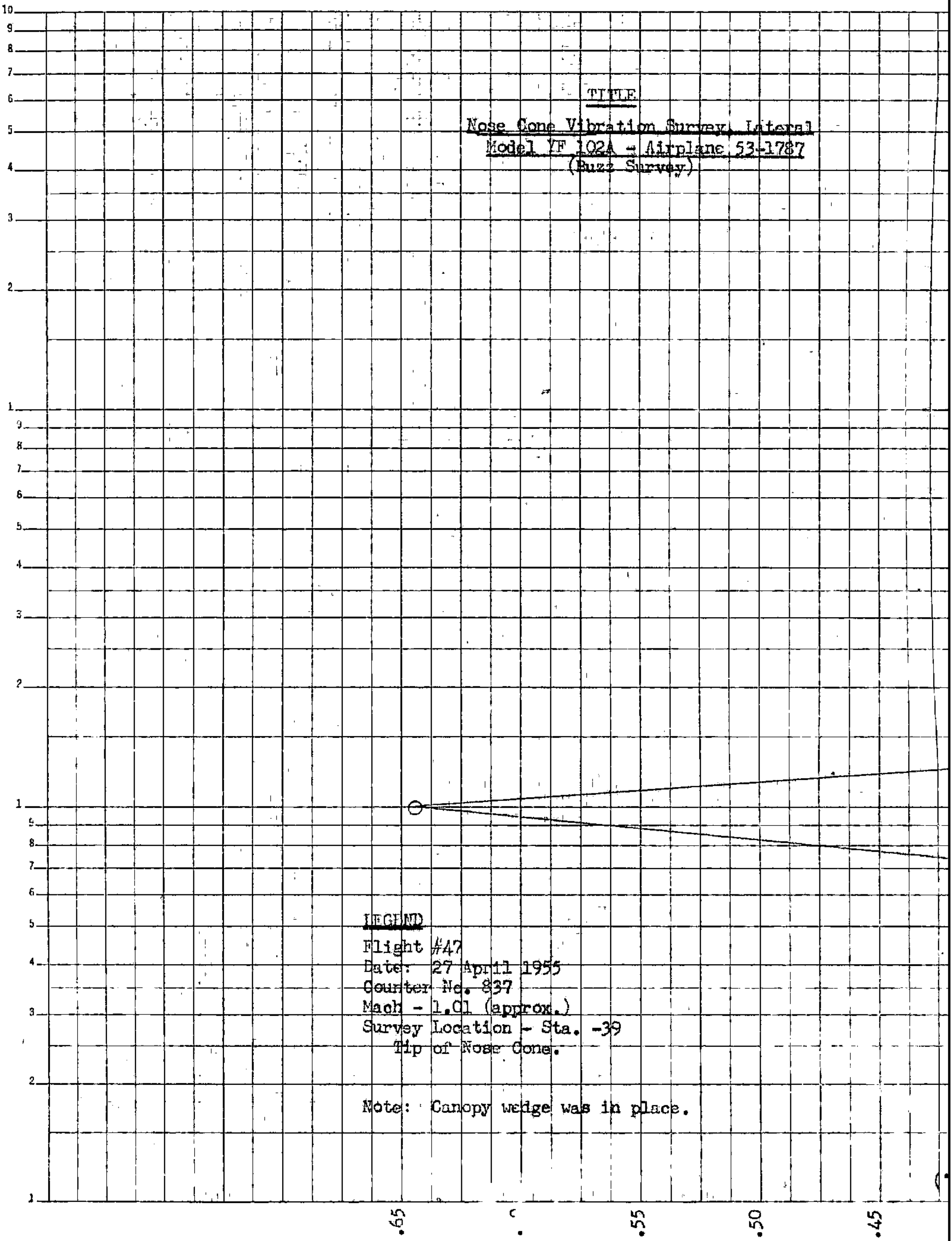

istory of Specific Frequencies
ons Vibration Survey, Vertical
l VF 102A - Airplane 53-1787
.14
1.16
1.18
1.20
1.22
1.24
1.26
1.28
1.30

Fig. 157



TITLE

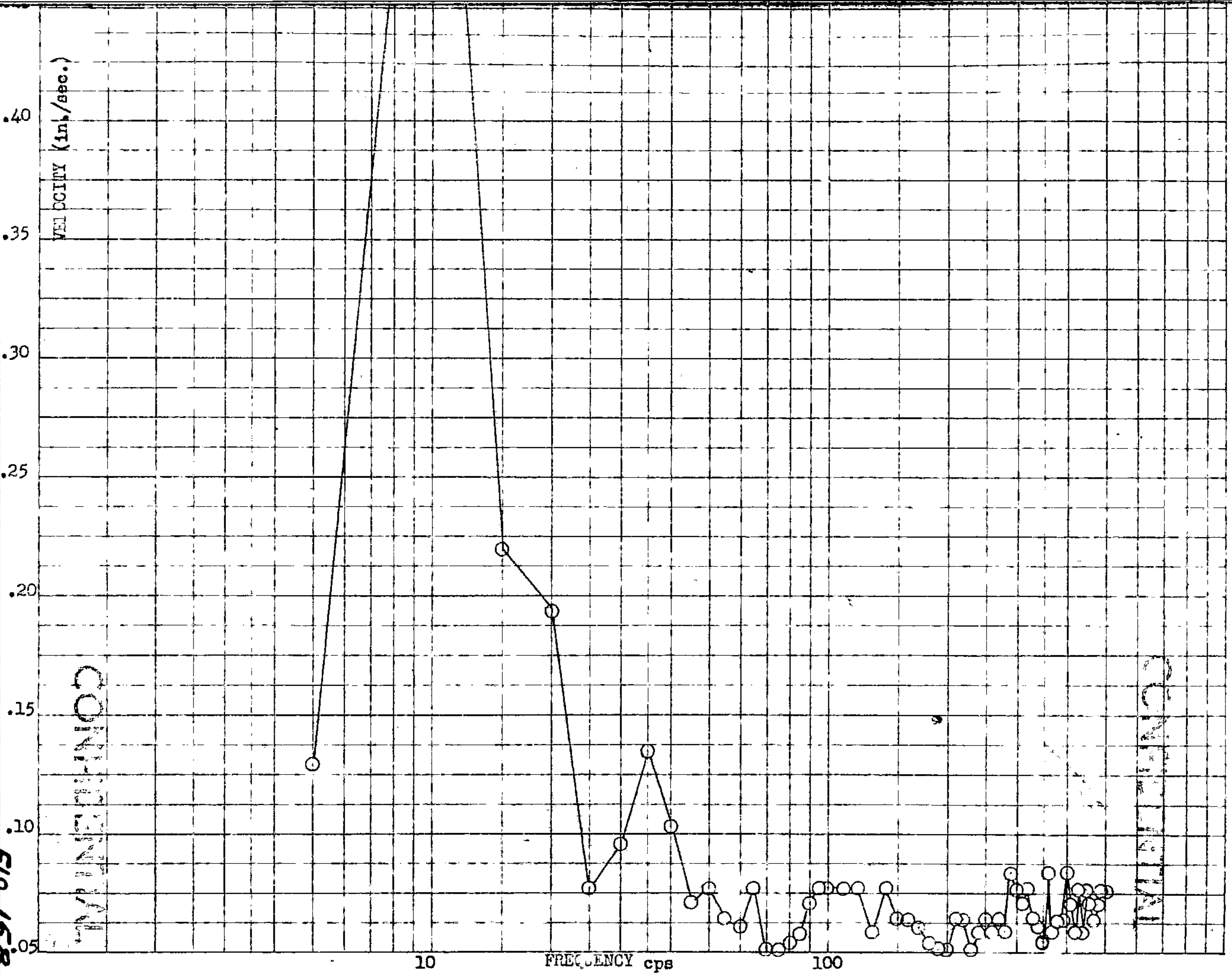
Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND

Flight #47
Date: 27 April 1955
Counter No. 837
Mach - 1.01 (approx.)
Survey Location - Sta. -39
Tip of Nose Cone.

Note: Canopy wedge was in place.

359-73L KEUFFEL & ESSER CO.

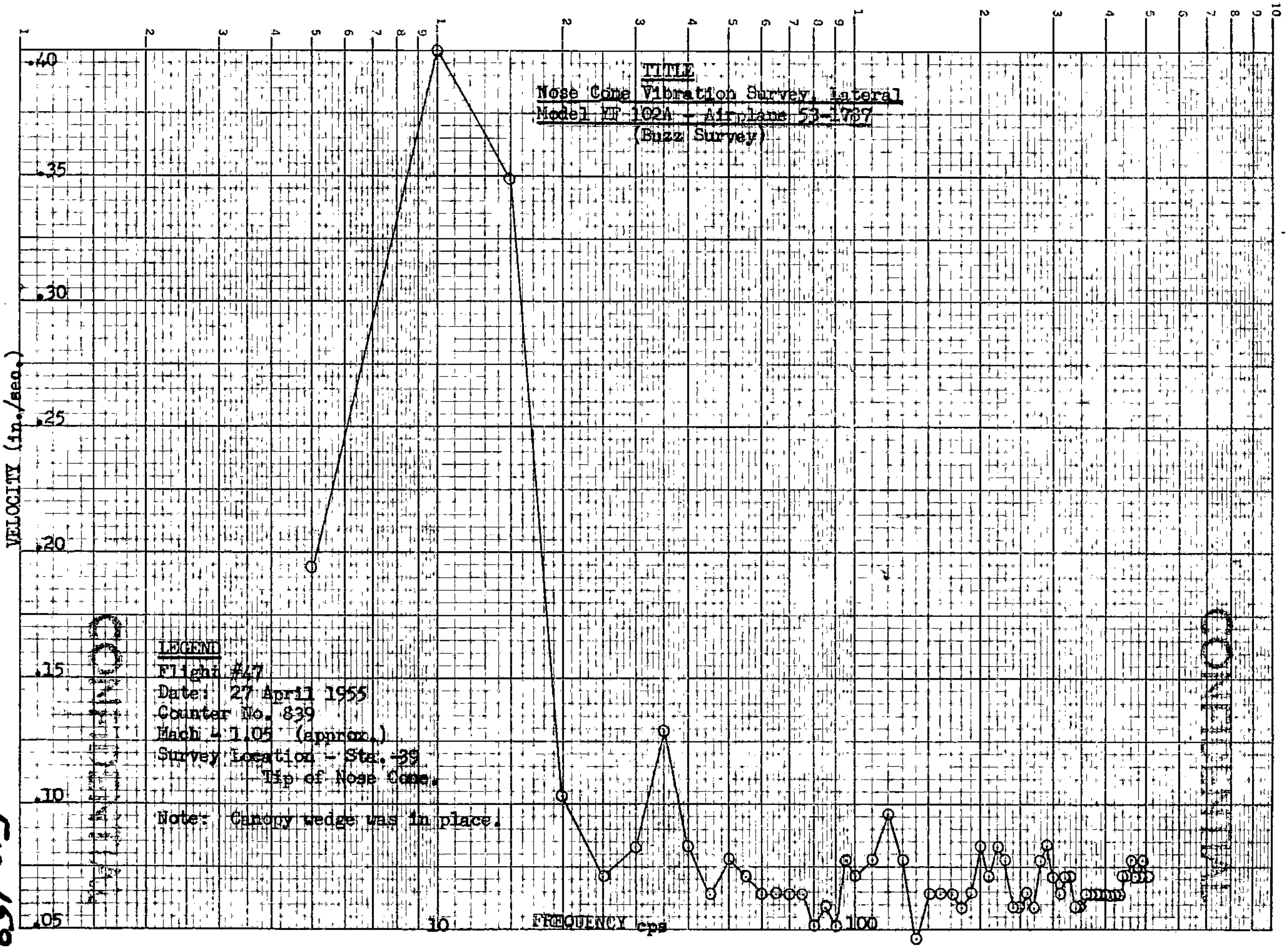


CONFIDENTIAL

CONFIDENTIAL

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Nose Cone Vibration Survey, lateral
Model F-102A - Airplane 53-1787
(Buzz Survey)

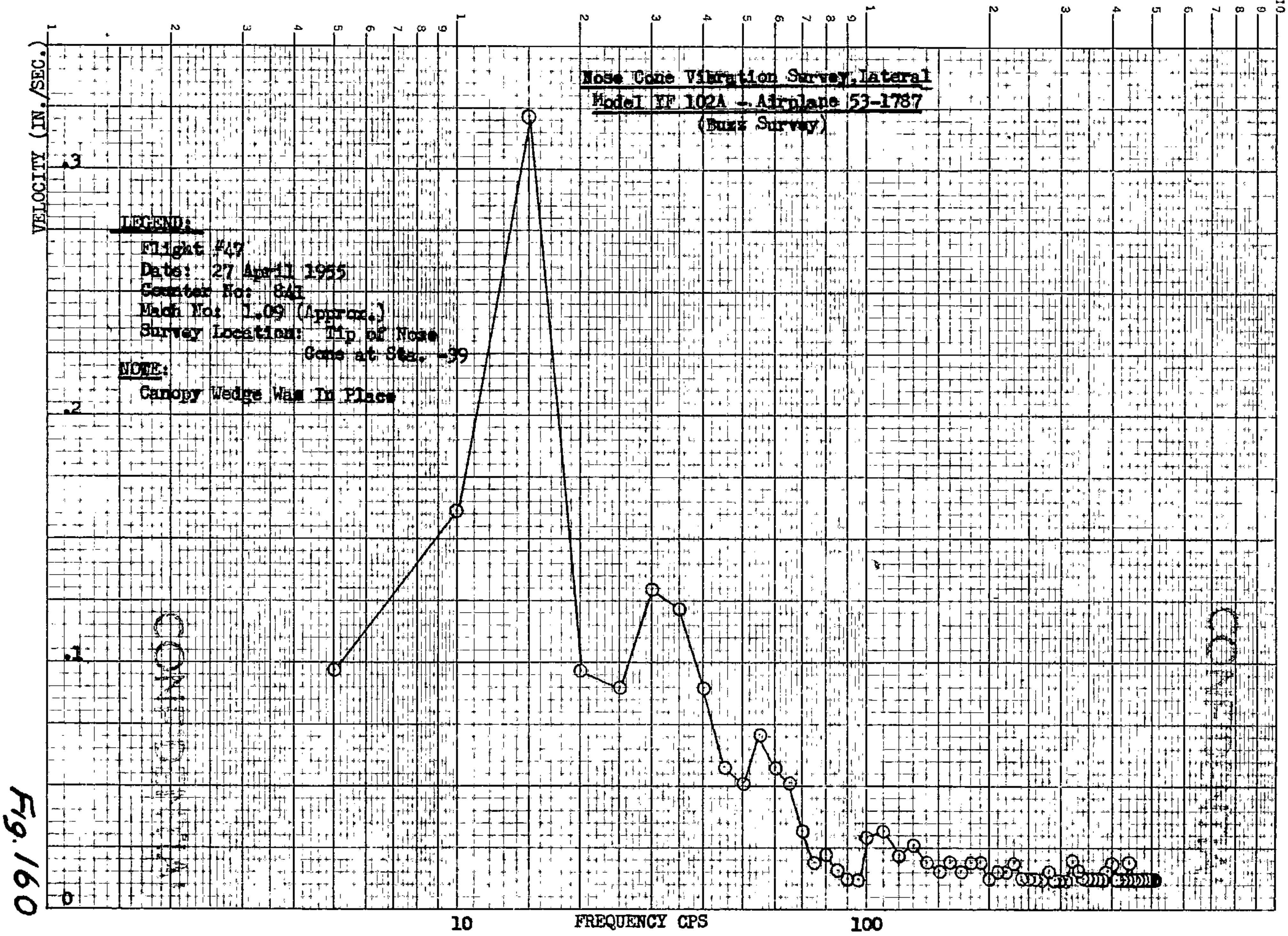
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 839
Mach ≈ 1.05 (approx.)
Survey Location - Sta. 39
Tip of Nose Cone.

Note: Canopy wedge was in place.

CONFIDENTIAL

Fig. 159

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



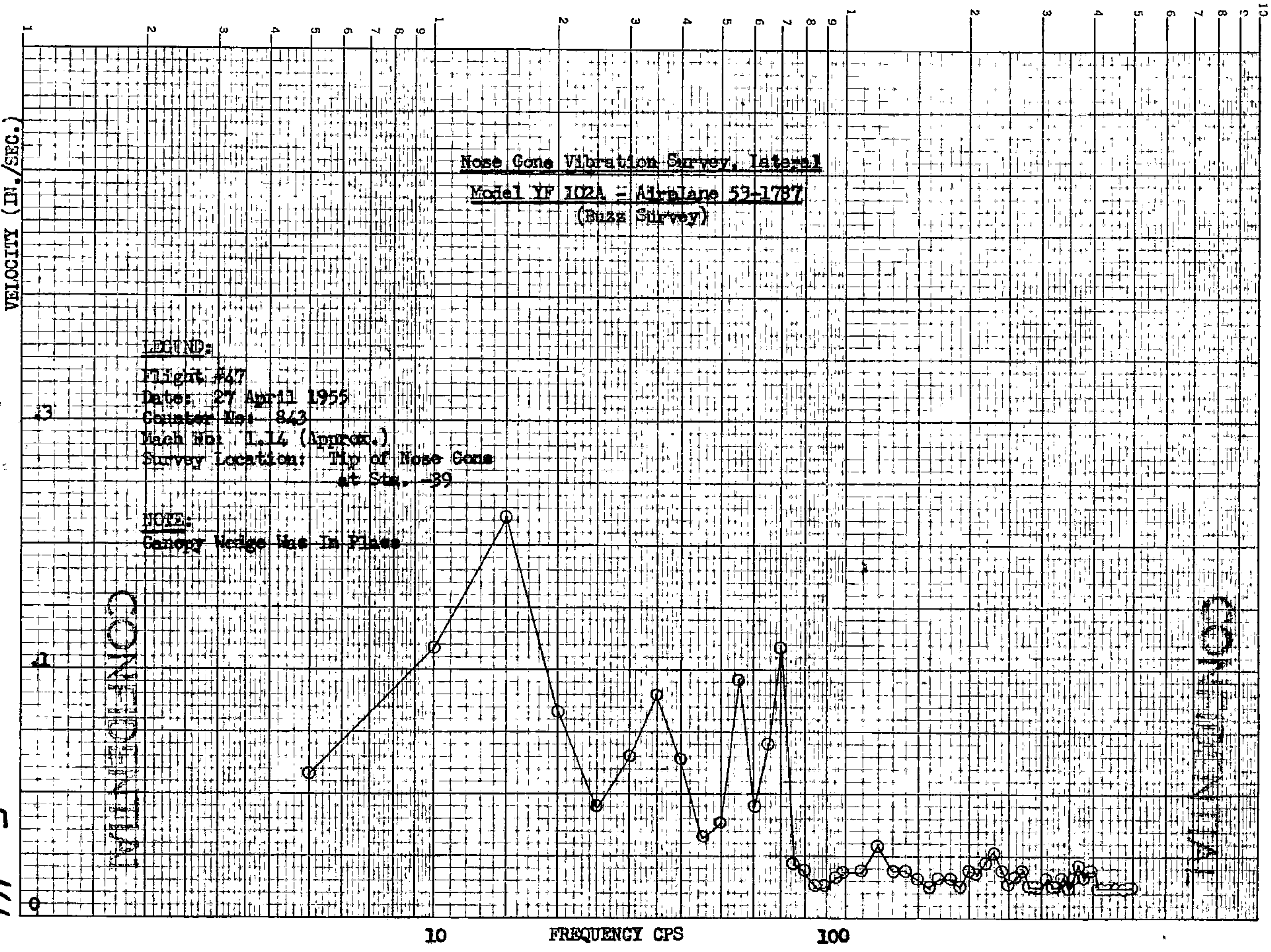
Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 841
Mach No: 1.09 (Approx.)
Survey Location: Tip of Nose
Cone at Sta. -99

NOTE:
Canopy Wedge Was In Place

Fig. 160

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



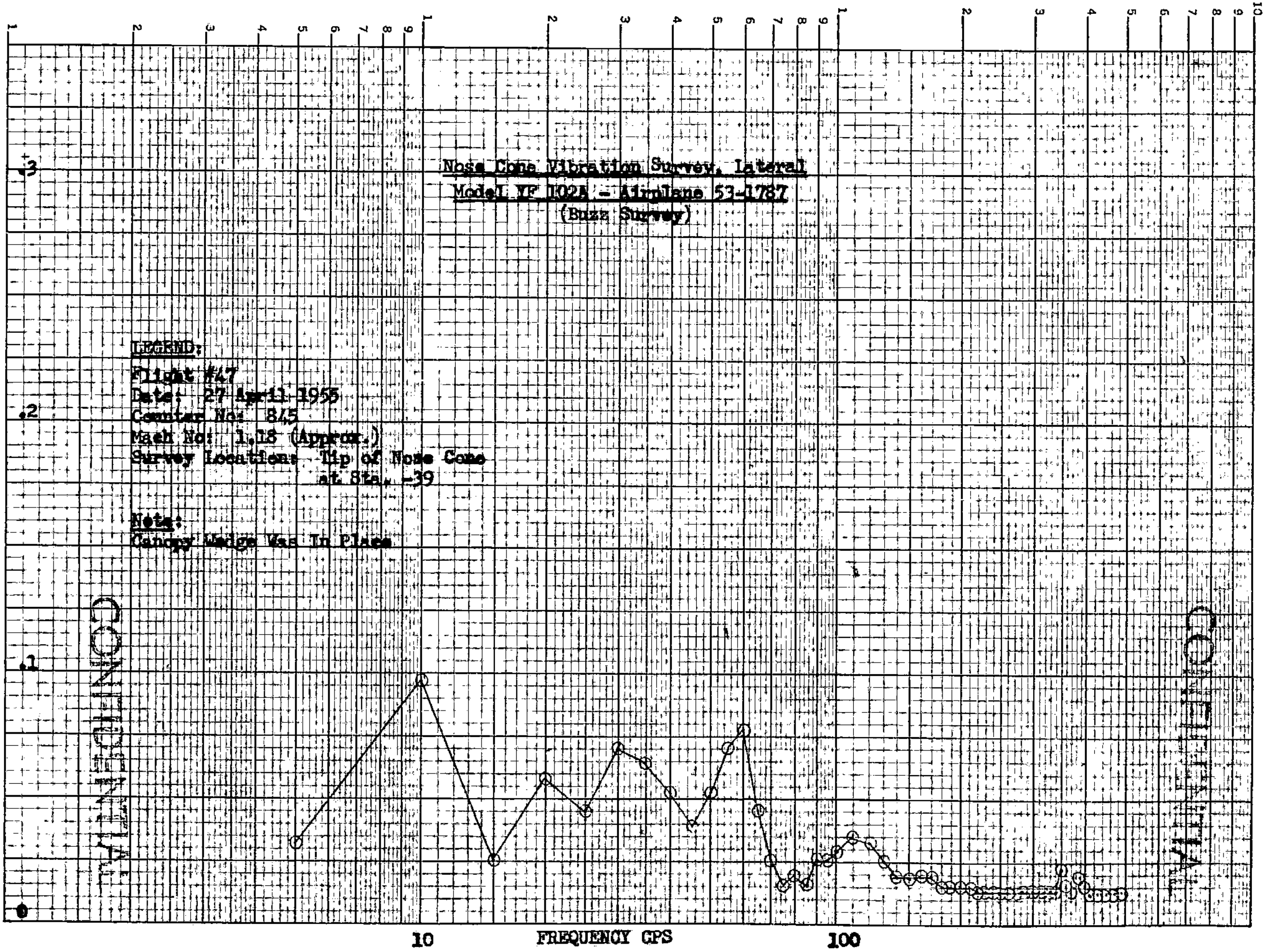
Nose Cone Vibration Survey, Lateral
Model YF 102A – Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 843
Mach No: 1.14 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

NOTE:
Canopy Wedge Was In Place

Fig. 161

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



Nose Cone Vibration Survey, Lateral

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 845
Mach No: 1.18 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

Note:
Canopy Wedge Was In Place

Fig. 162

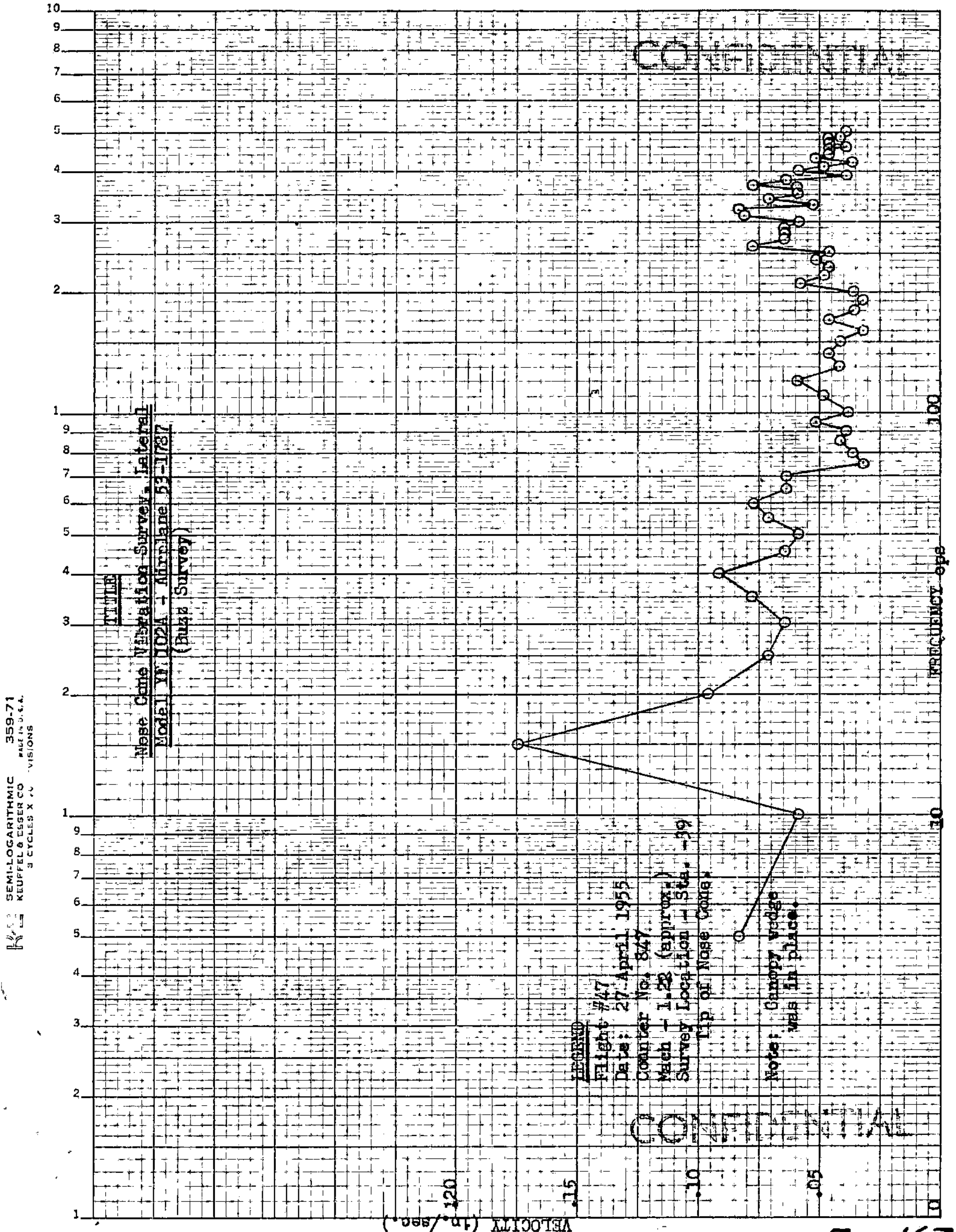
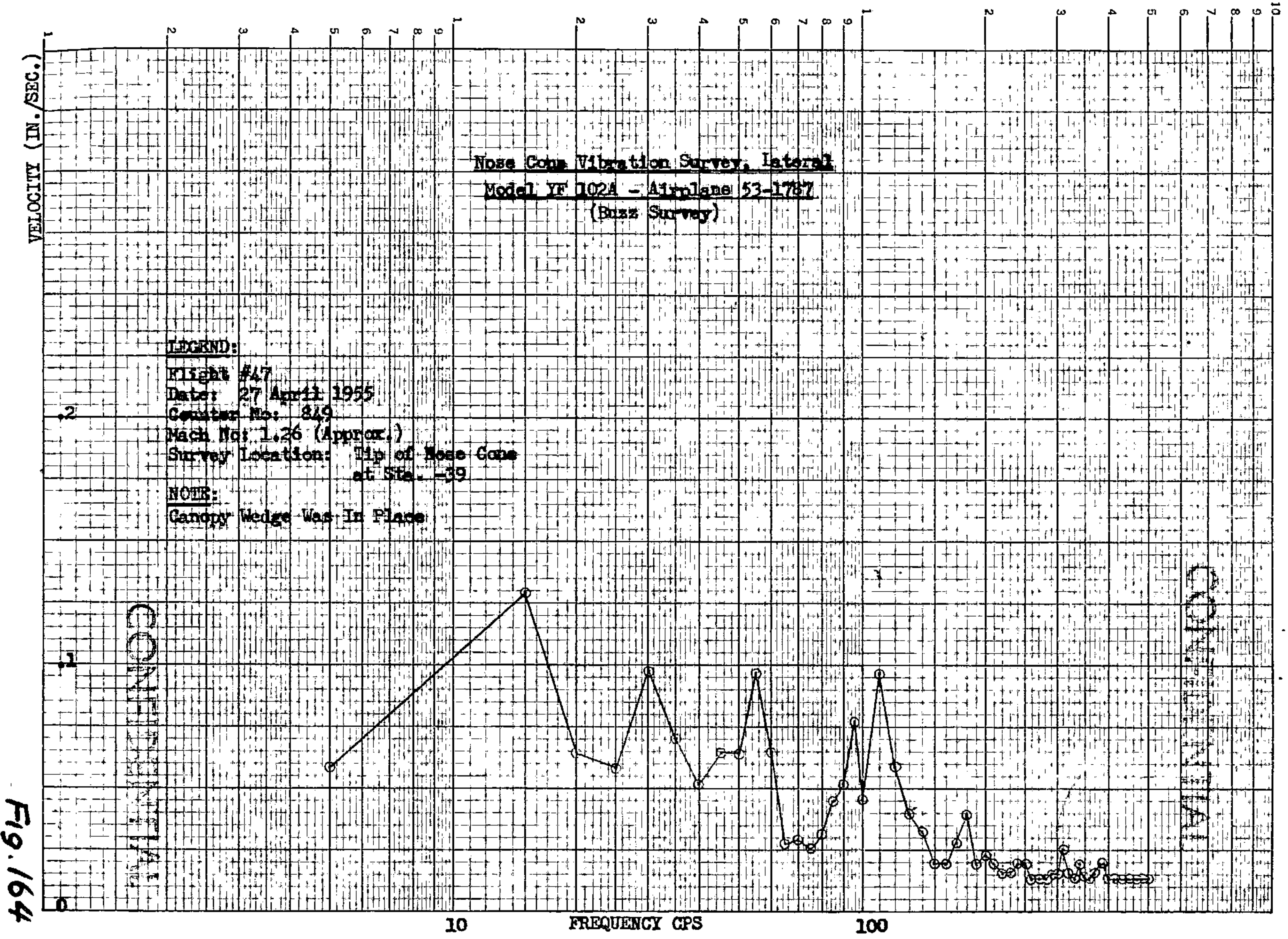

TITLE
Nose Cone Vibration Survey, Lateral
Model YF-102A - Airplane 53-1787
(Buzz Survey)
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 847
Mach - 1.22 (approx.)
Survey Location - Sta. -39
Tip of Nose Cone.
Note: Canopy wedge
was in place.
VELOCITY (in./sec.)
.20
.15
.10
.05
0
FREQUENCY cps
30
100
SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

Fig. 163

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 849
Mach No: 1.26 (Approx.)
Survey Location: Tip of Nose Cone at Sta. -39

NOTE:
Canopy Wedge Was In Place

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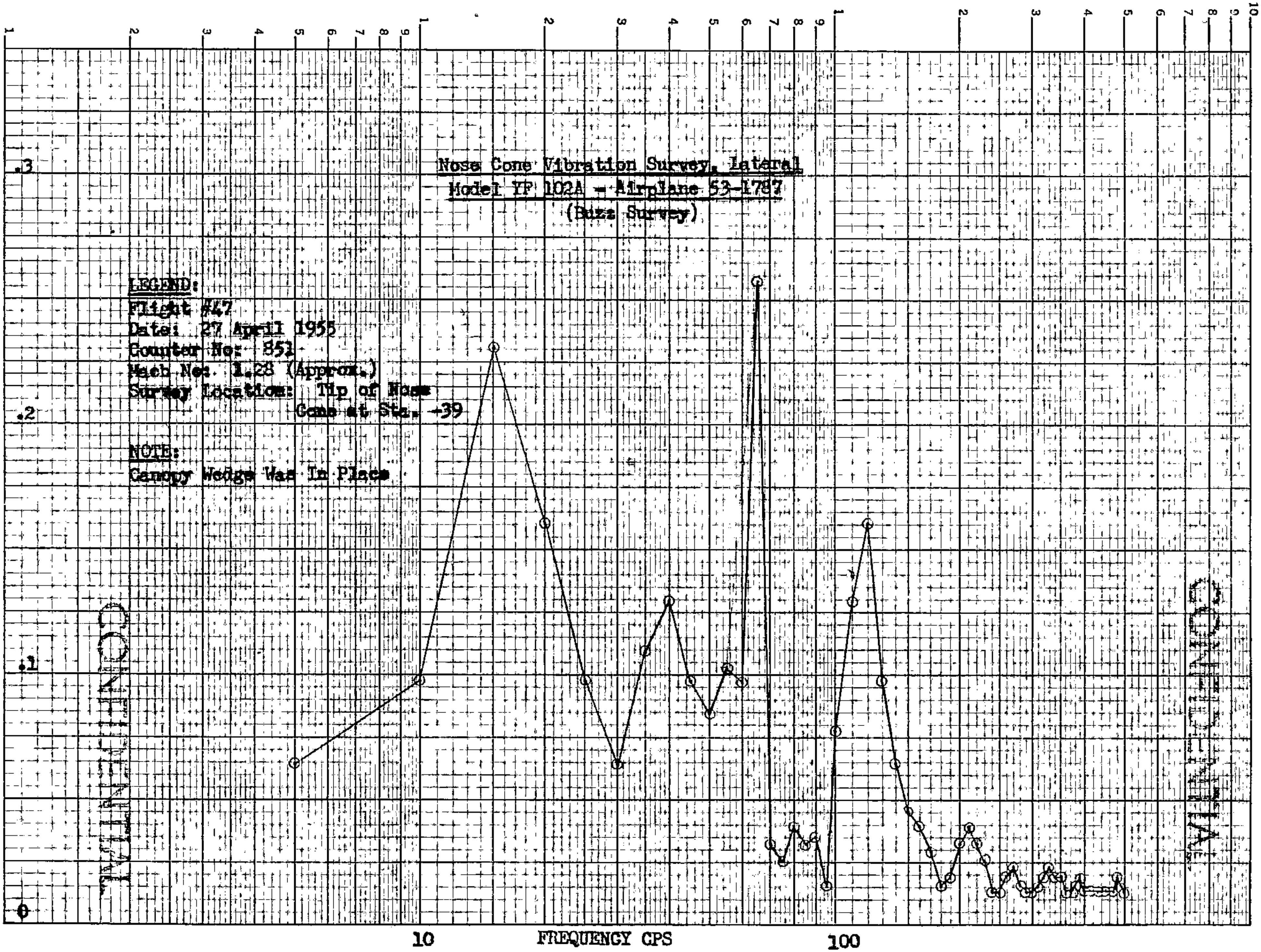
CONFIDENTIAL

Fig. 164

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

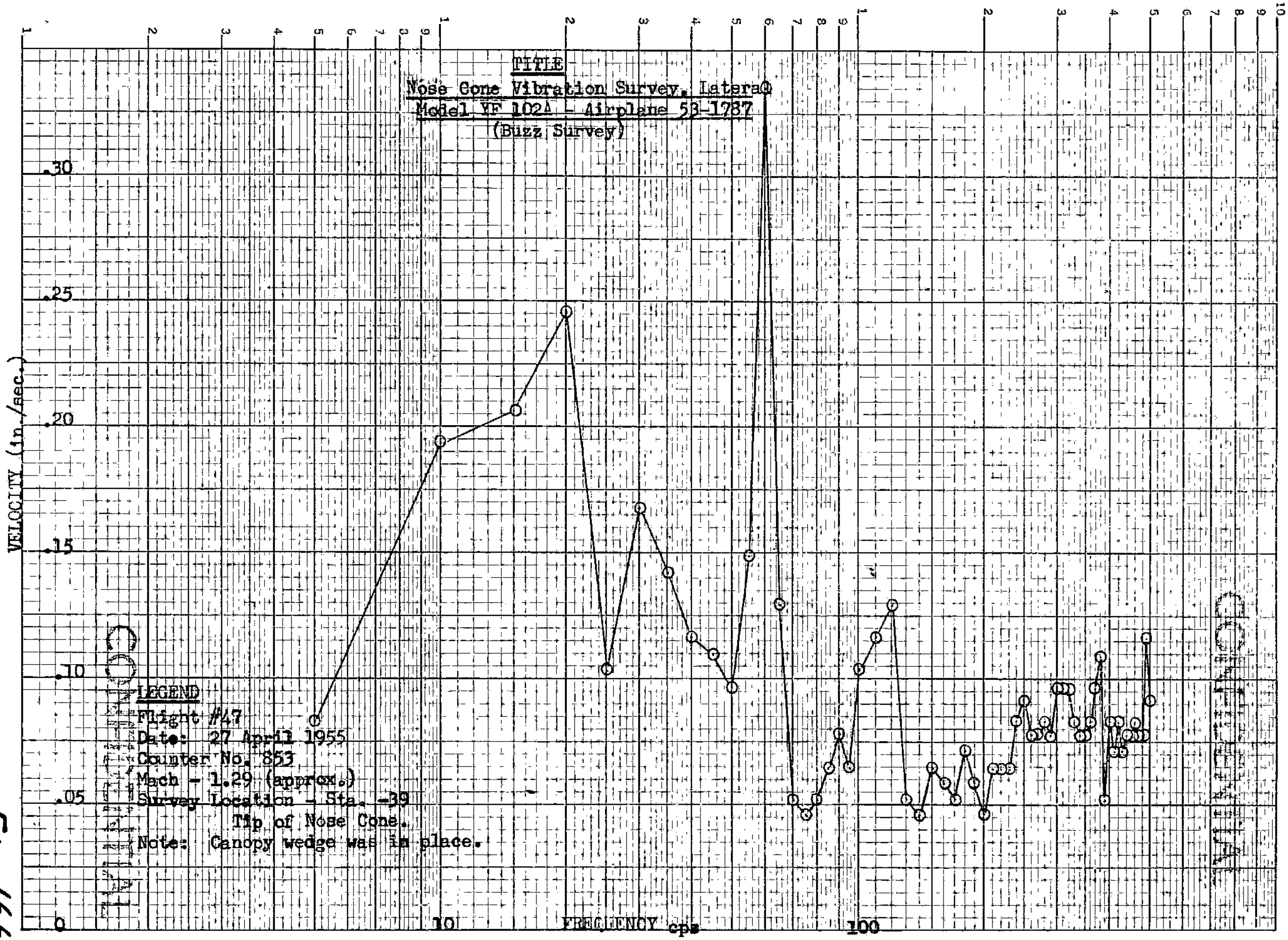
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 851
Mach No: 1.28 (Approx.)
Survey Location: Tip of Nose Cone at Sta. +39

NOTE:
Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 165

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

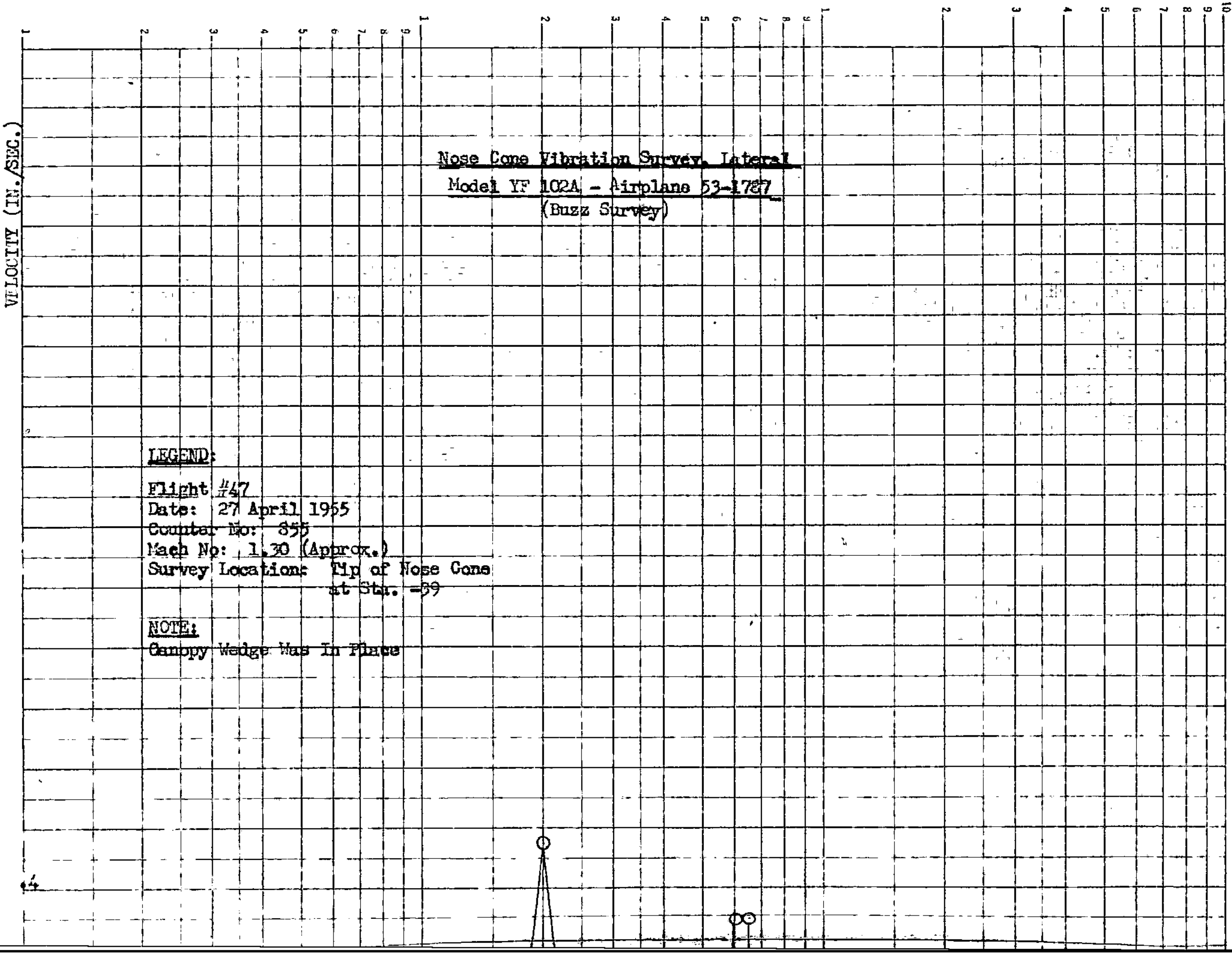


TITLE
Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 853
Mach - 1.29 (approx.)
Survey Location - Sta. -39
Tip of Nose Cone.
Note: Canopy wedge was in place.

Fig. 166

359-73L KEUFFEL & ESSER CO.



Nose Cone Vibration Survey, Lateral

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:

Flight #47
Date: 27 April 1955
Counter No: 855
Mach No: 1.30 (Approx.)
Survey Location: Tip of Nose Cone at Sta. -59

NOTE:

Canopy Wedge Was In Place

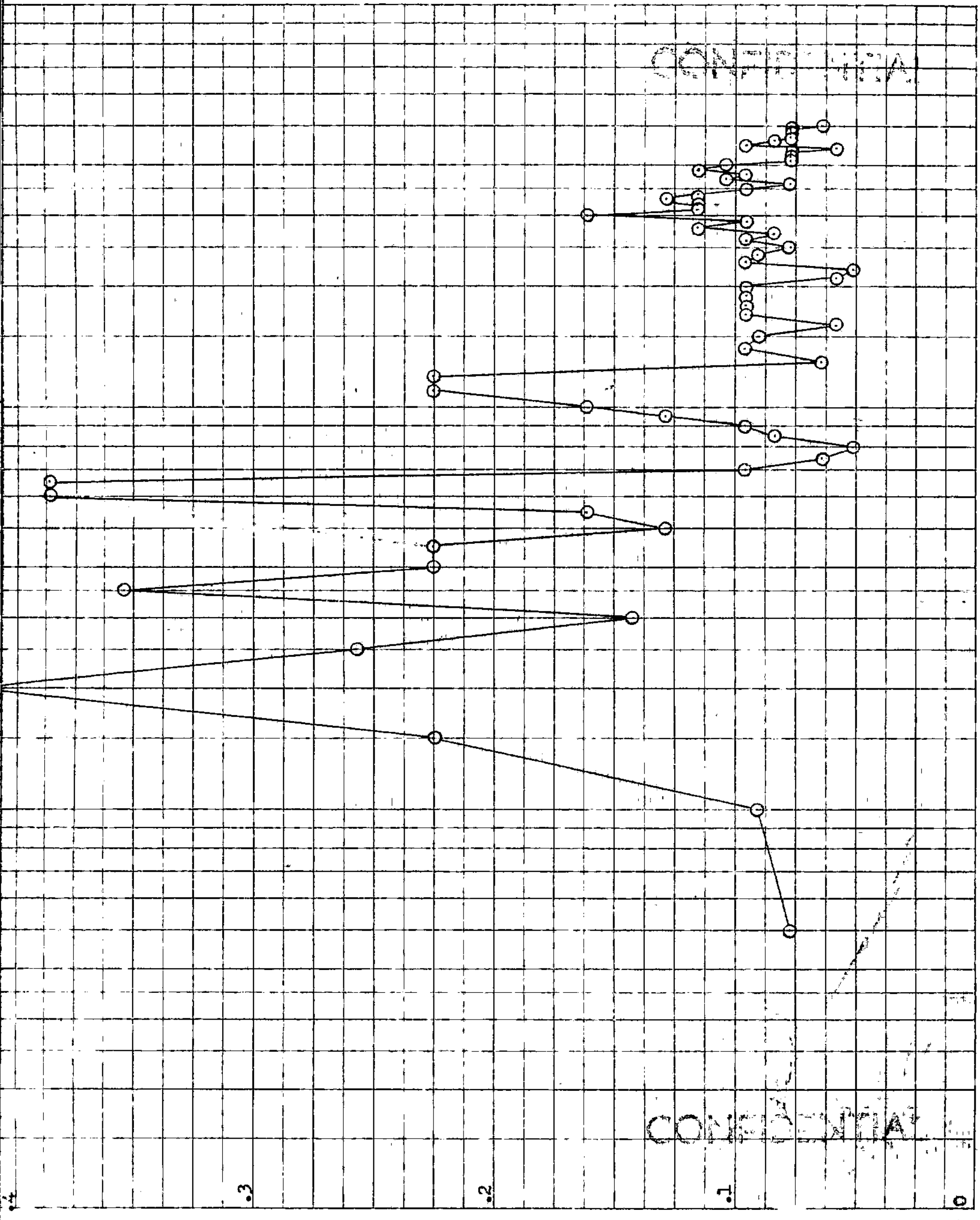
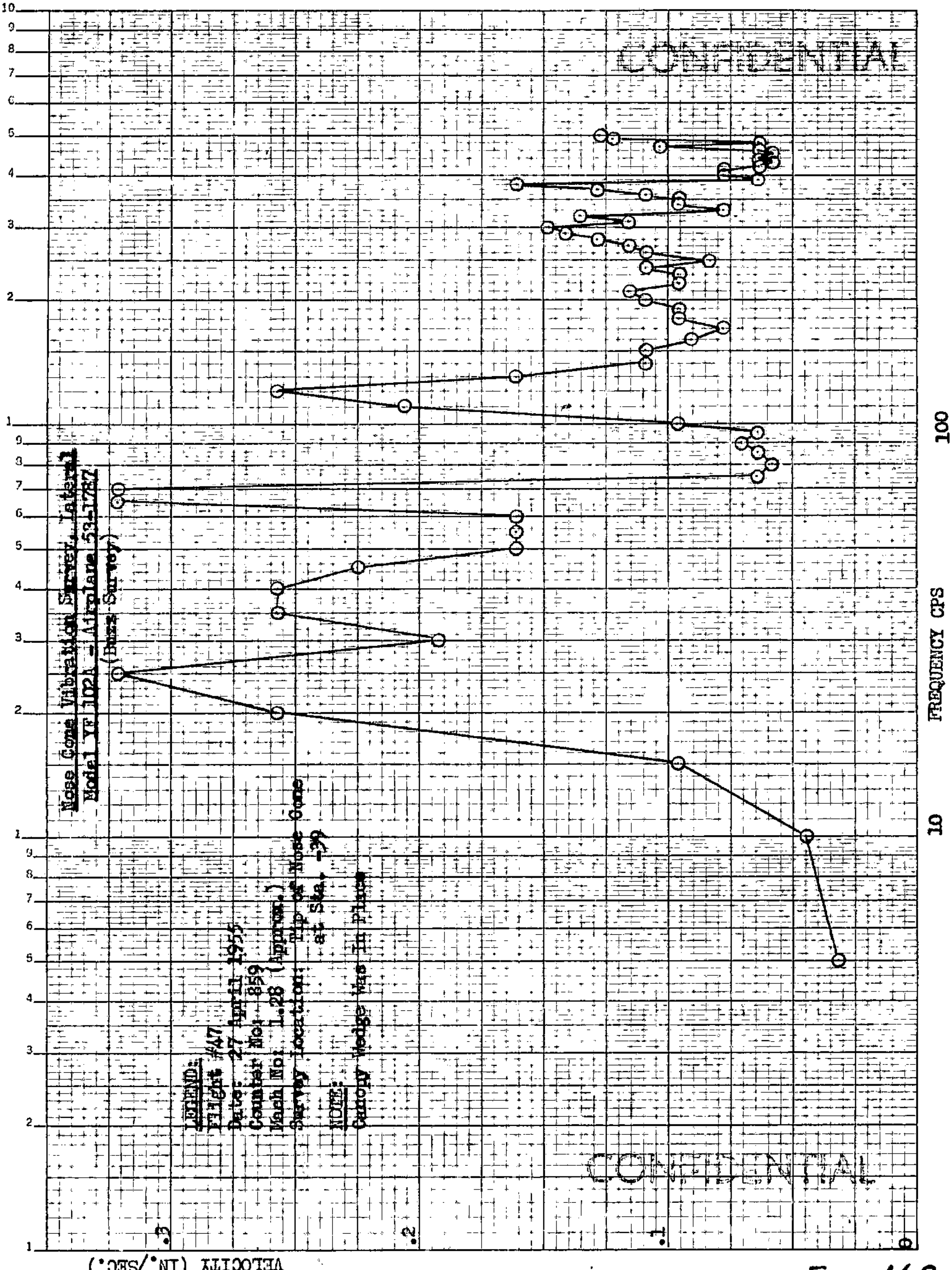


Fig. 167

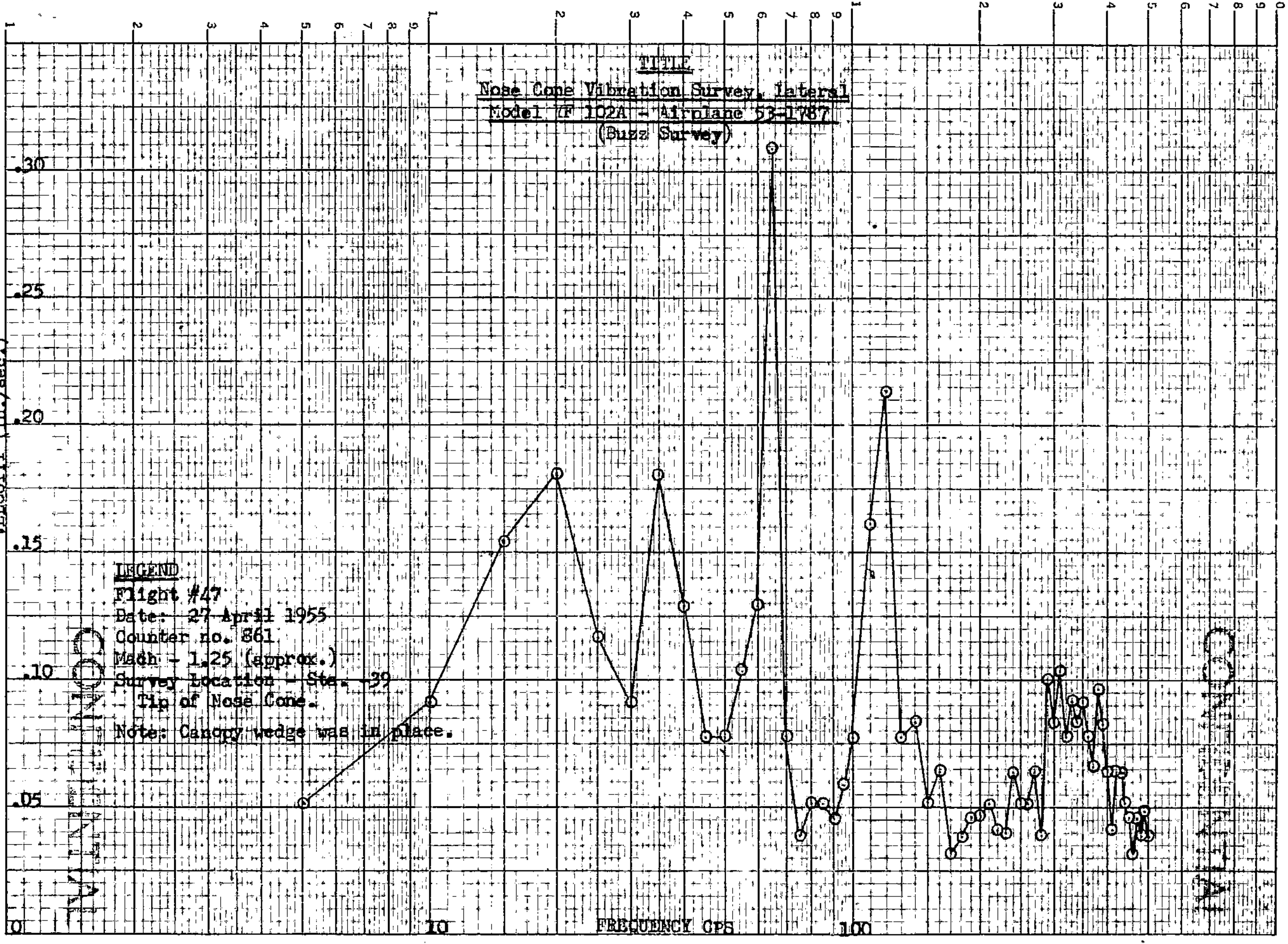


SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

Fig. 168

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

198

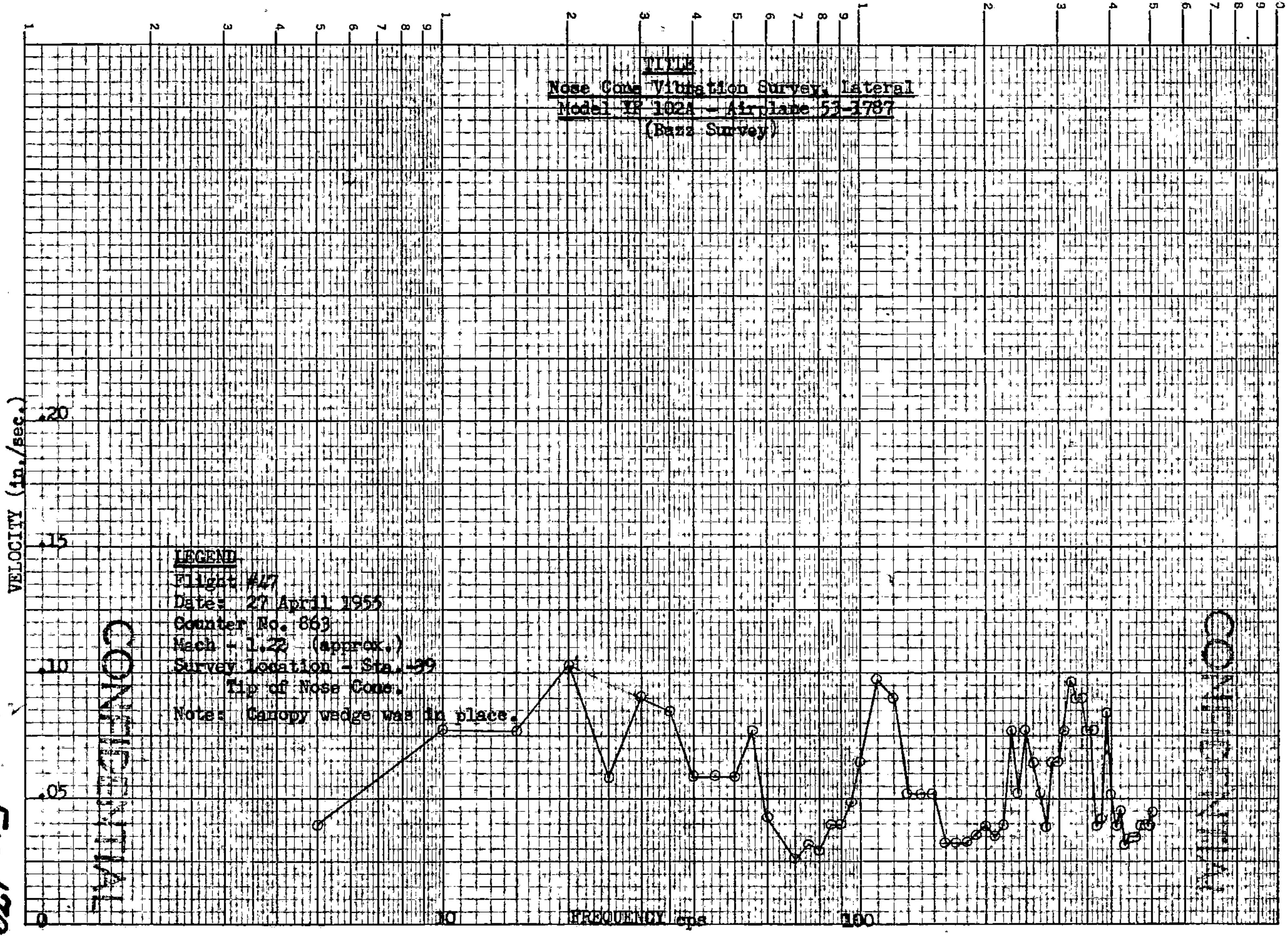


TITLE
Nose Cone Vibration Survey, lateral
Model (F 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter no. 861
Mach - 1.25 (approx.)
Survey Location - Sta. -39
- Tip of Nose Cone.
Note: Canopy wedge was in place.

Fig. 169

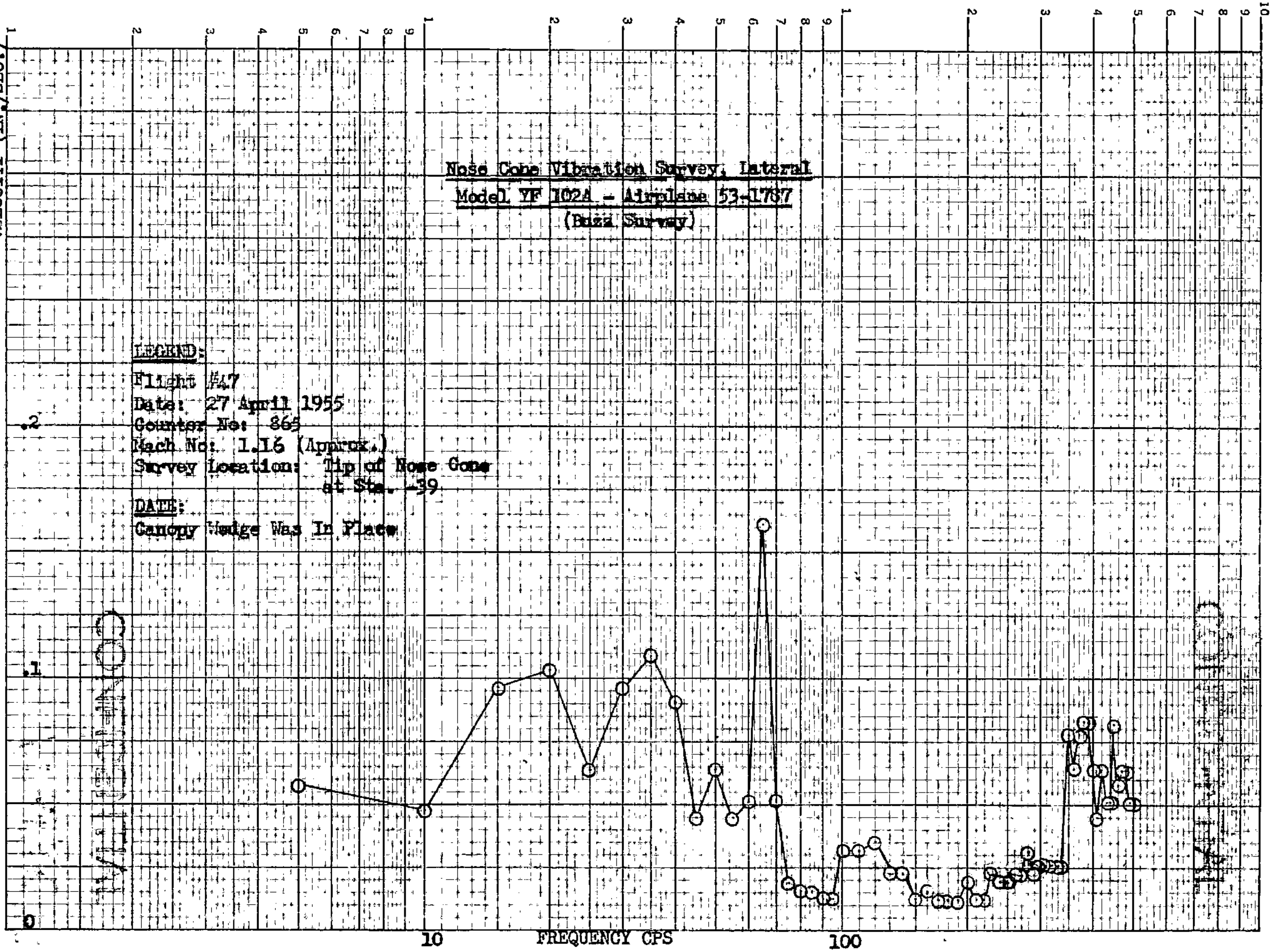
K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



TITLE
Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 863
Mach - 1.22 (approx.)
Survey Location - Sta. -39
Tip of Nose Cone.
Notes: Canopy wedge was in place.

Fig. 170



Nose Cone Vibration Survey, Lateral

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:

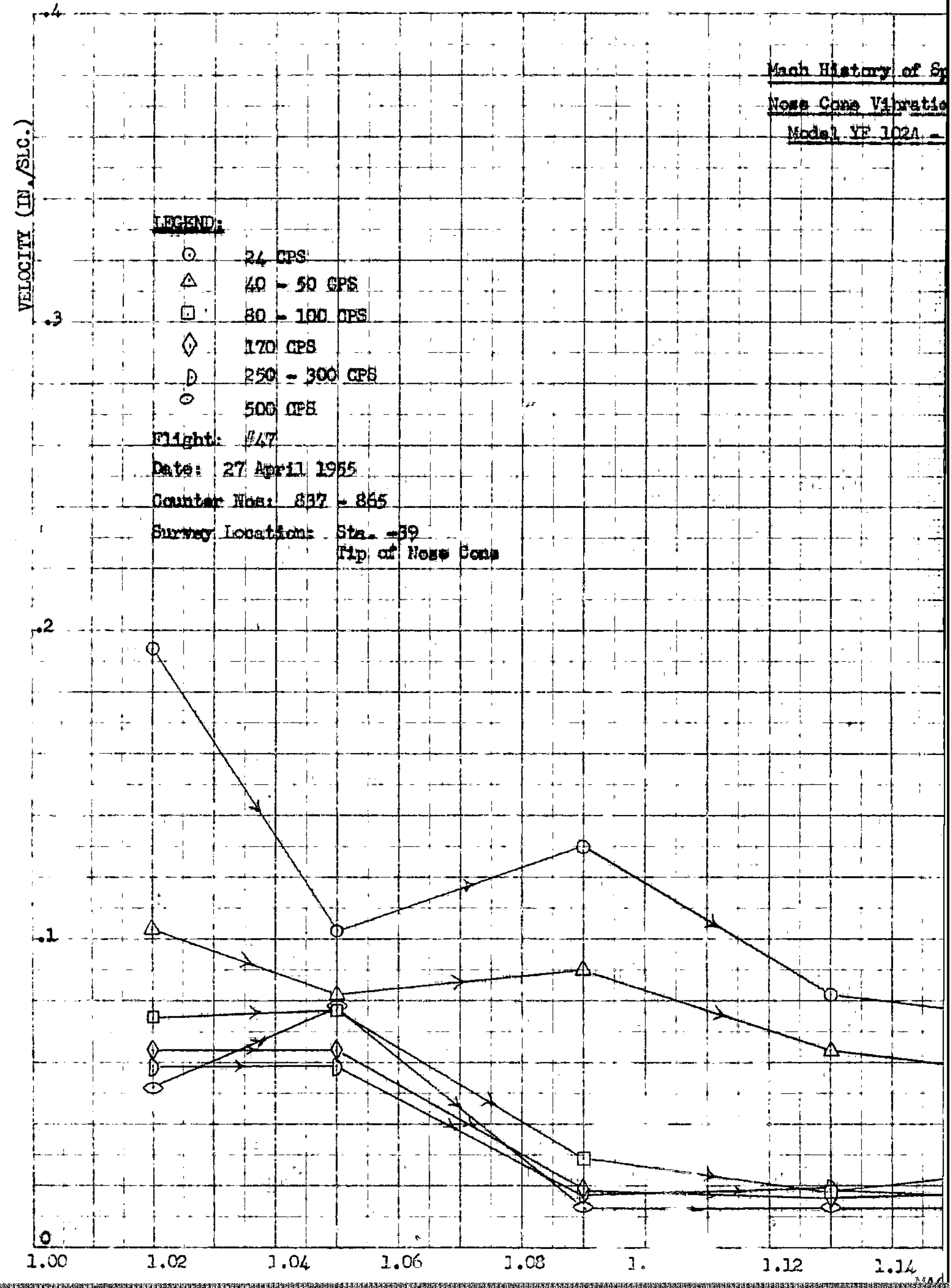
Flight #47
Date: 27 April 1955
Counter No: 865
Mach No: 1.16 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

DATE:

Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 171



Mach History of Sp

Nose Cone Vibratio

Model YF 102A -

Flight: #47

Date: 27 April 1955

Counter Nos: 837 - 865

Survey Location: Sta. -59
Tip of Nose Cone

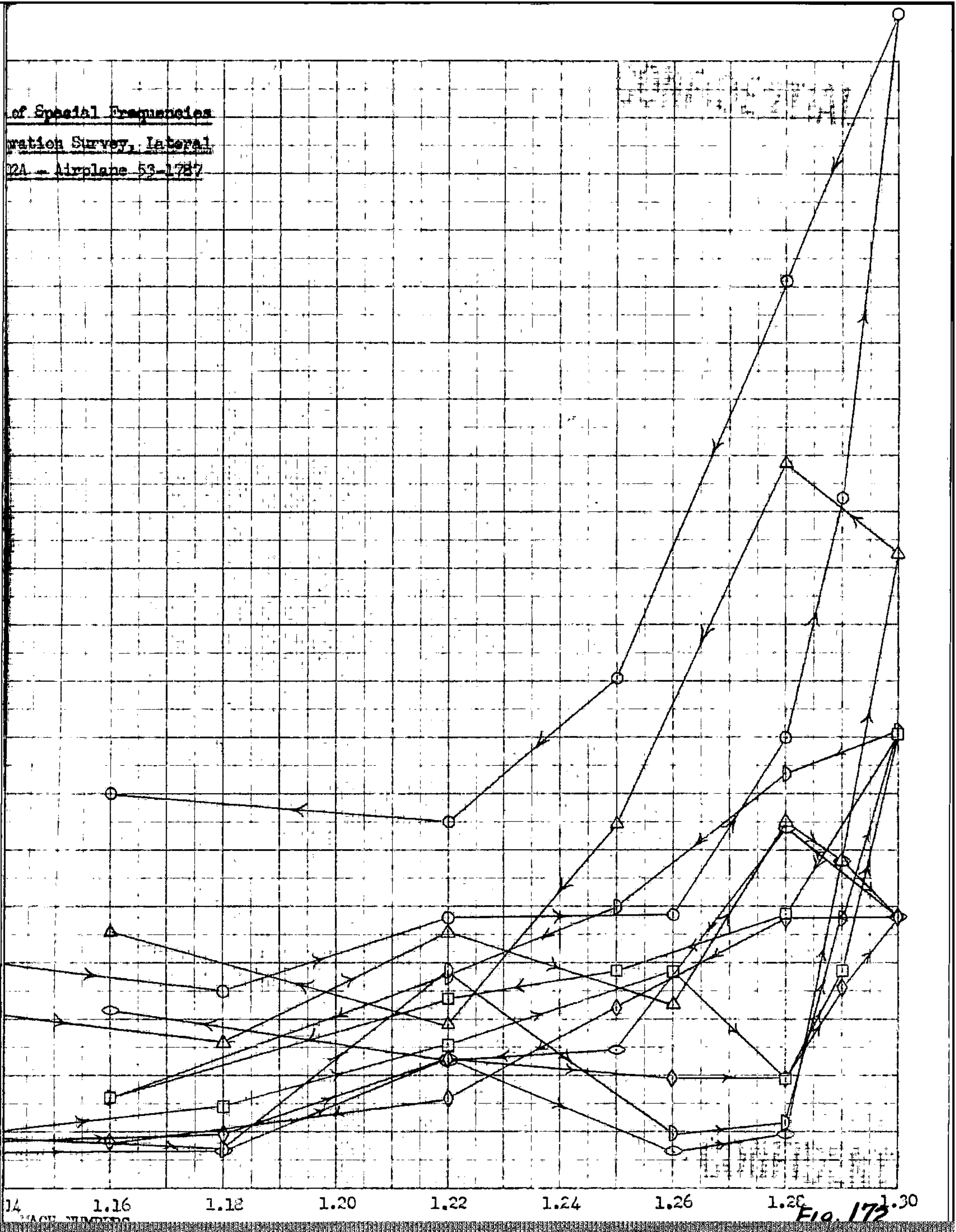
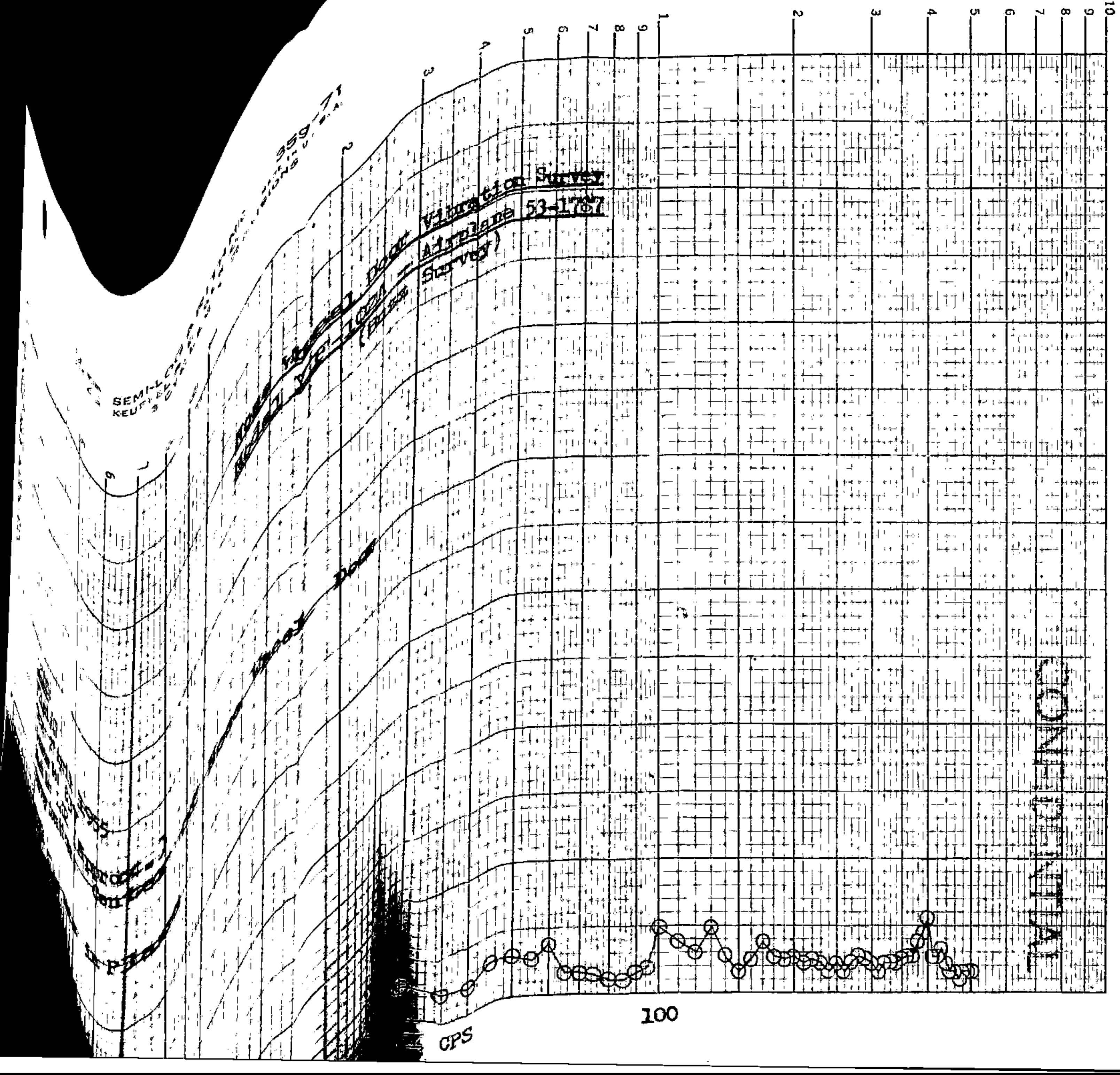

of Special Frequencies
ration Survey, Lateral
02A – Airplane 53-1787
14
1.16
1.18
1.20
1.22
1.24
1.26
1.28
1.30
MACH NUMBERS

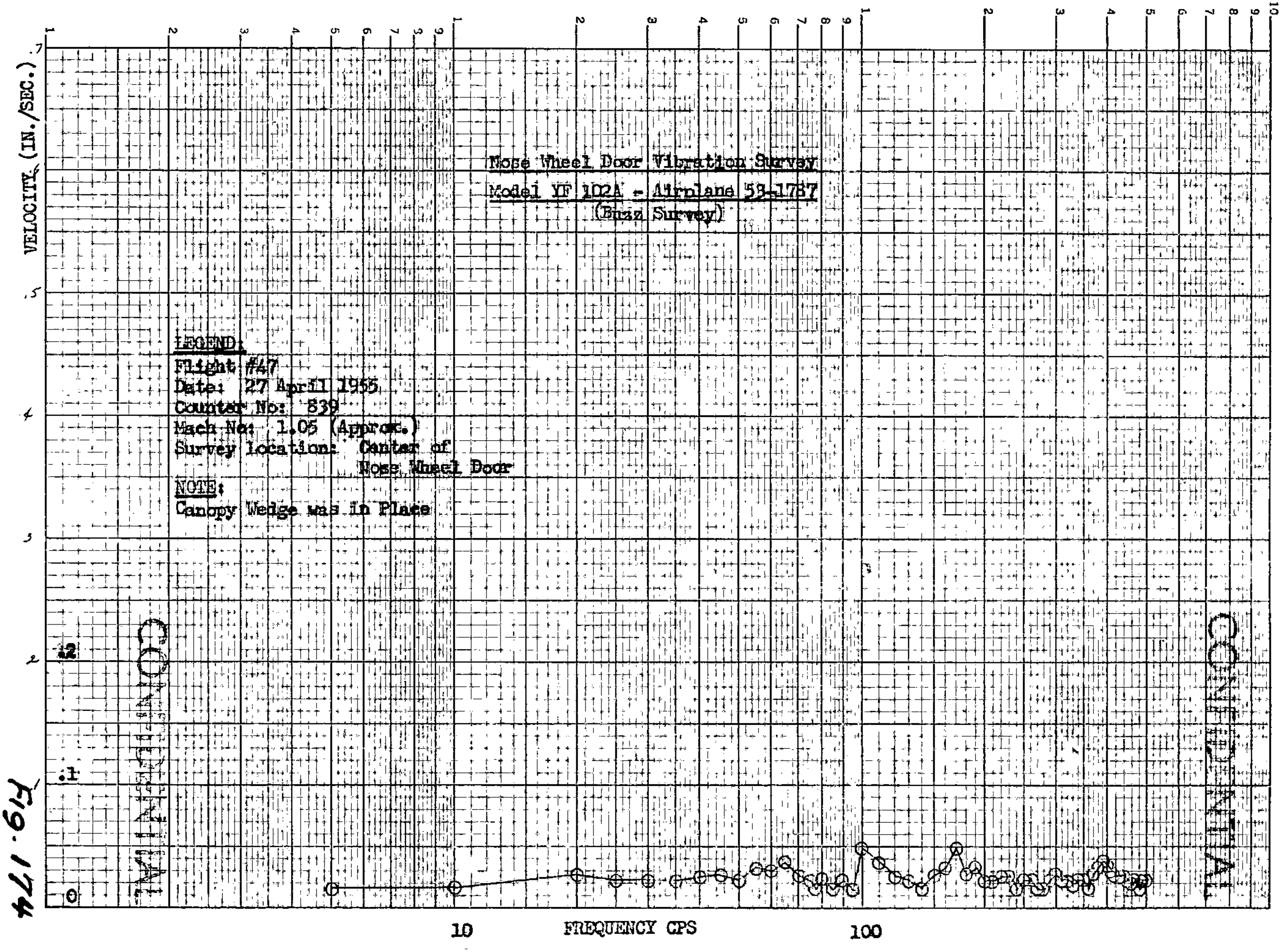
Fig. 172

CONFIDENTIAL



K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Wheel Door Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 839
Mach No: 1.05 (Approx.)
Survey Location: Center of Nose Wheel Door

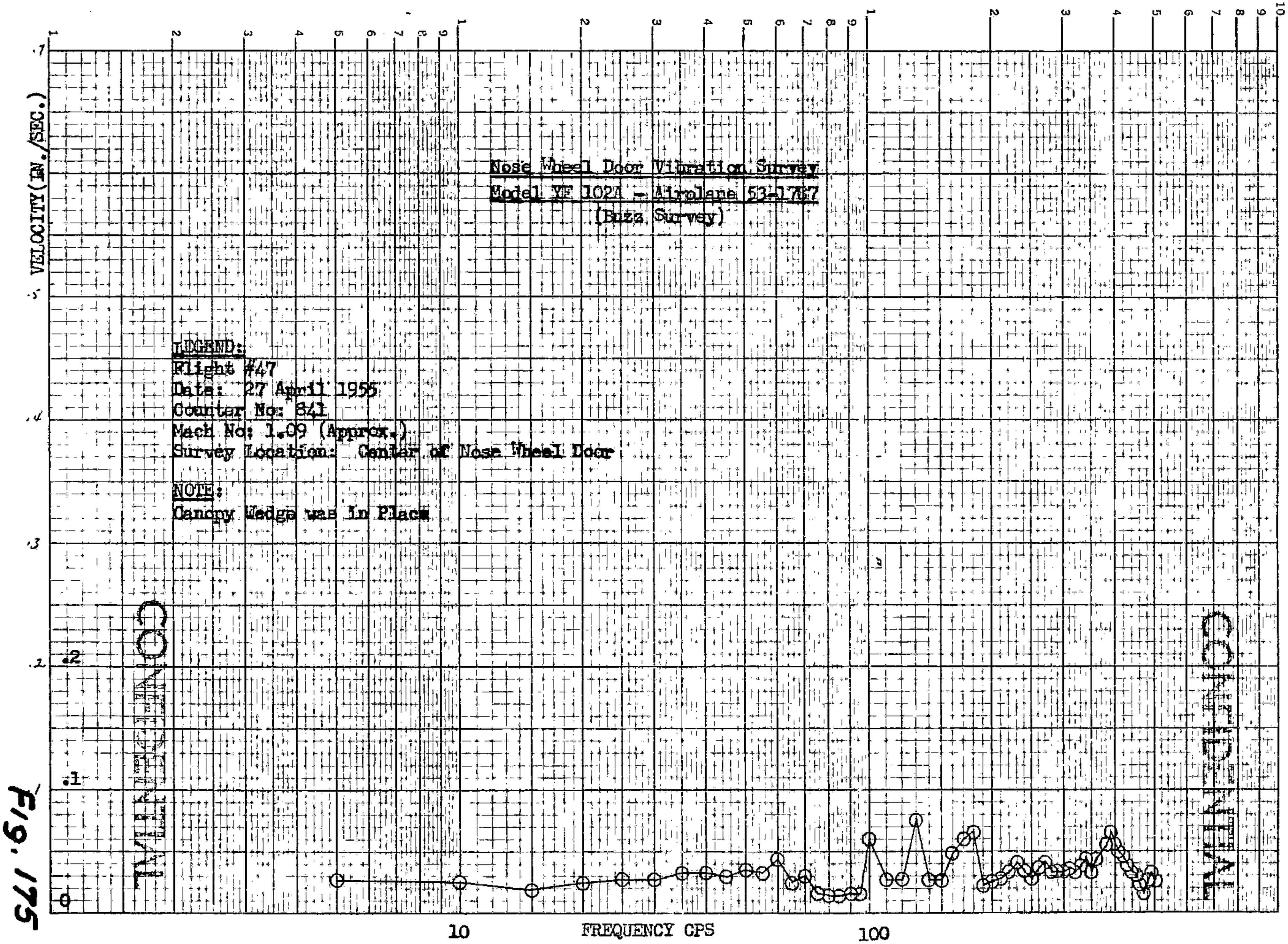
NOTE:
Canopy Wedge was in Place

CONFIDENTIAL

Fig. 174

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Wheel Door Vibration Survey
Model XF 102A - Airplane 53-1787
(Buzz Survey)

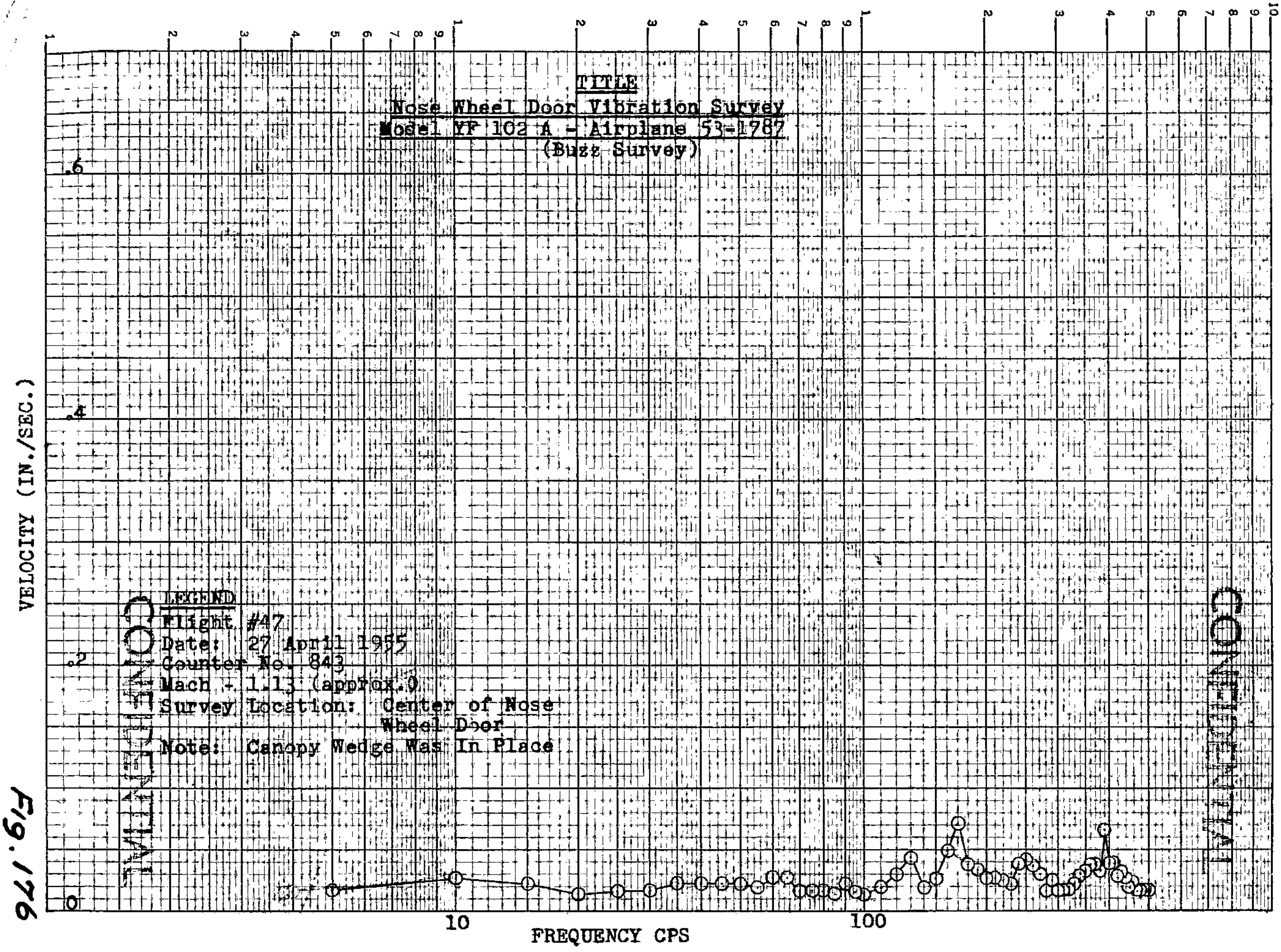
LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 841
Mach No: 1.09 (Approx.)
Survey Location: Center of Nose Wheel Door

NOTE:
Canopy Wedge was In Place

CONFIDENTIAL

Fig. 175

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



TITLE
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 843
Mach - 1.13 (approx.)
Survey Location: Center of Nose
Wheel Door
Note: Canopy Wedge Was In Place

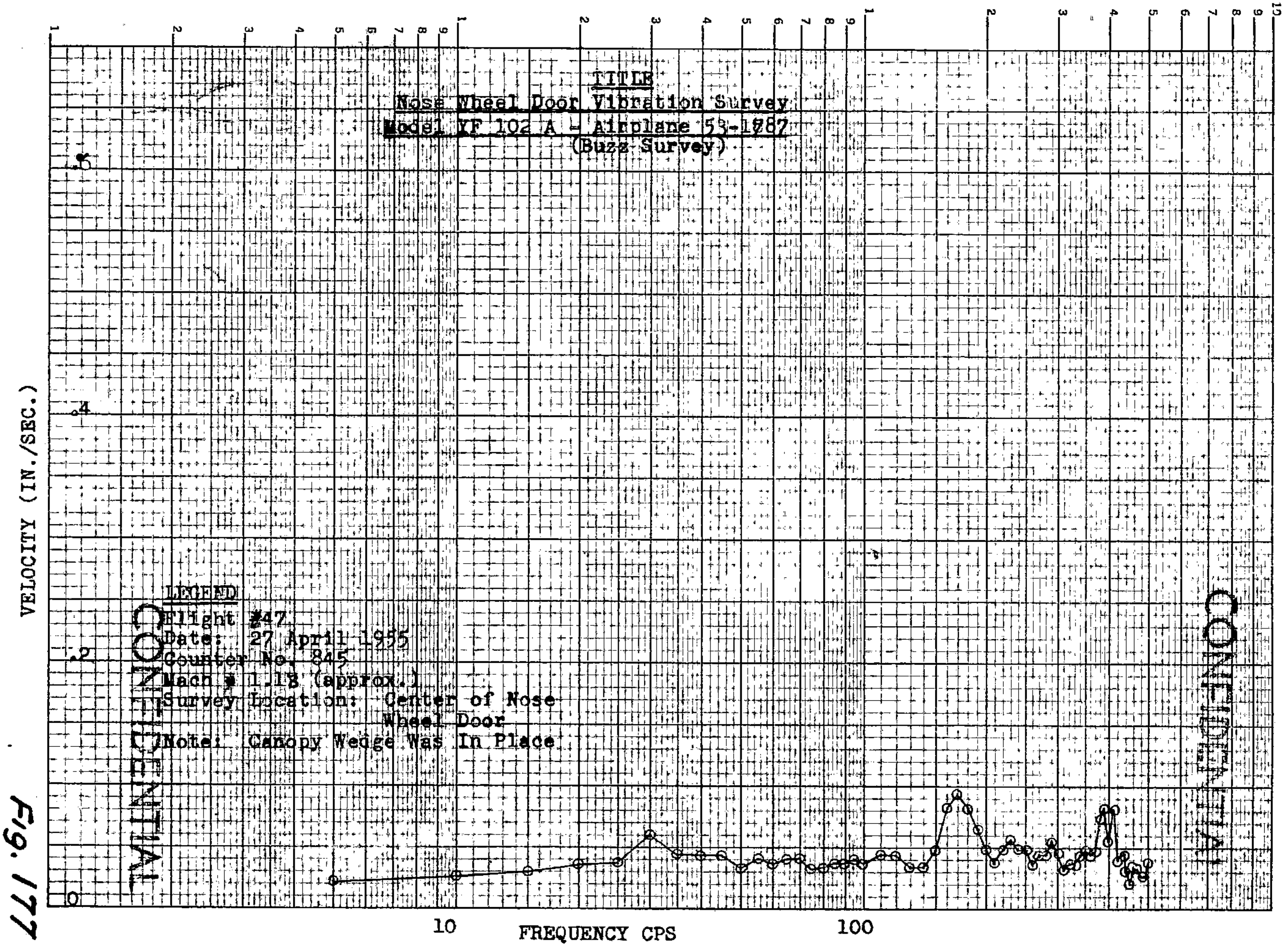
CONFIDENTIAL

CONFIDENTIAL

Fig. 176

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 845
Mach # 1.18 (approx.)
Survey Location: Center of Nose Wheel Door
Note: Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 177

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

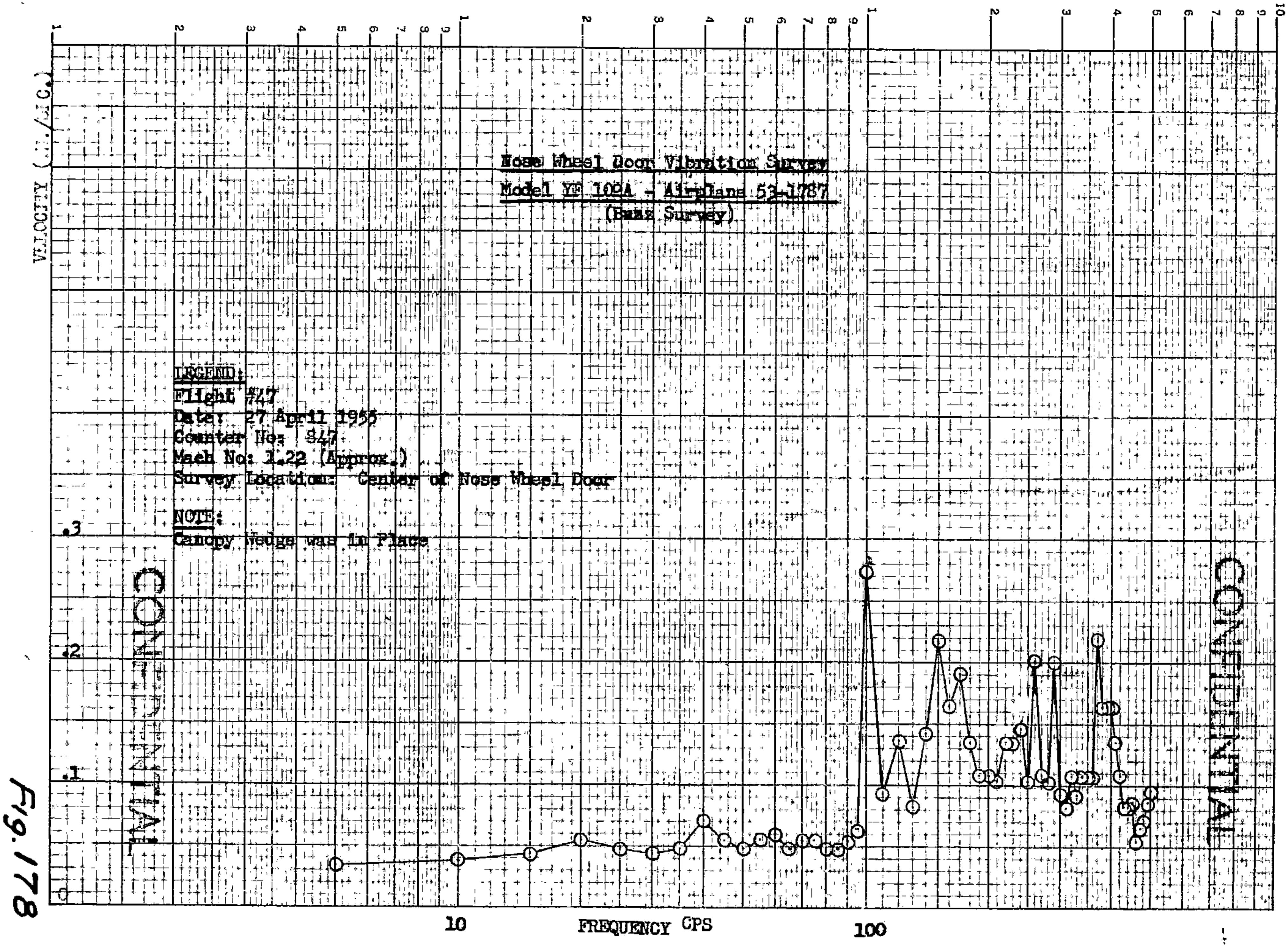
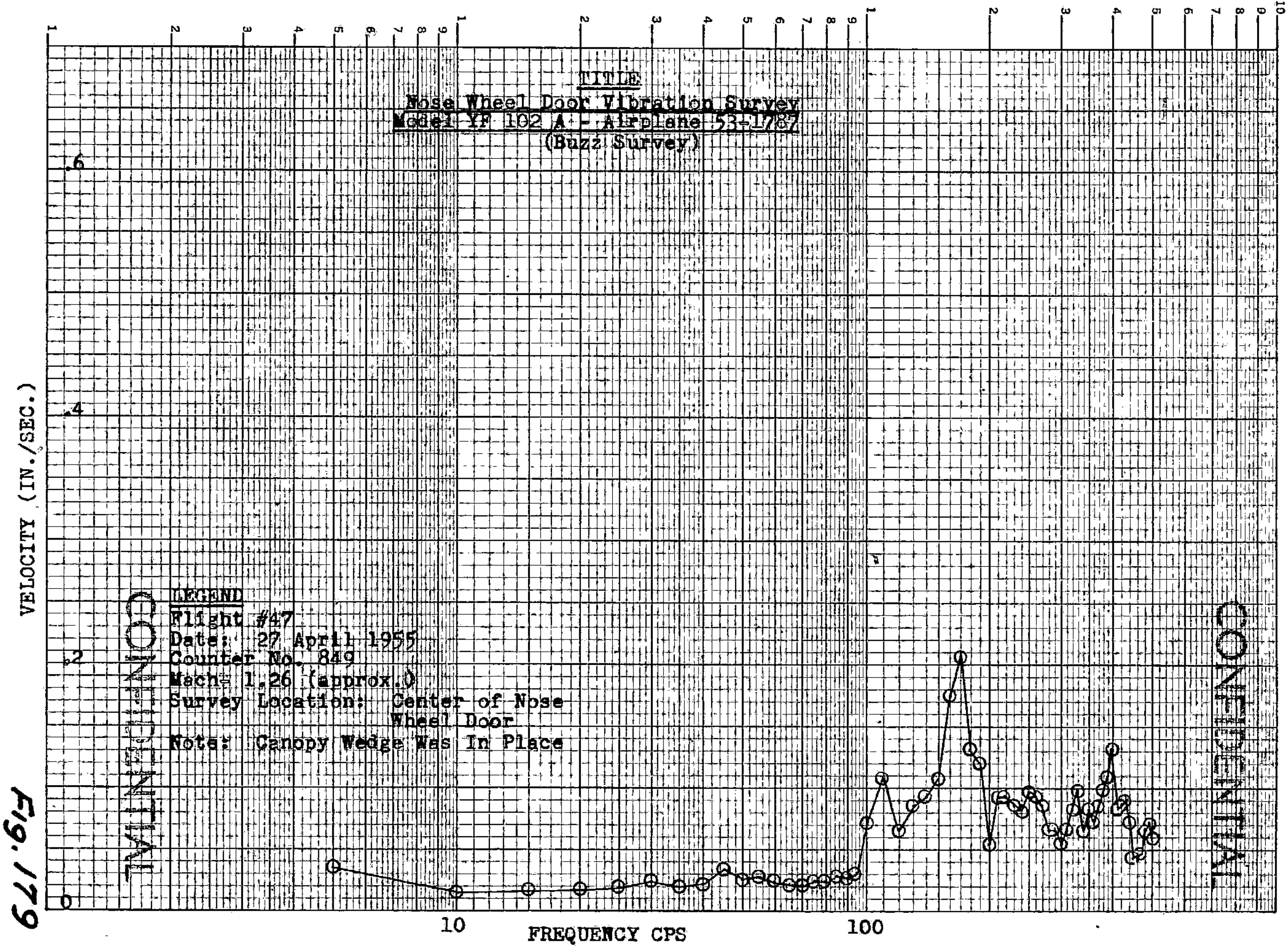


Fig. 178

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



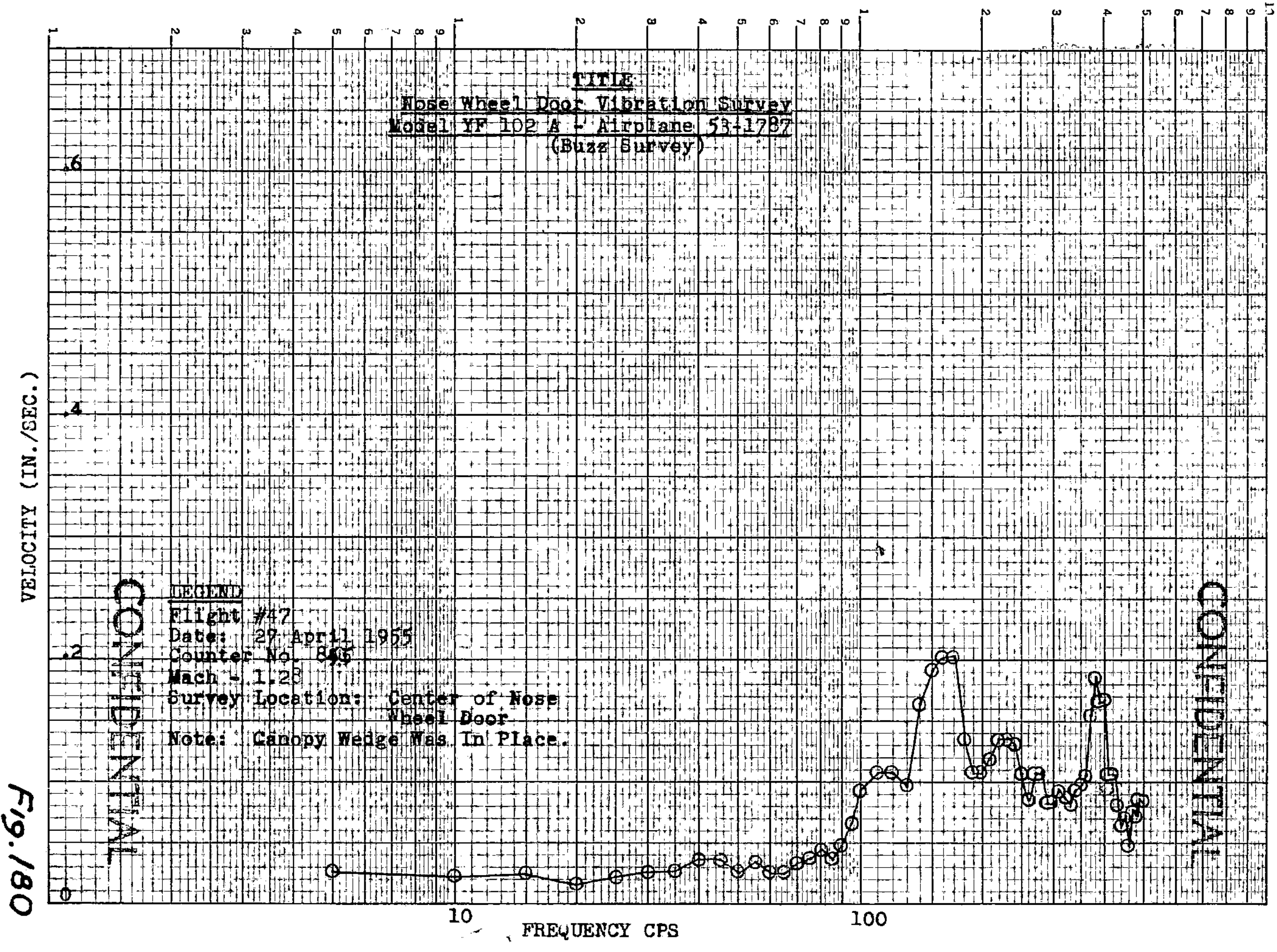
TITLE
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 849
Mach= 1.26 (approx.)
Survey Location: Center of Nose Wheel Door
Note: Canopy Wedge Was In Place

Fig. 179

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL

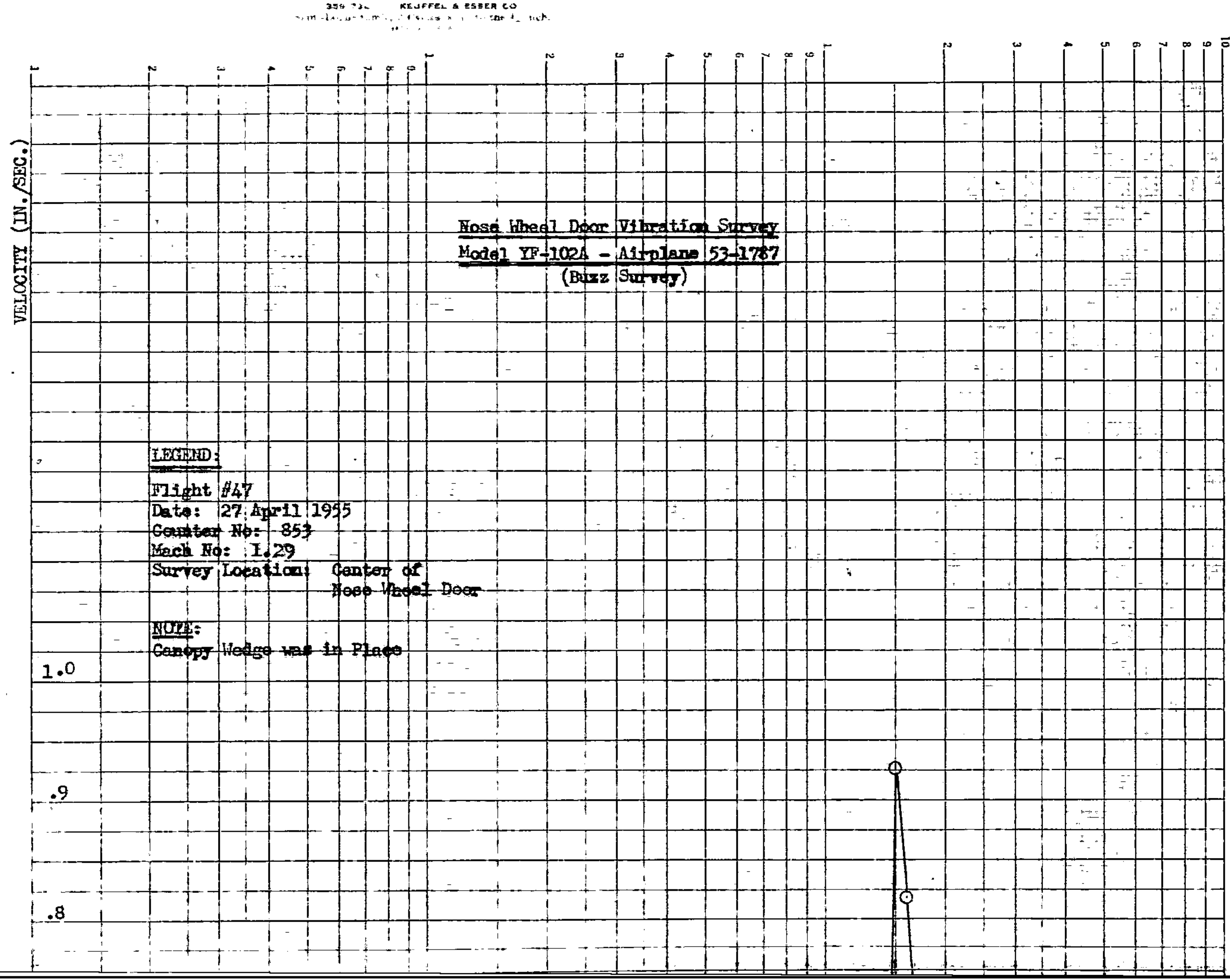


TITLE
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 845
Mach - 1.28
Survey Location: Center of Nose Wheel Door
Note: Canopy Wedge Was In Place.

CONFIDENTIAL

Fig. 180



Nose Wheel Door Vibration Survey

Model YF-102A - Airplane 53-1787

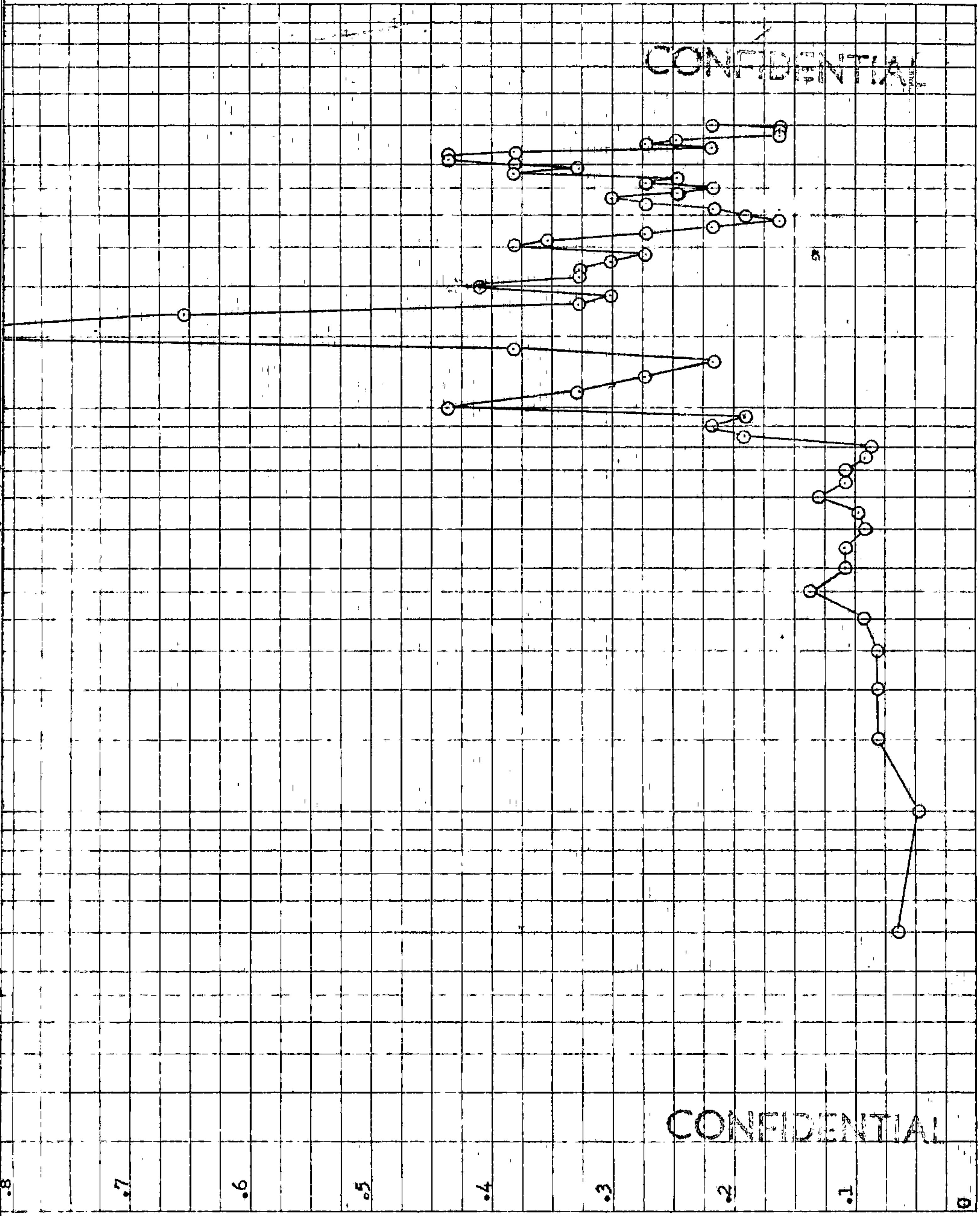
(Buzz Survey)

LEGEND:

Flight #47
Date: 27 April 1955
Counter No: 853
Mach No: 1.29
Survey Location: Center of
Nose Wheel Door

NOTE:
Canopy Wedge was in Place

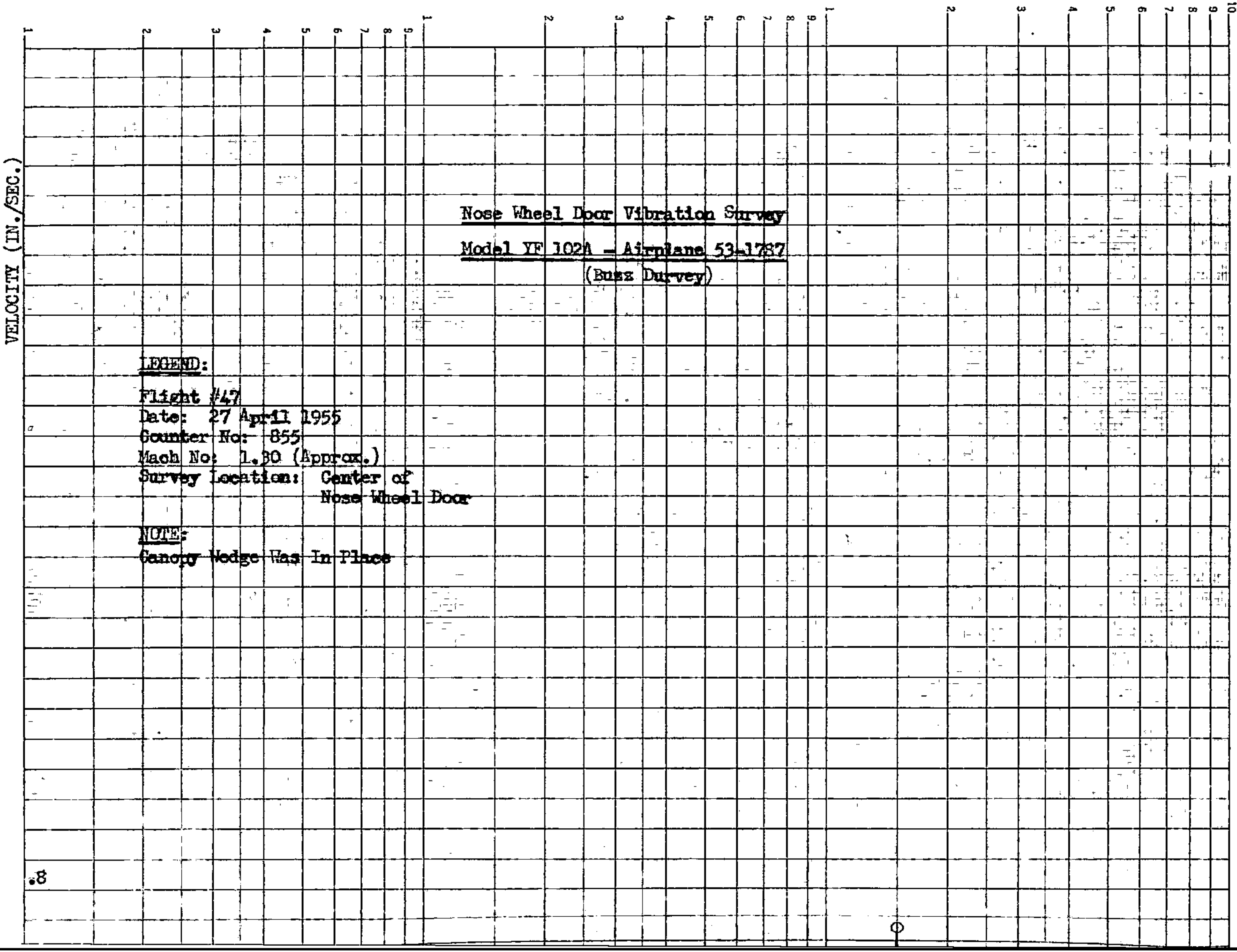
CONFIDENTIAL



CONFIDENTIAL

Fig. 181

359-73L KEUFFEL & ESSER CO.
Semi-Logarithmic 3 Cycles × 10 to the ½ inch.



Nose Wheel Door Vibration Survey

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:

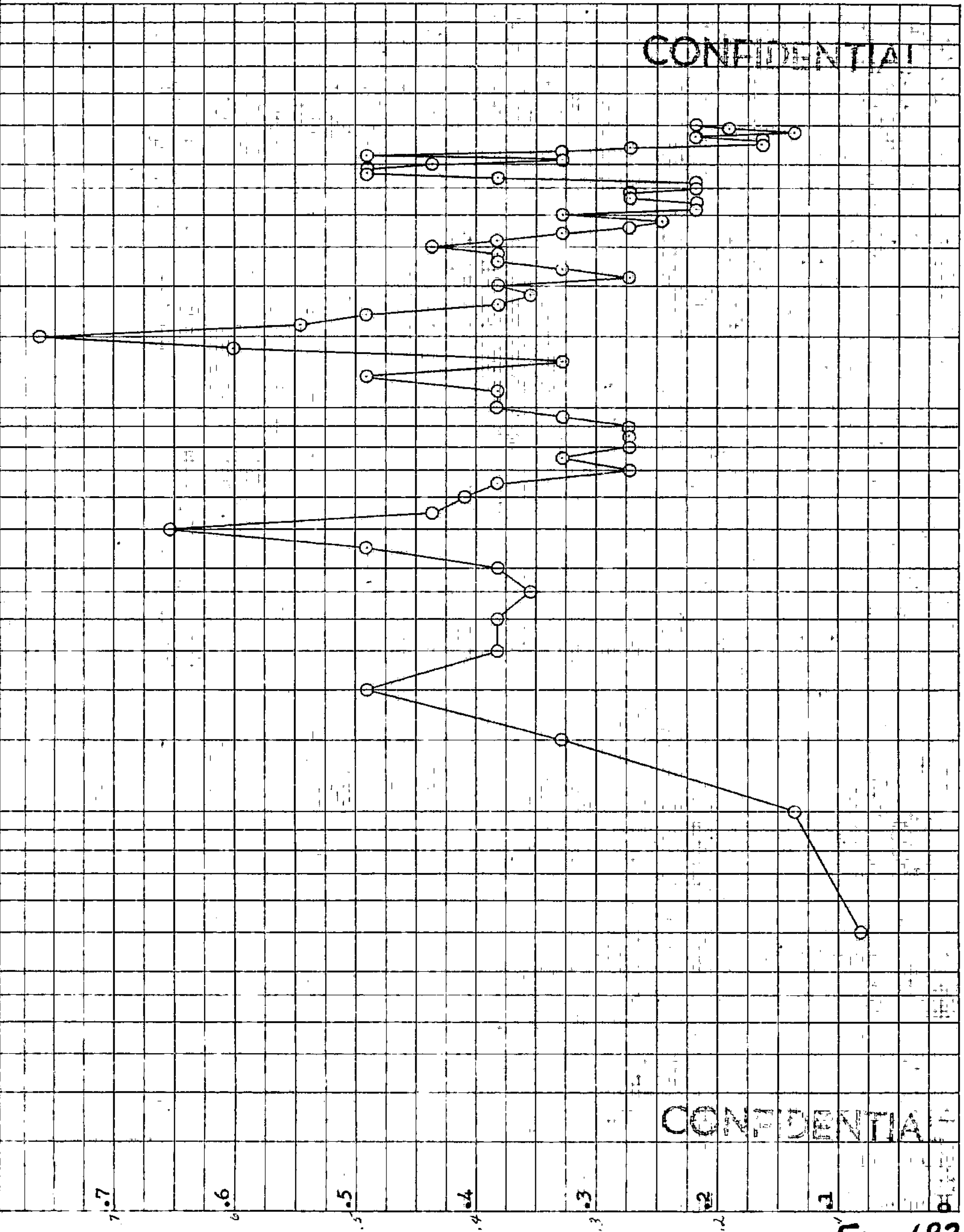
Flight #47
Date: 27 April 1955
Counter No: 855
Mach No: 1.30 (Approx.)
Survey Location: Center of
Nose Wheel Door

NOTE:

Canopy Wedge Was In Place

.8

CONFIDENTIAL

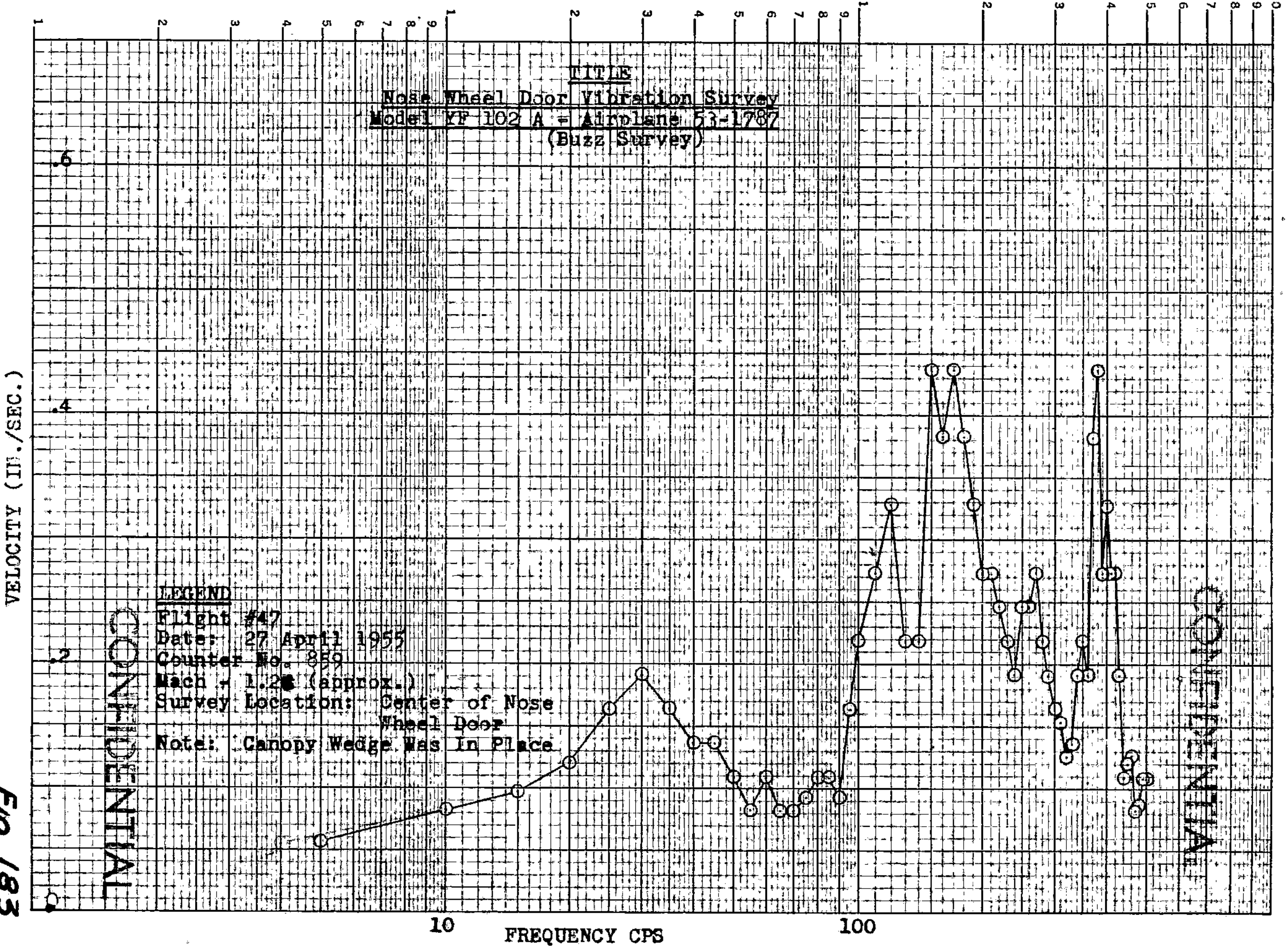


CONFIDENTIAL

Fig. 182

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



TITLE
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

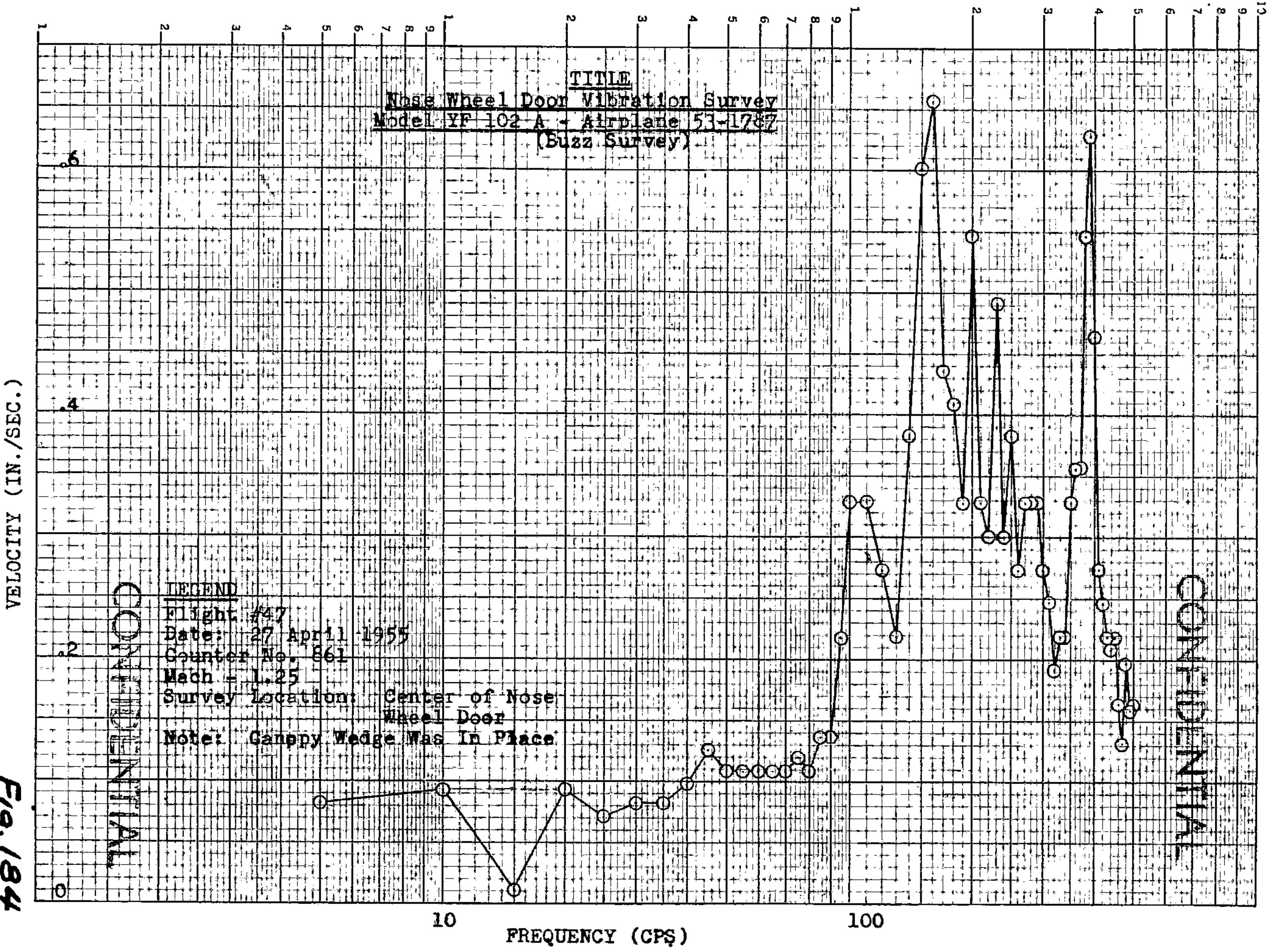
LEGEND
Flight #47
Date: 27 April 1955
Counter No. 859
Mach - 1.2[illegible] (approx.)
Survey Location: Center of Nose
Wheel Door
Note: Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 183

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



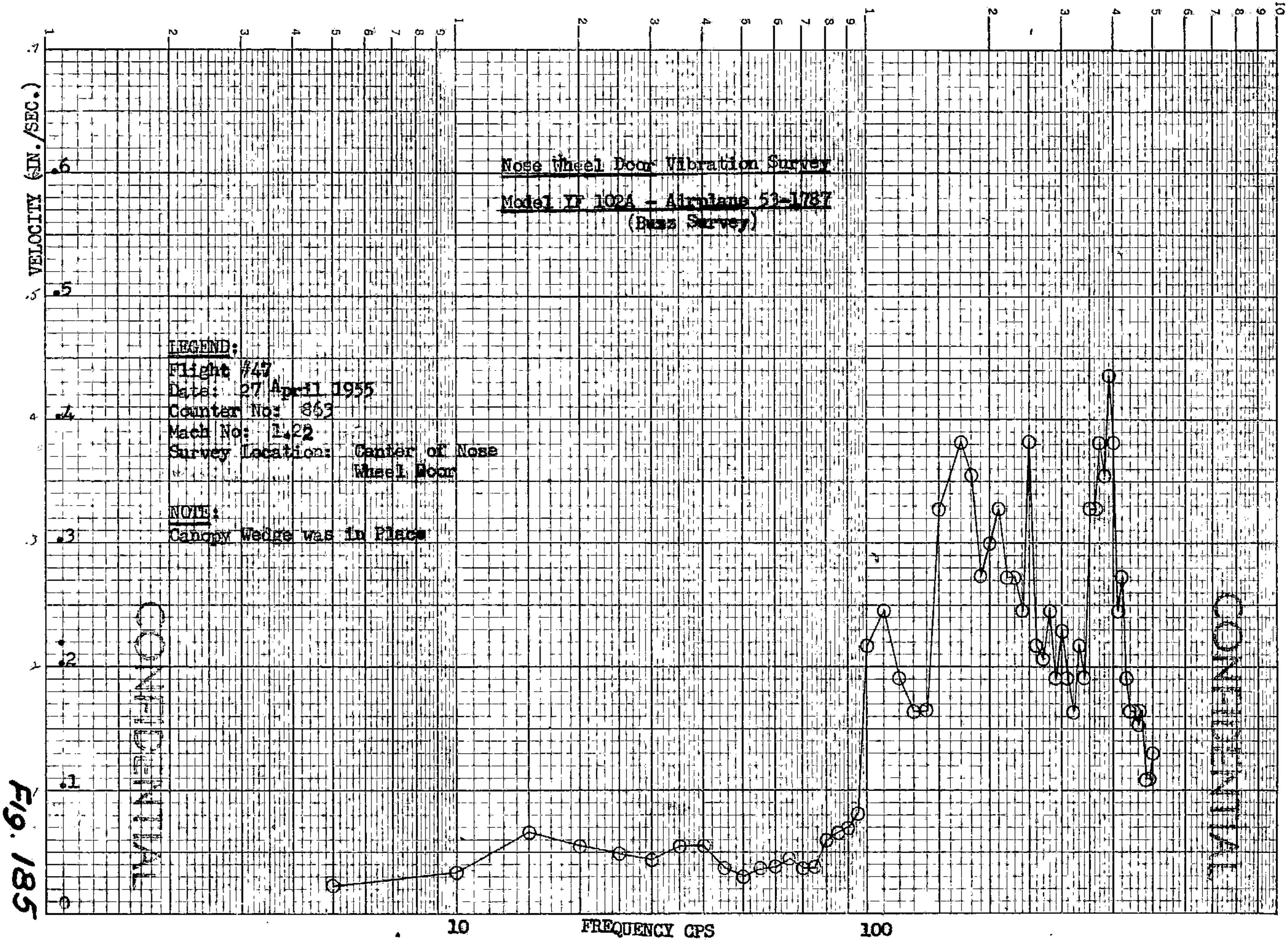
TITLE
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND
Flight #47
Date: 27 April 1955
Counter No. 861
Mach = 1.25
Survey Location: Center of Nose Wheel Door
Note: Canopy Wedge Was In Place

CONFIDENTIAL

Fig. 184

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



Nose Wheel Door Vibration Survey

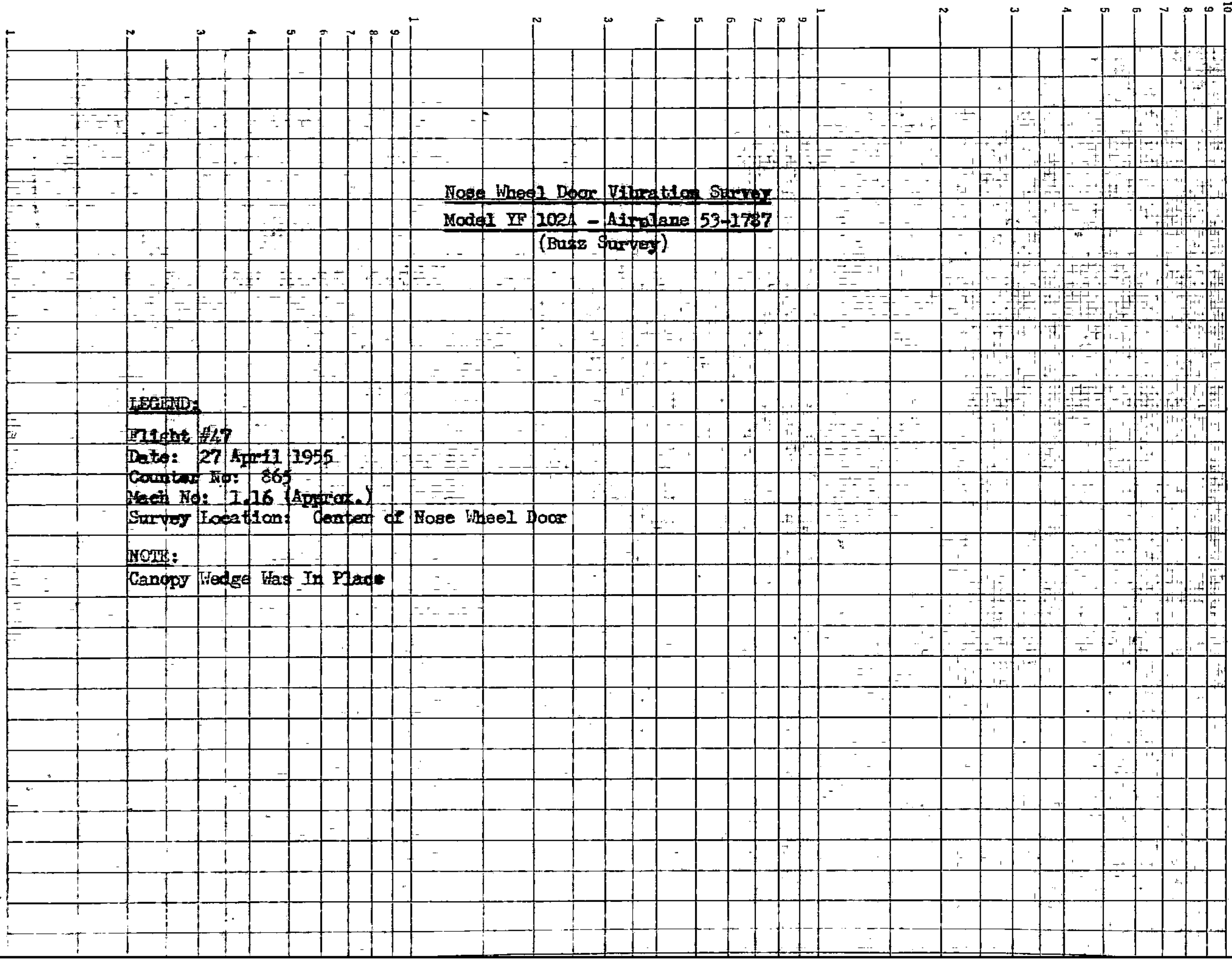
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #47
Date: 27 April 1955
Counter No: 863
Mach No: 1.22
Survey Location: Center of Nose Wheel Door

NOTE:
Canopy Wedge was in Place

Fig. 185

359-73L KEUFFEL & ESSER CO.
Semi-Logarithmic, 3 Cycles x 10 to the 1/2 inch
MADE IN U.S.A.



Nose Wheel Door Vibration Survey

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:

Flight #47
Date: 27 April 1955
Counter No: 865
Mach No: 1.16 (Approx.)
Survey Location: Center of Nose Wheel Door

NOTE:

Canopy Wedge Was In Place

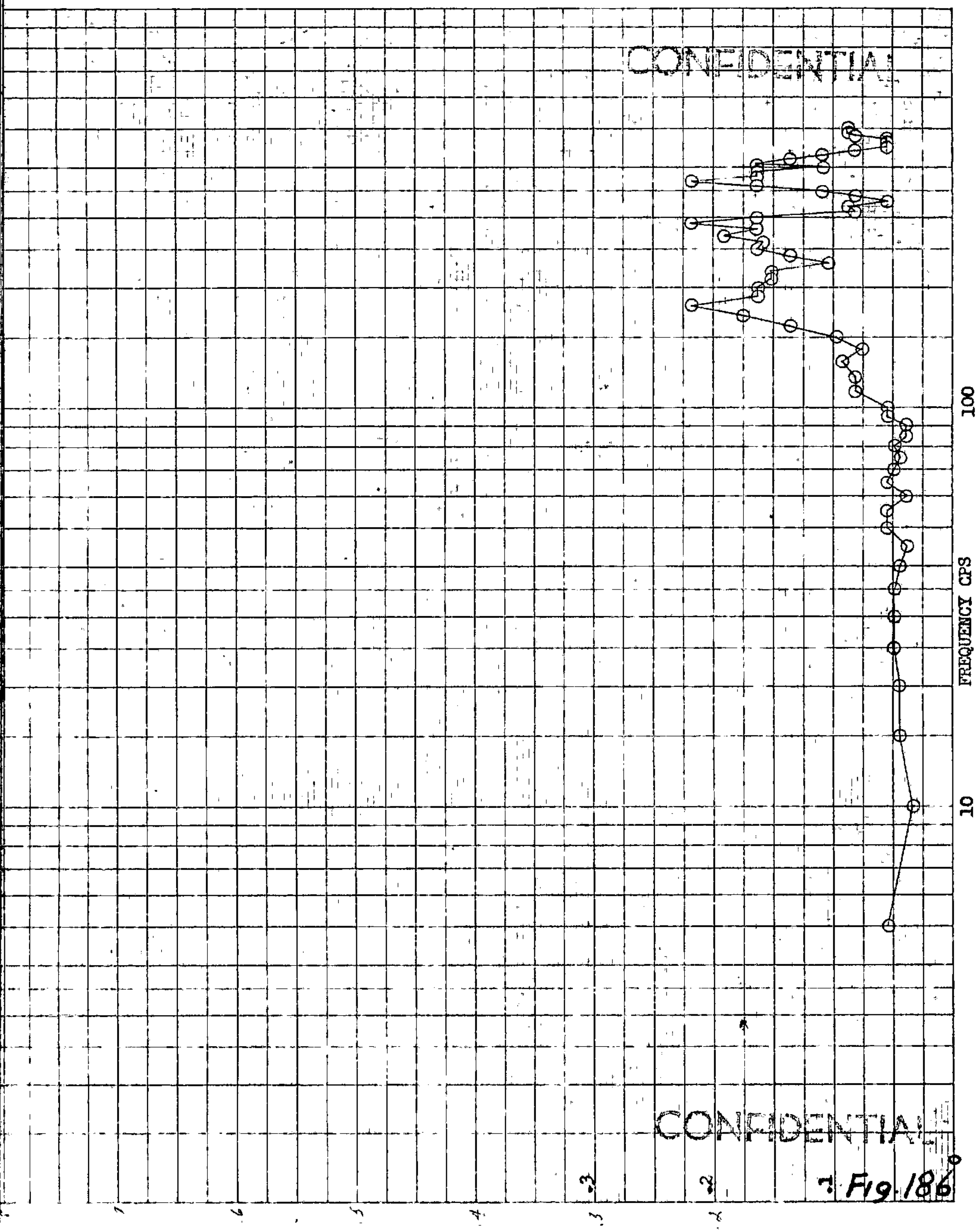
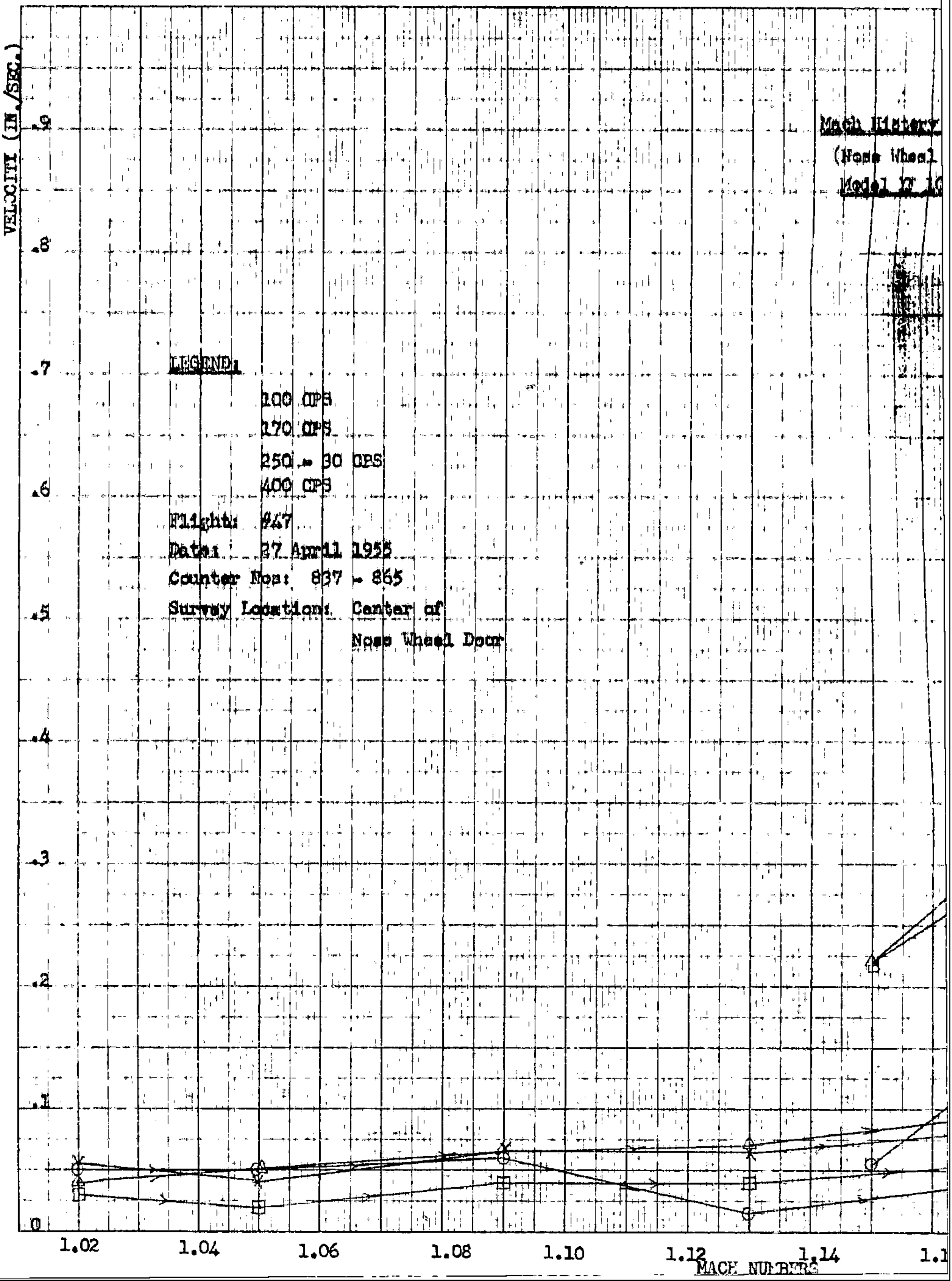
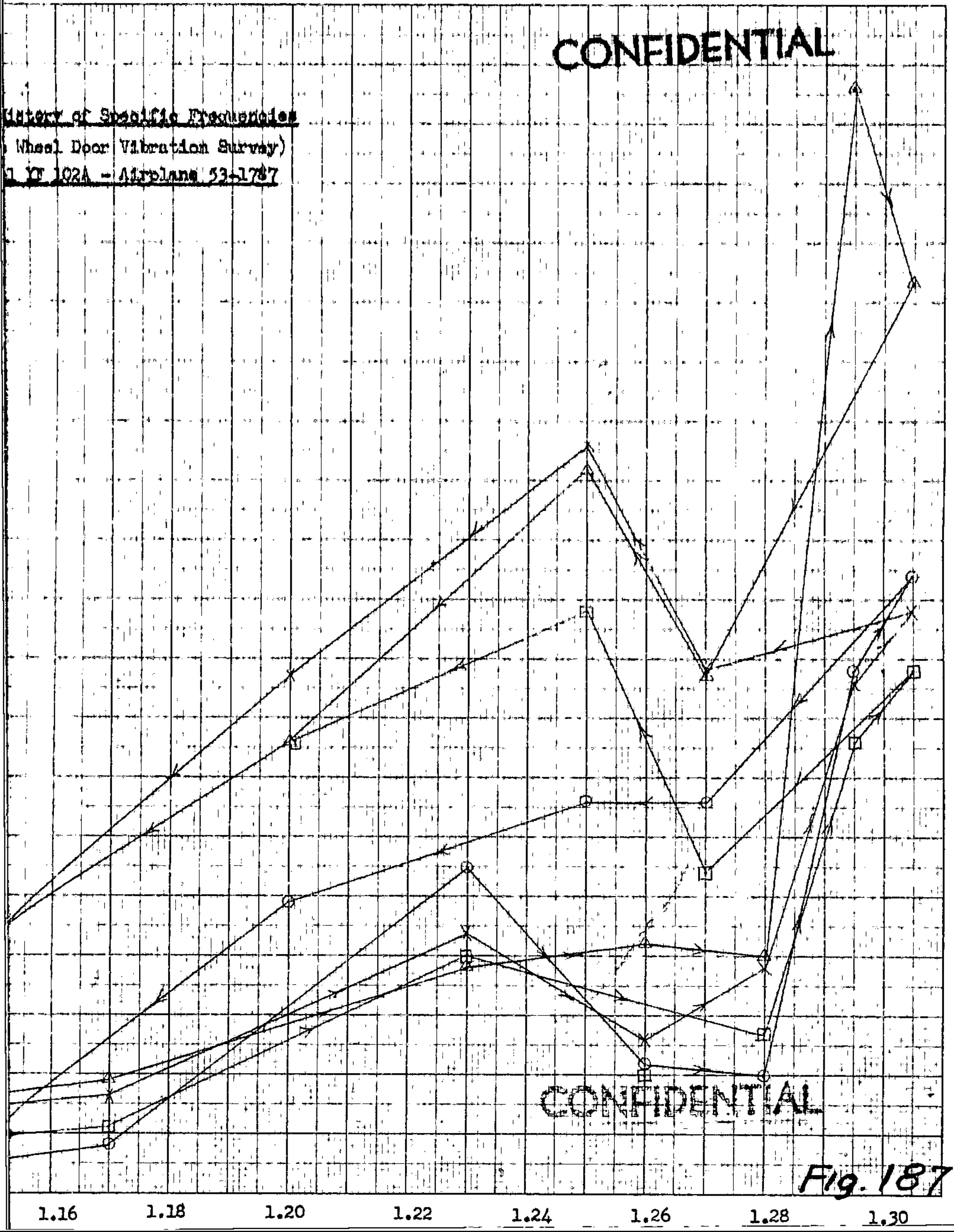


Fig 18b


Mach History
(Nose Wheel
Model YF 10
LEGEND:
100 CPS
170 CPS
250 - 30 CPS
400 CPS
Flight: #47
Date: 27 April 1955
Counter Nos: 837 - 865
Survey Location: Center of
Nose Wheel Door
VELOCITY (IN./SEC.)
.9
.8
.7
.6
.5
.4
.3
.2
.1
0
1.02
1.04
1.06
1.08
1.10
1.12
1.14
1.1
MACH NUMBERS

CONFIDENTIAL



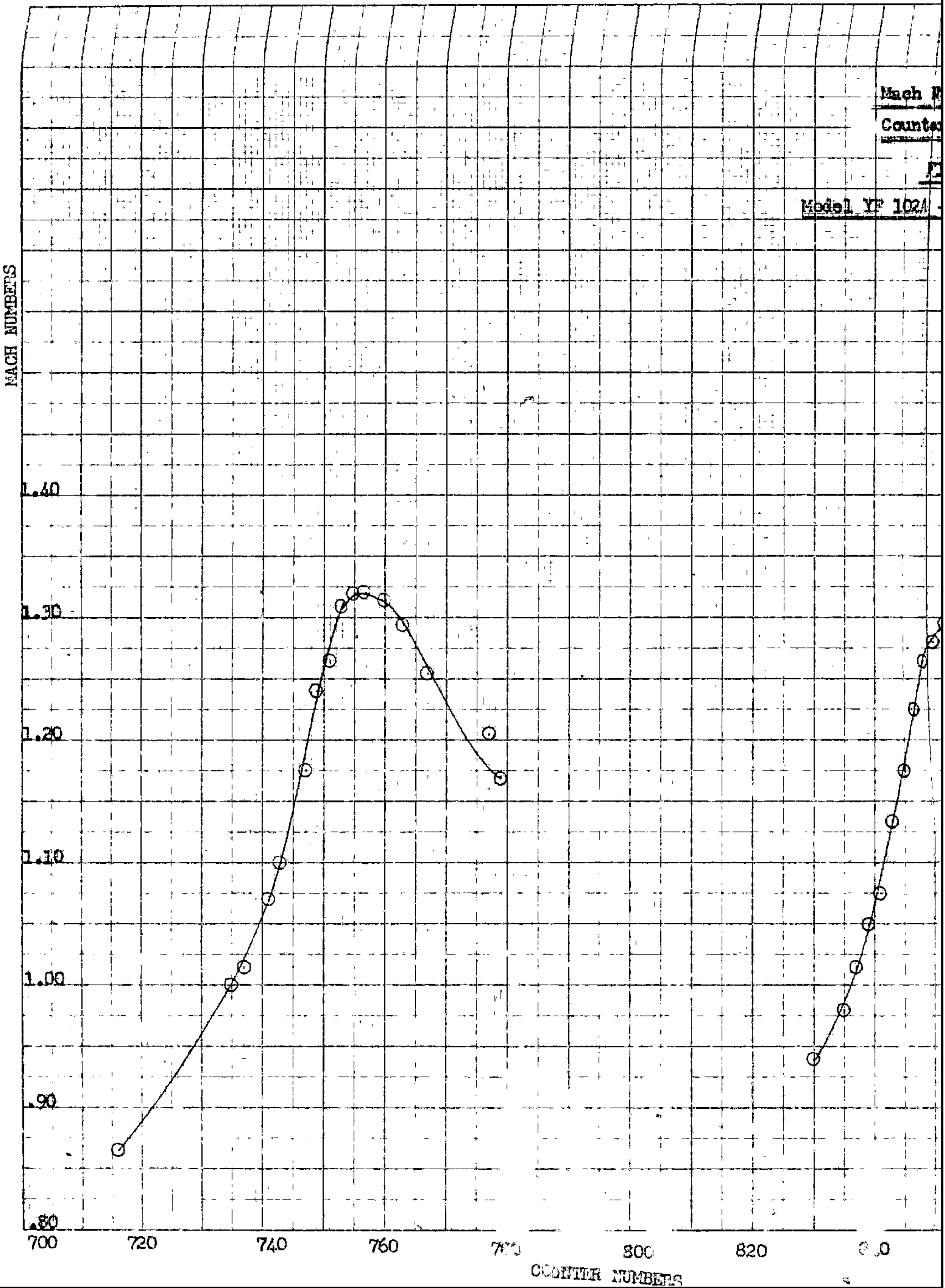
istory of Specific Frequencies

Wheel Door Vibration Survey)

1 YF 102A - Airplane 53-1787

CONFIDENTIAL

Fig. 187


Mach
Model YF 102A
MACH NUMBERS
1.40
1.30
1.20
1.10
1.00
.90
.80
700
720
740
760
800
820
COUNTER NUMBERS

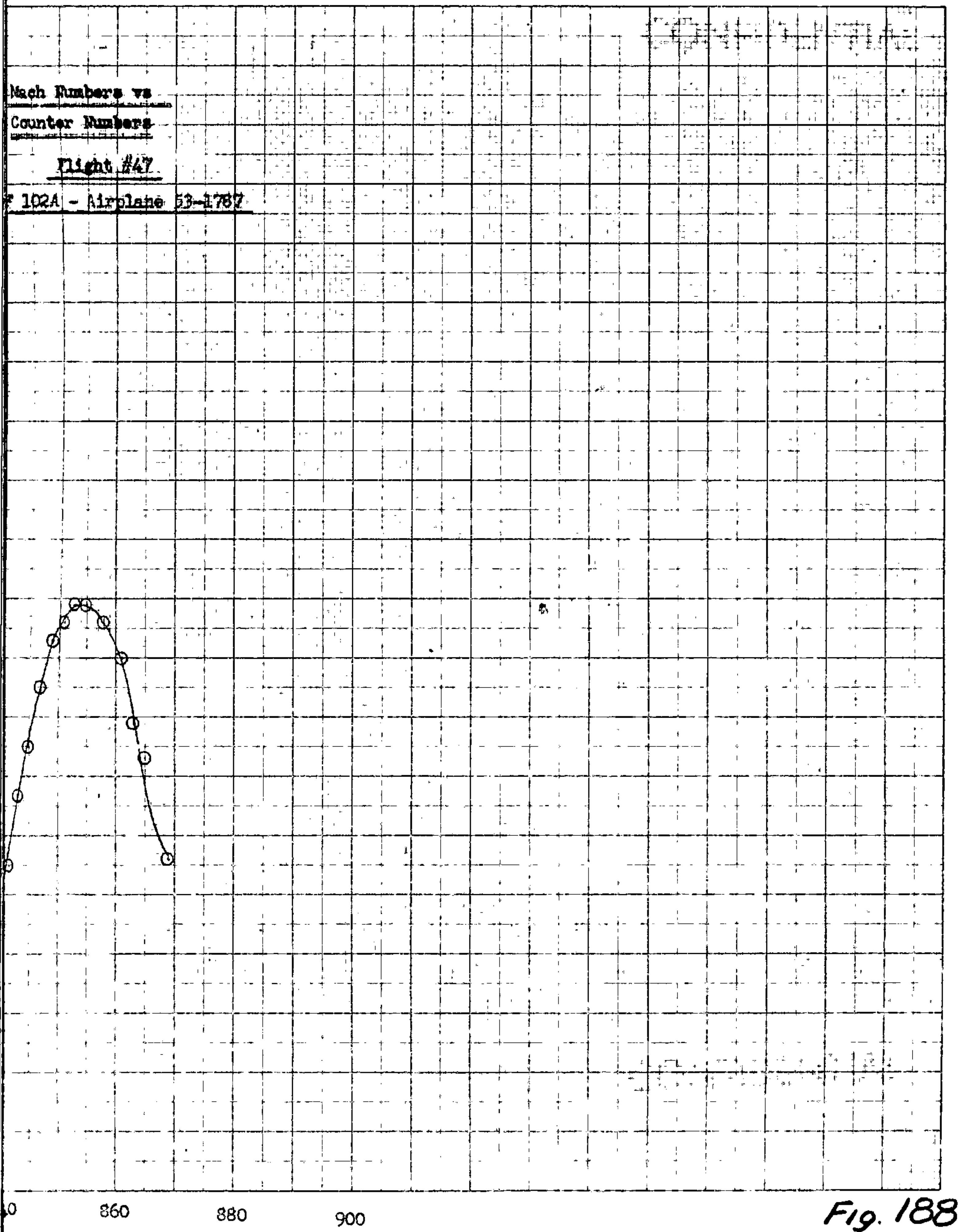
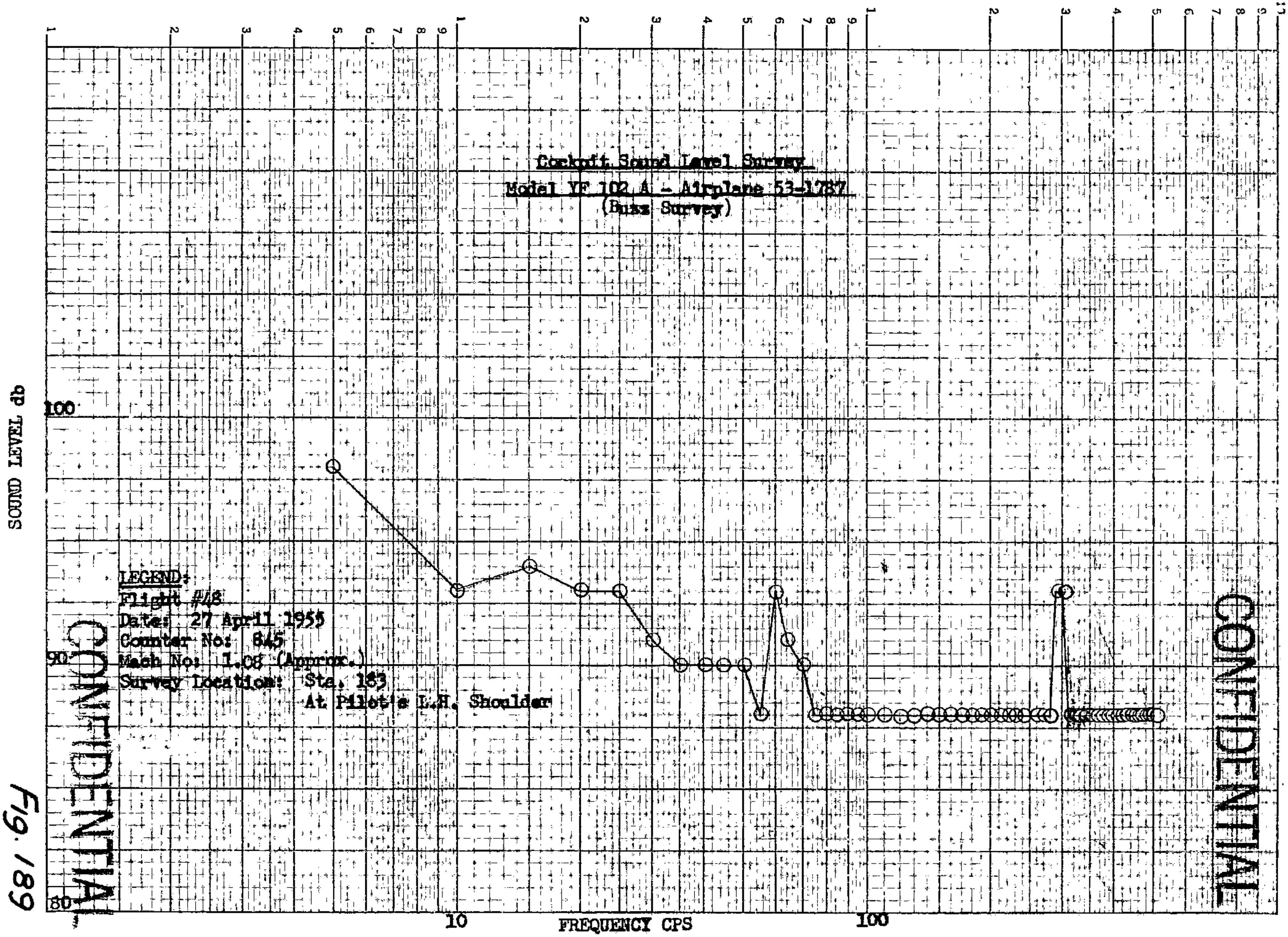

Mach Numbers vs
Counter Numbers
Flight #47
102A - Airplane 53-1787
0
860
880
900

Fig. 188

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

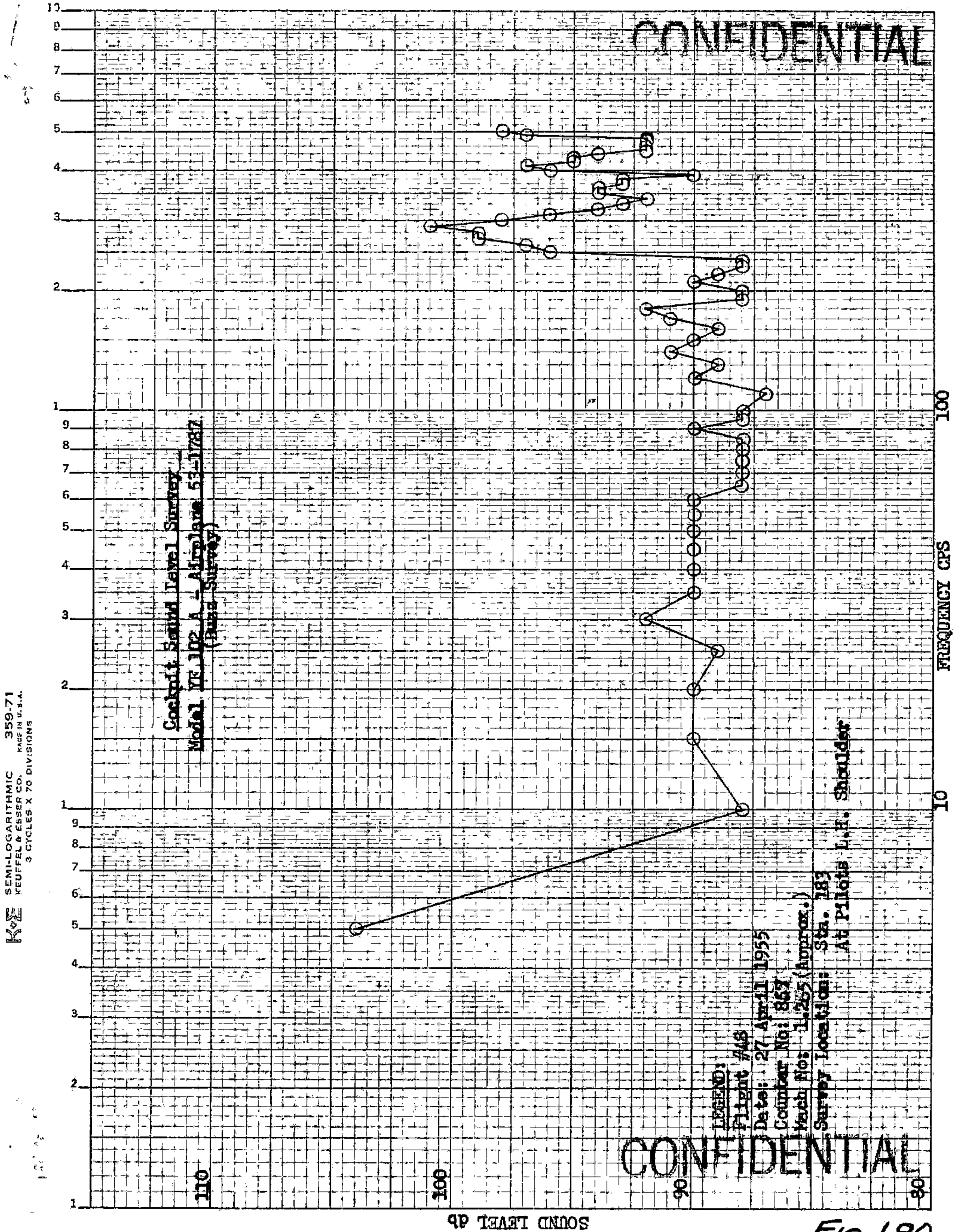
CONFIDENTIAL



CONFIDENTIAL

Fig. 189

CONFIDENTIAL

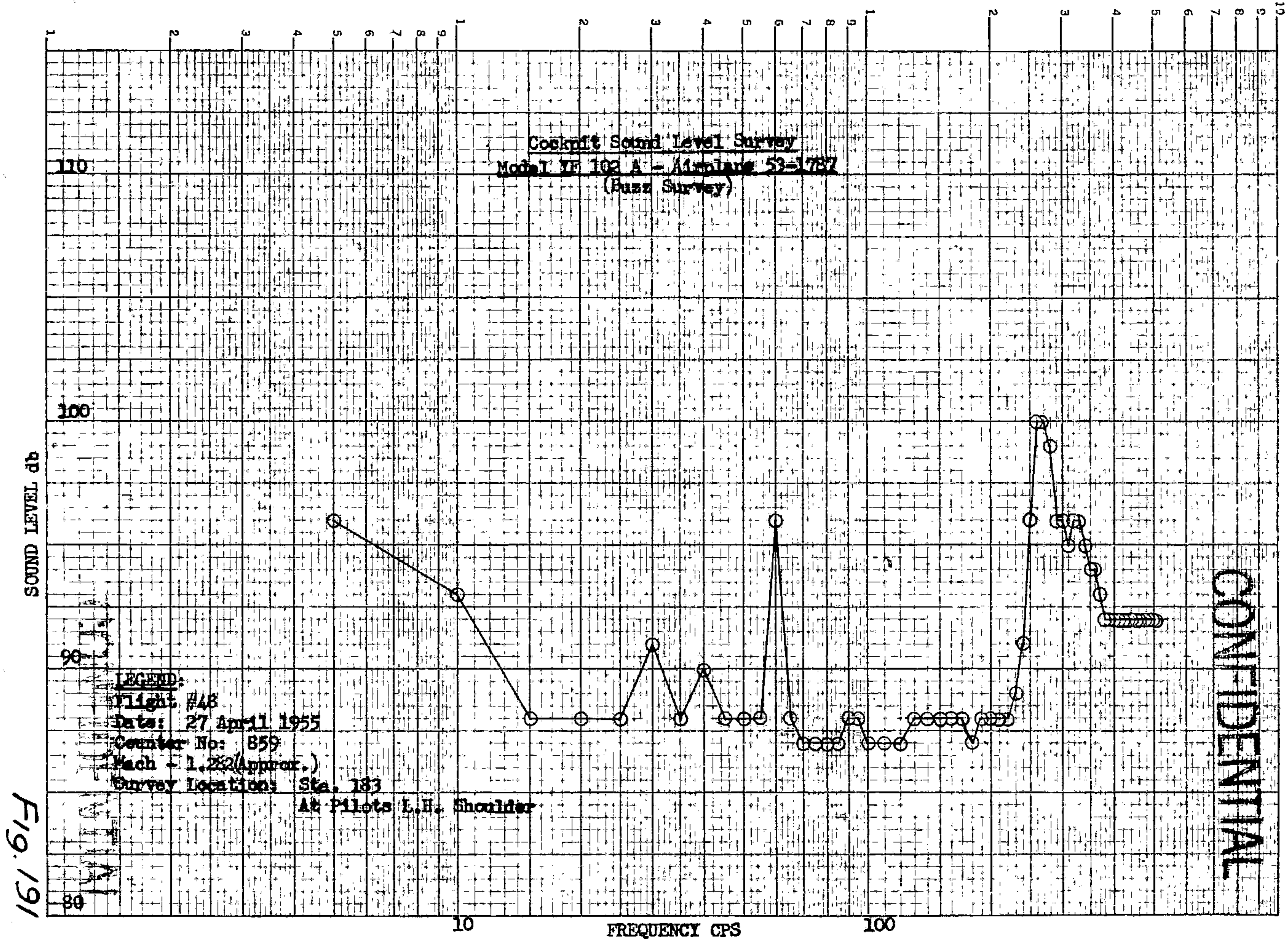


CONFIDENTIAL

Fig. 190

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Cockpit Sound Level Survey
Model XF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 859
Mach - 1.282(Approx.)
Survey Location: Sta. 183
At Pilots L.H. Shoulder

Fig. 191

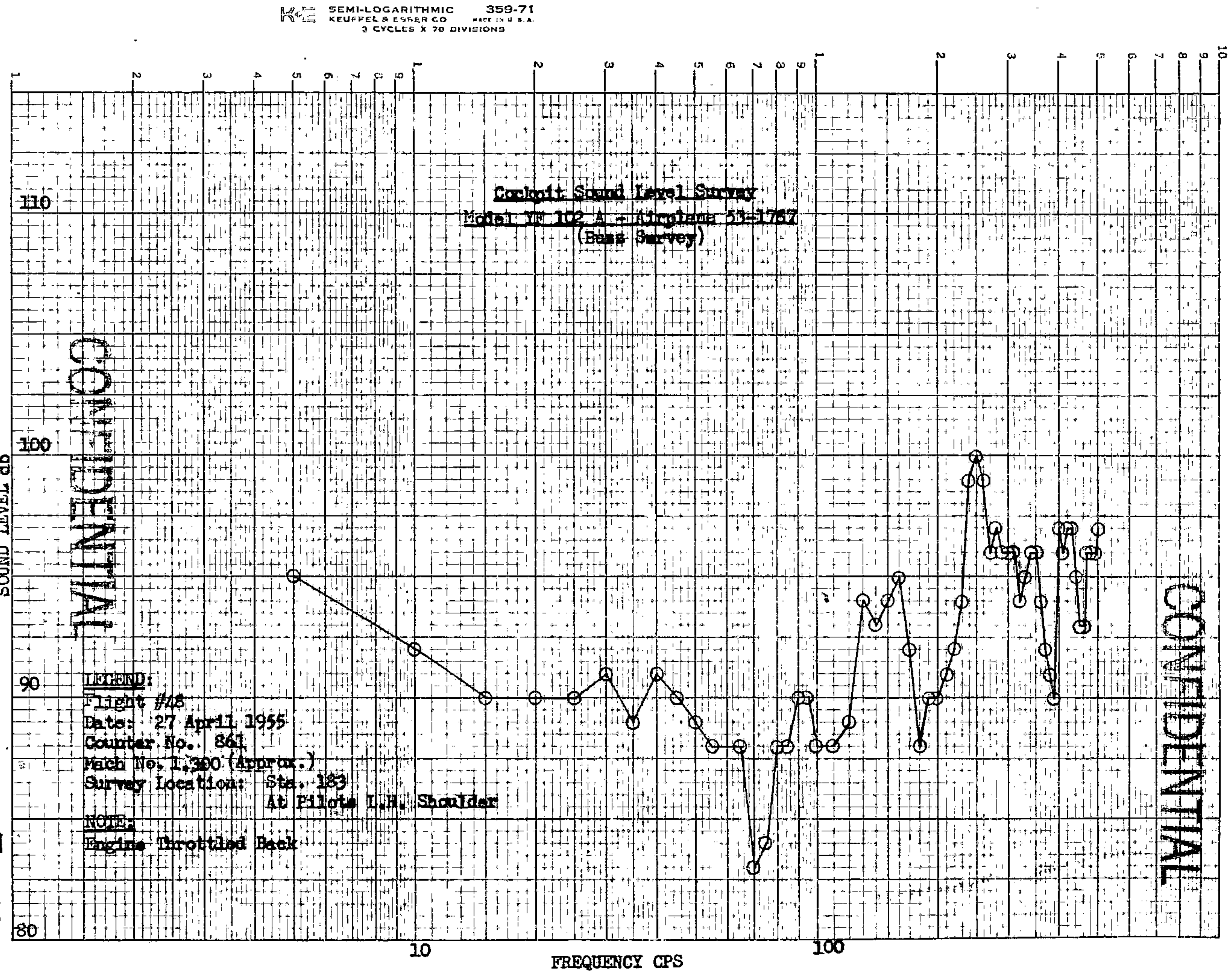
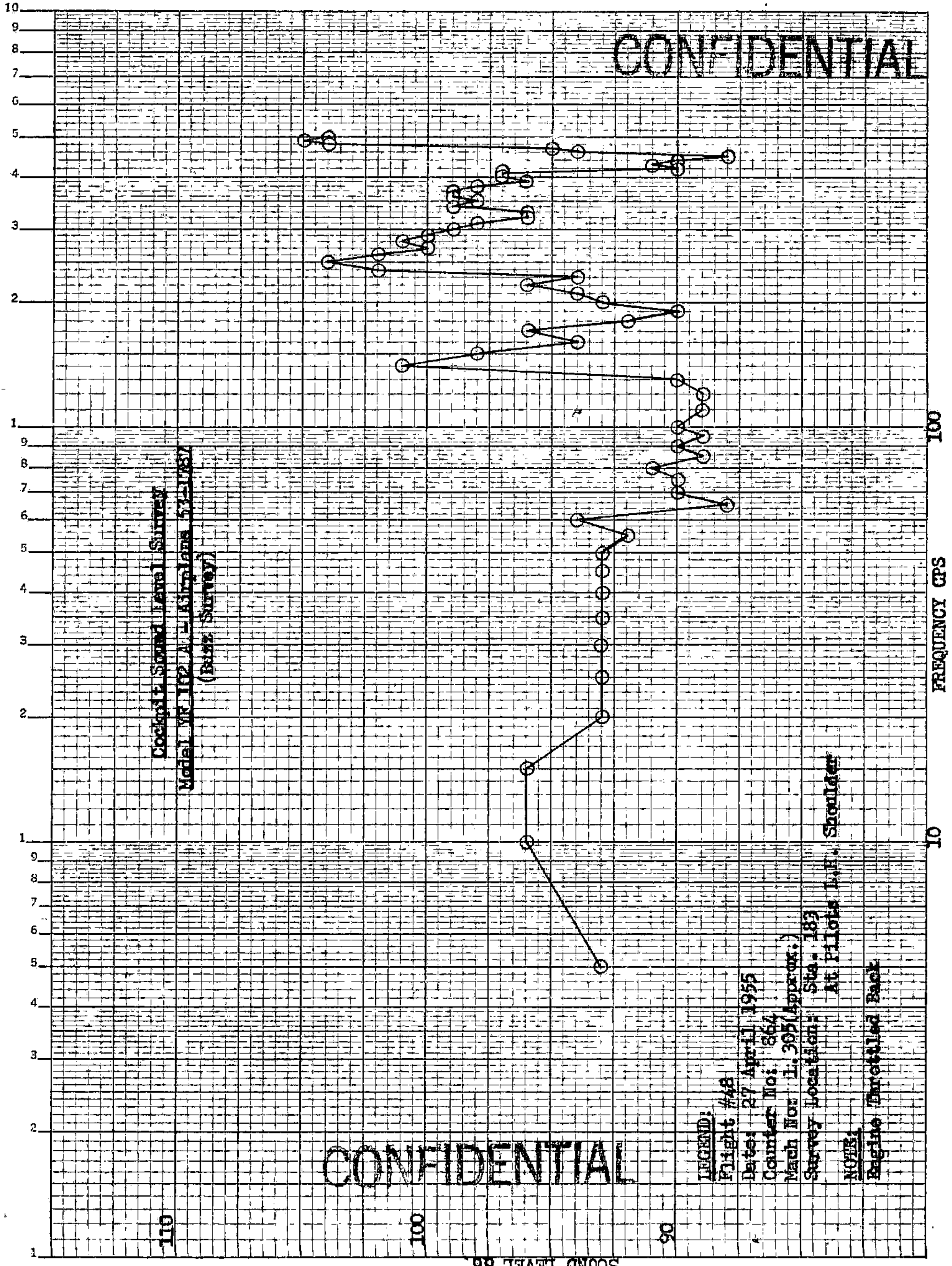
K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS
Cockpit Sound Level Survey
Model YF 102 A - Airplane 53-1767
(Buzz Survey)
CONFIDENTIAL
CONFIDENTIAL
LEGEND:
Flight #48
Date: 27 April 1955
Counter No. 861
Mach No. 1.390 (Approx.)
Survey Location: Sta. 183
At Pilots L.H. Shoulder
NOTE:
Engine Throttled Back
SOUND LEVEL db
110
100
90
80
FREQUENCY CPS
10
100
1
2
3
4
5
6
7
8
9
10

Fig. 192

CONFIDENTIAL



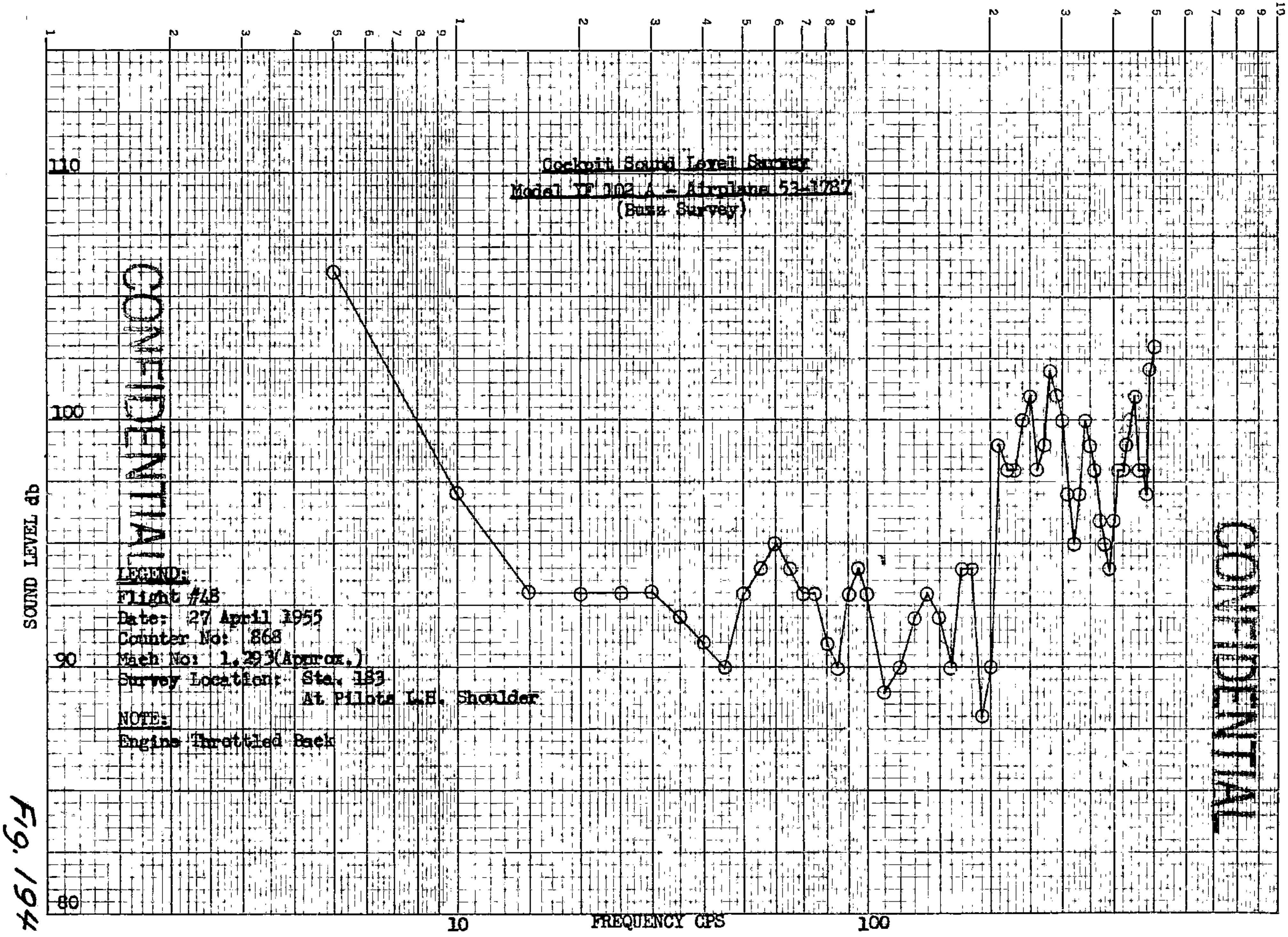
CONFIDENTIAL

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

Fig. 193

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Cockpit Sound Level Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 868
Mach No: 1.293 (Approx.)
Survey Location: Sta. 183
At Pilots L.H. Shoulder

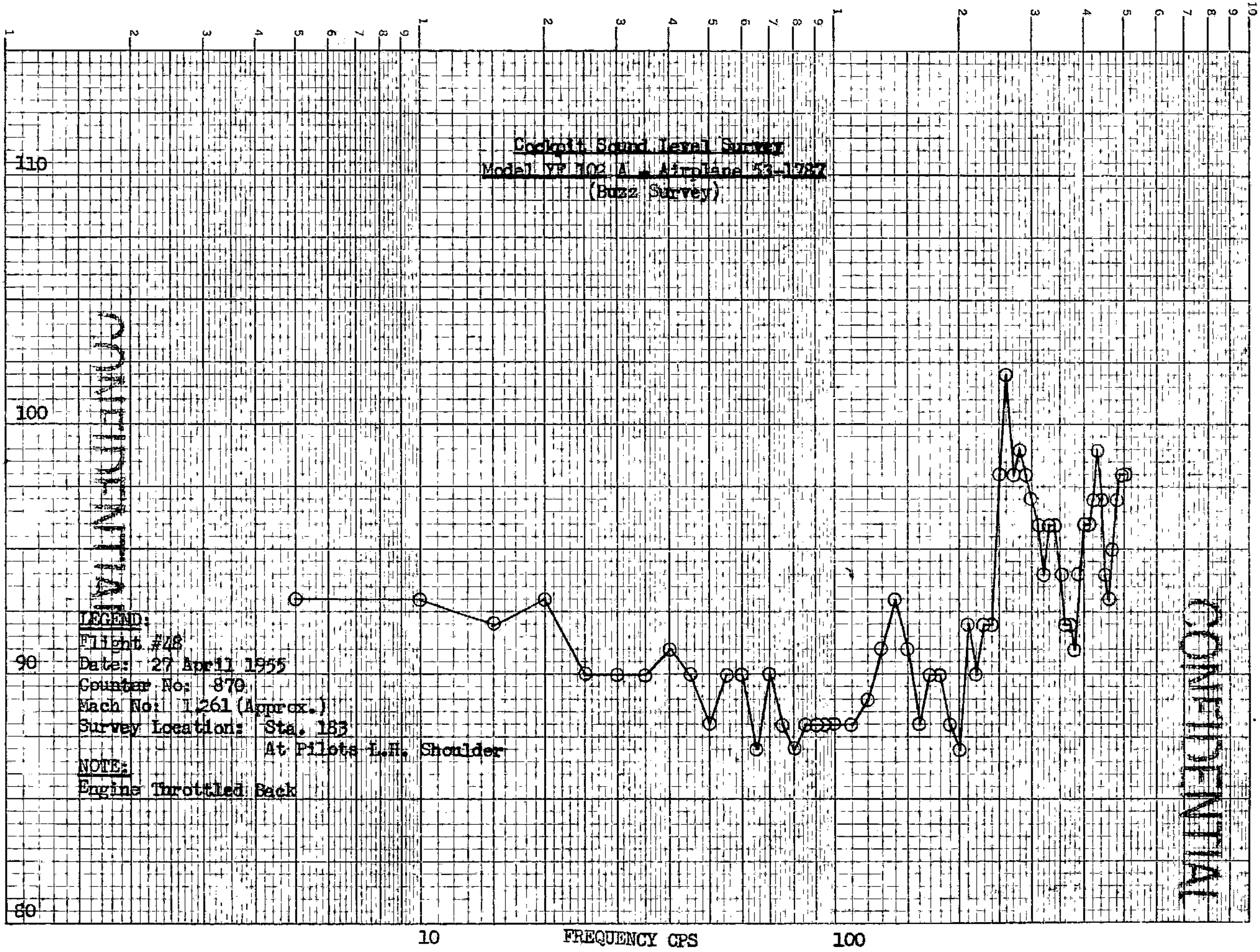
NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 194

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



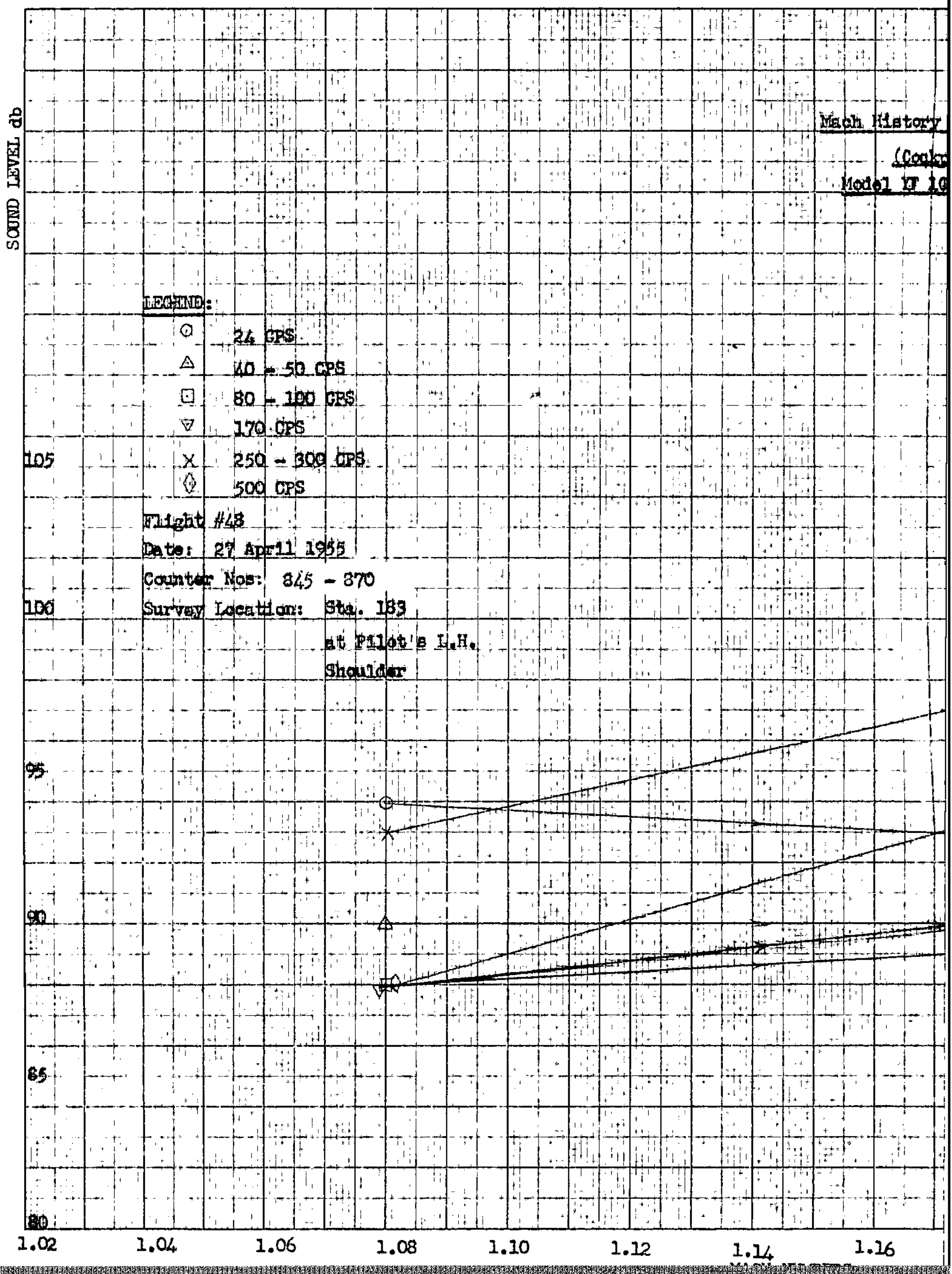
Cockpit Sound Level Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 870
Mach No: 1.261 (Approx.)
Survey Location: Sta. 183
At Pilots L.H. Shoulder

NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 195



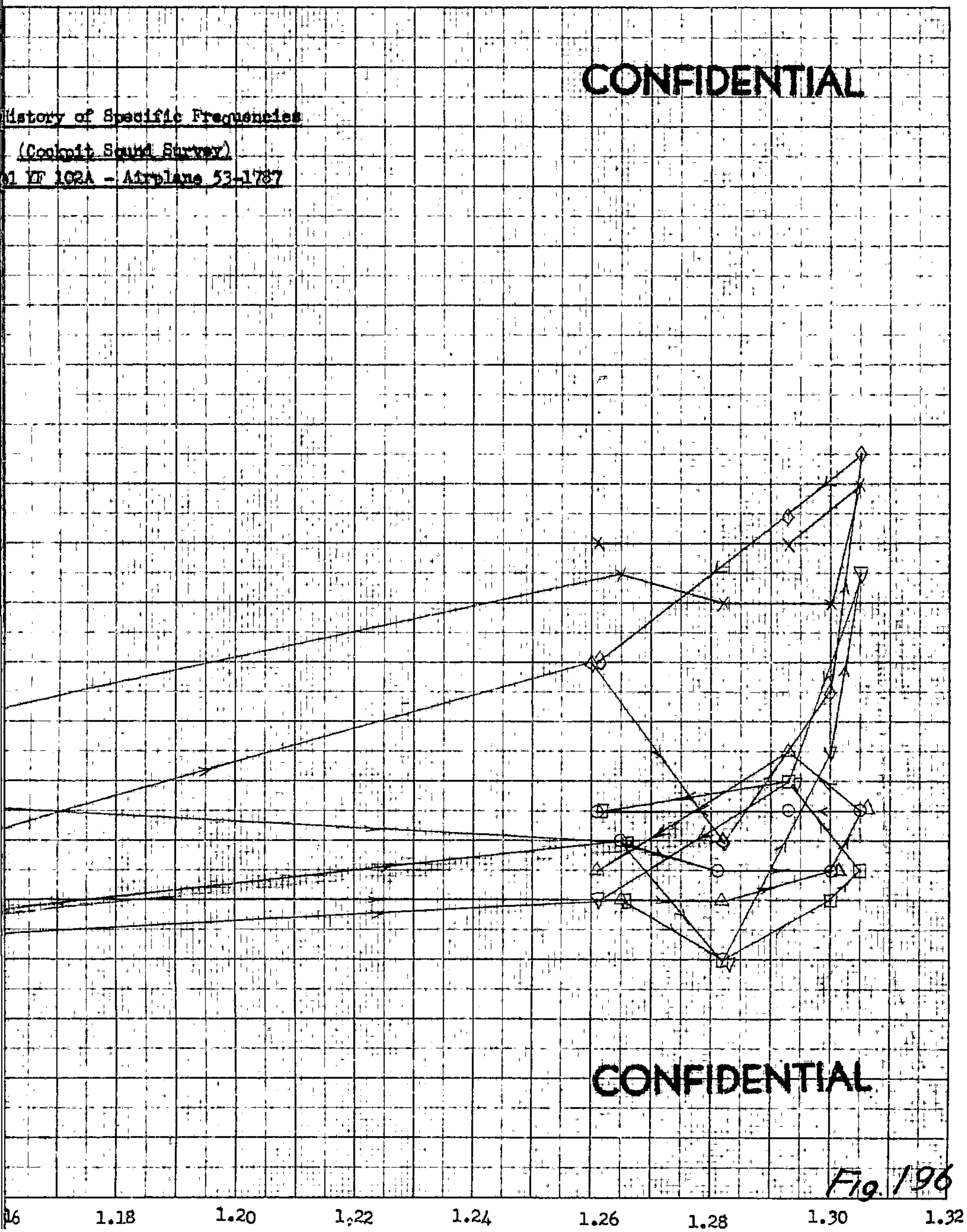
10 X 10 TO THE ½ INCH 359-11L

CONFIDENTIAL

istory of Specific Frequencies

(Cockpit Sound Survey)

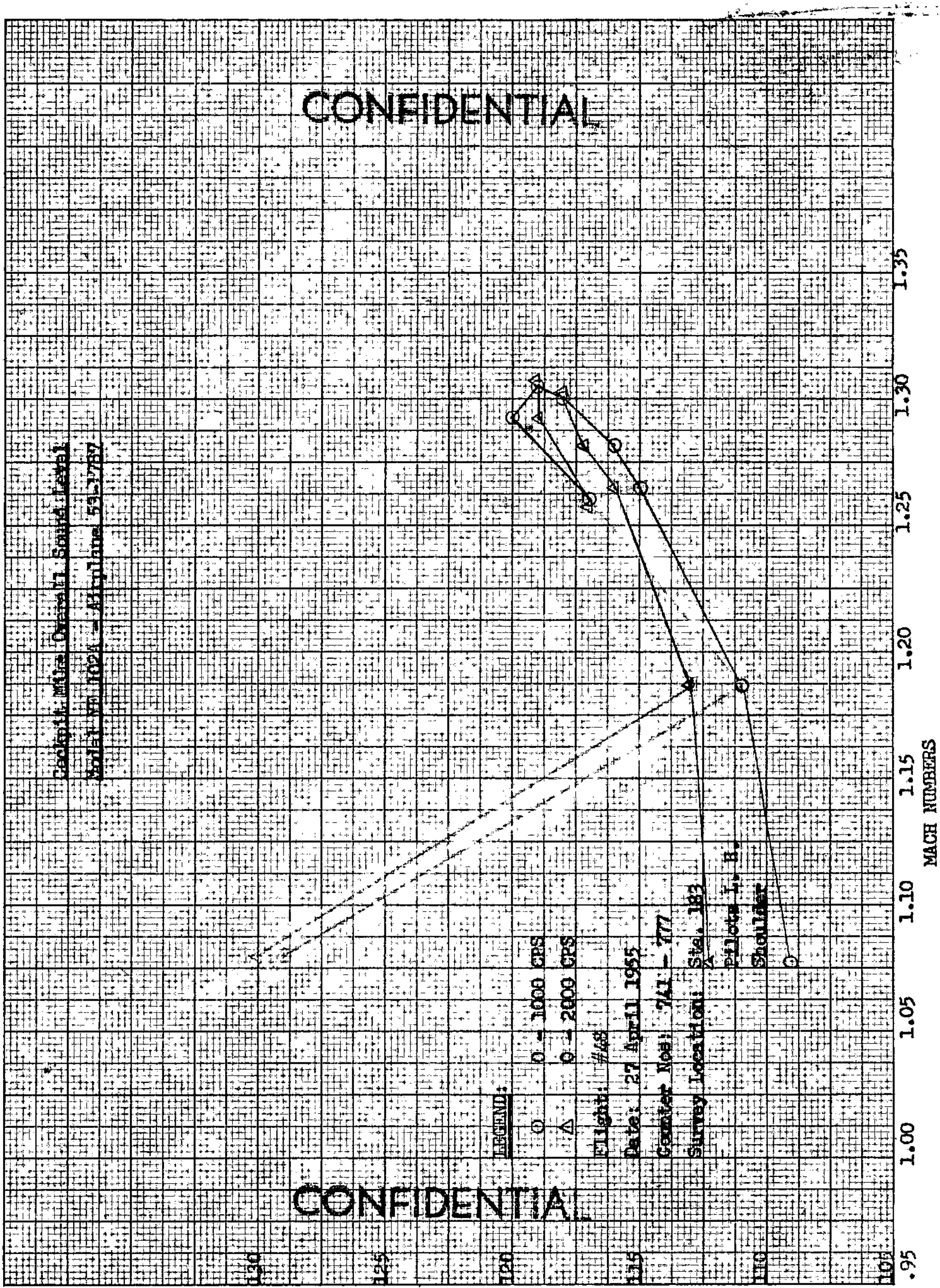
1 YF 102A - Airplane 53-1787



CONFIDENTIAL

Fig. 196

CONFIDENTIAL



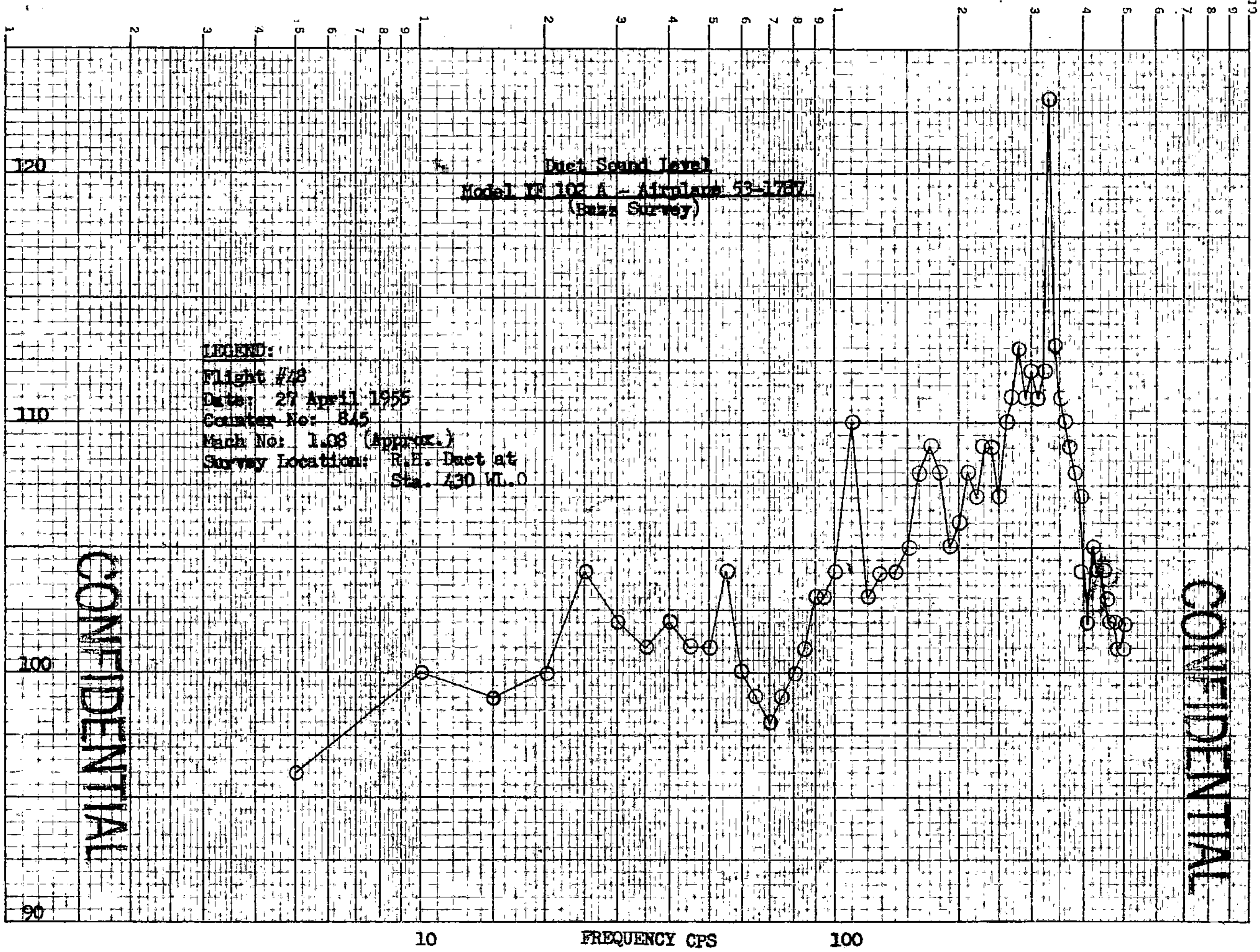
CONFIDENTIAL

K+E 10 X 10 TO THE ½ INCH 359.11
KEUFFEL & ESSER CO.

Fig 197

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Sound Level
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

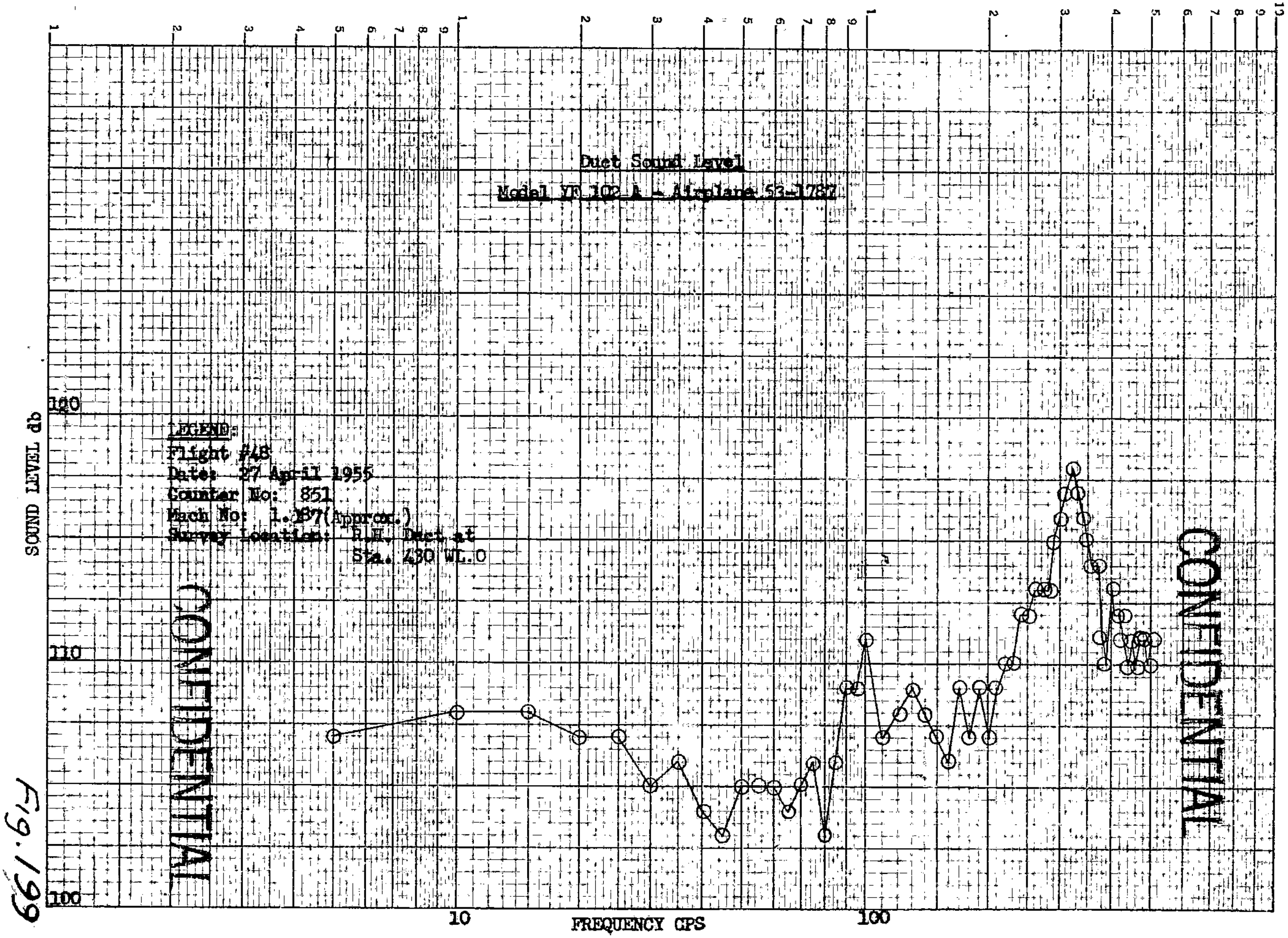
LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 845
Mach No: 1.08 (Approx.)
Survey Location: R.H. Duct at
Sta. 430 WL.0

CONFIDENTIAL

Fig. 198

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Sound Level

Model YF-102 A - Airplane 53-1787

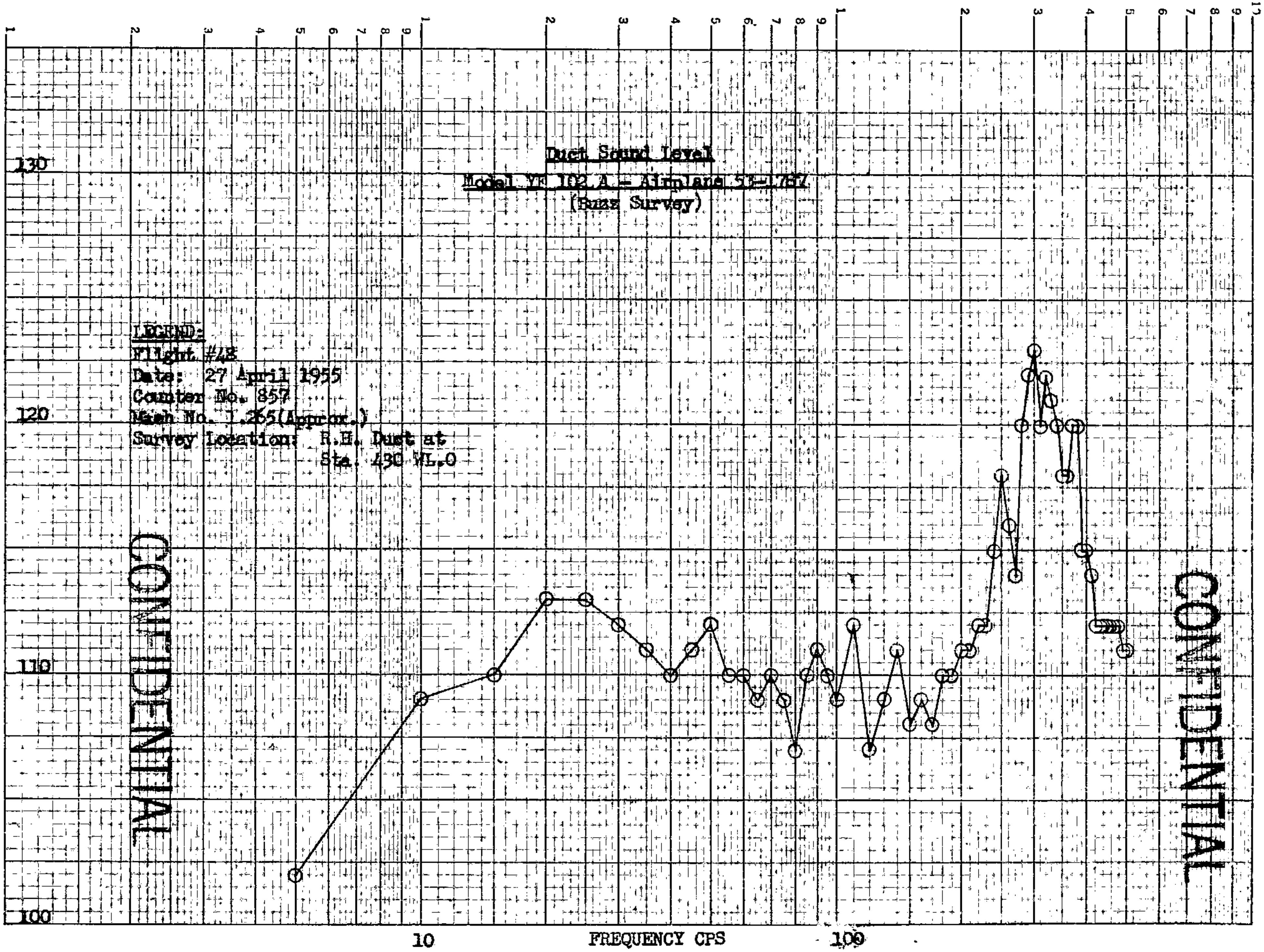
LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 851
Mach No: 1.187(Approx.)
Survey Location: R.H. Duct at
Sta. 430 WL.0

CONFIDENTIAL

Fig. 199

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



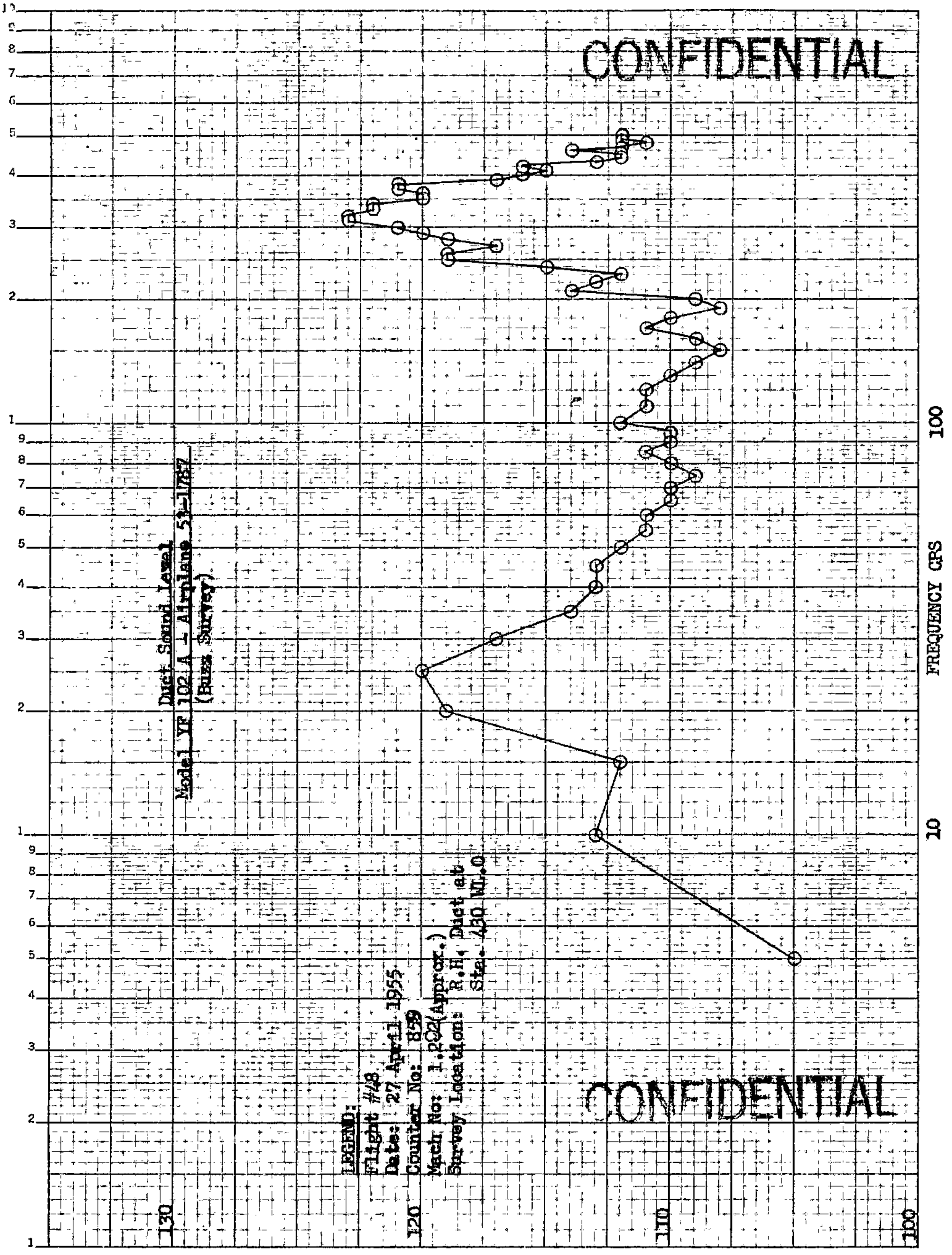
Duct Sound Level
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No. 857
Mach No. 1.265(Approx.)
Survey Location: R.H. Duct at
Sta. 430 WL.0

CONFIDENTIAL

Fig. 200

CONFIDENTIAL



Duct Sound Level
Model YF 102 A - Airplane 51-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 859
Mach No: 1.202(Approx.)
Survey Location: R.H. Duct at
Sta. 430 WL 0

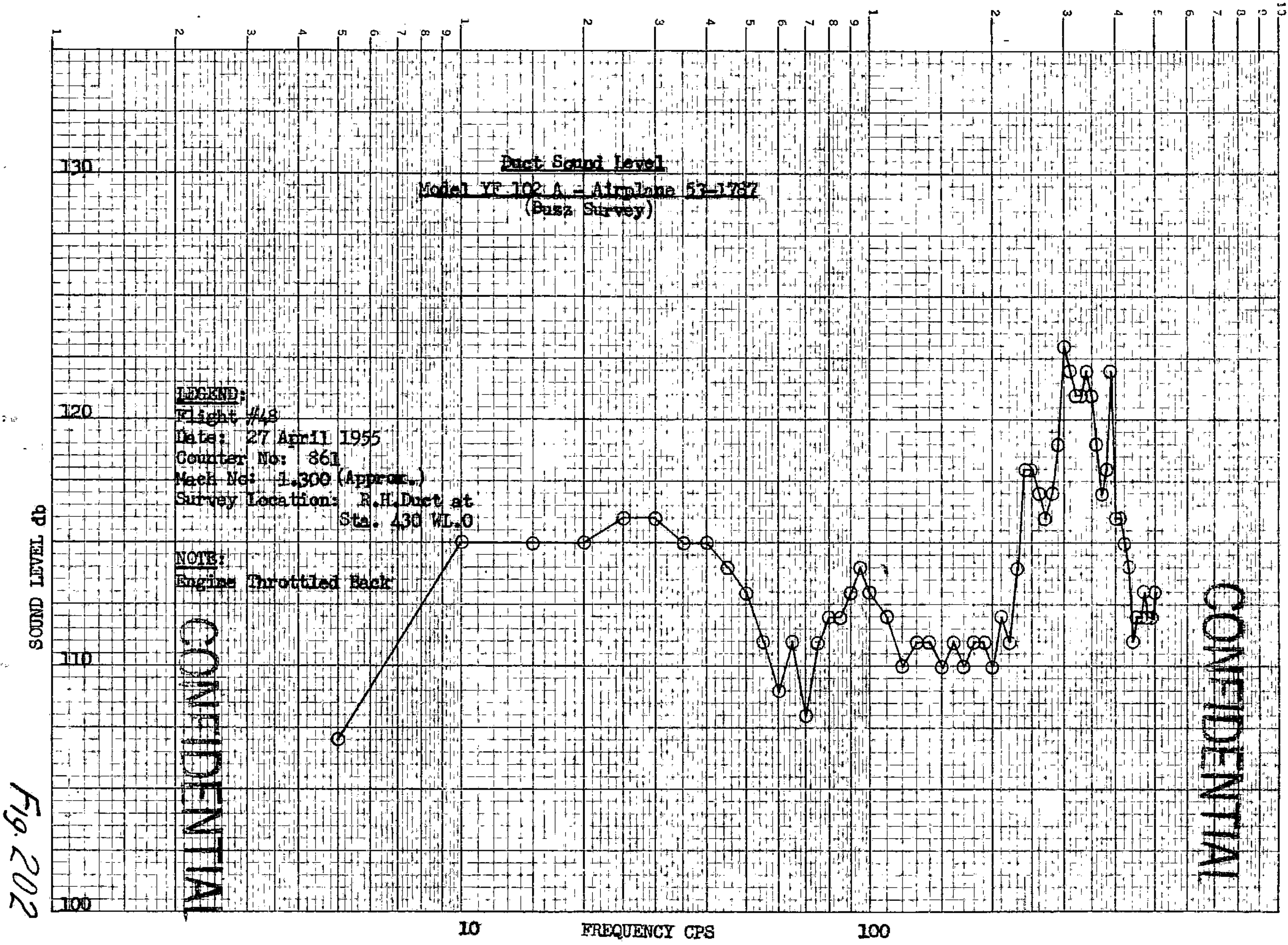
CONFIDENTIAL

Fig. 201

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Sound Level

Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 861
Mach No: 1.300 (Approx.)
Survey Location: R.H.Duct at
Sta. 430 WL.0

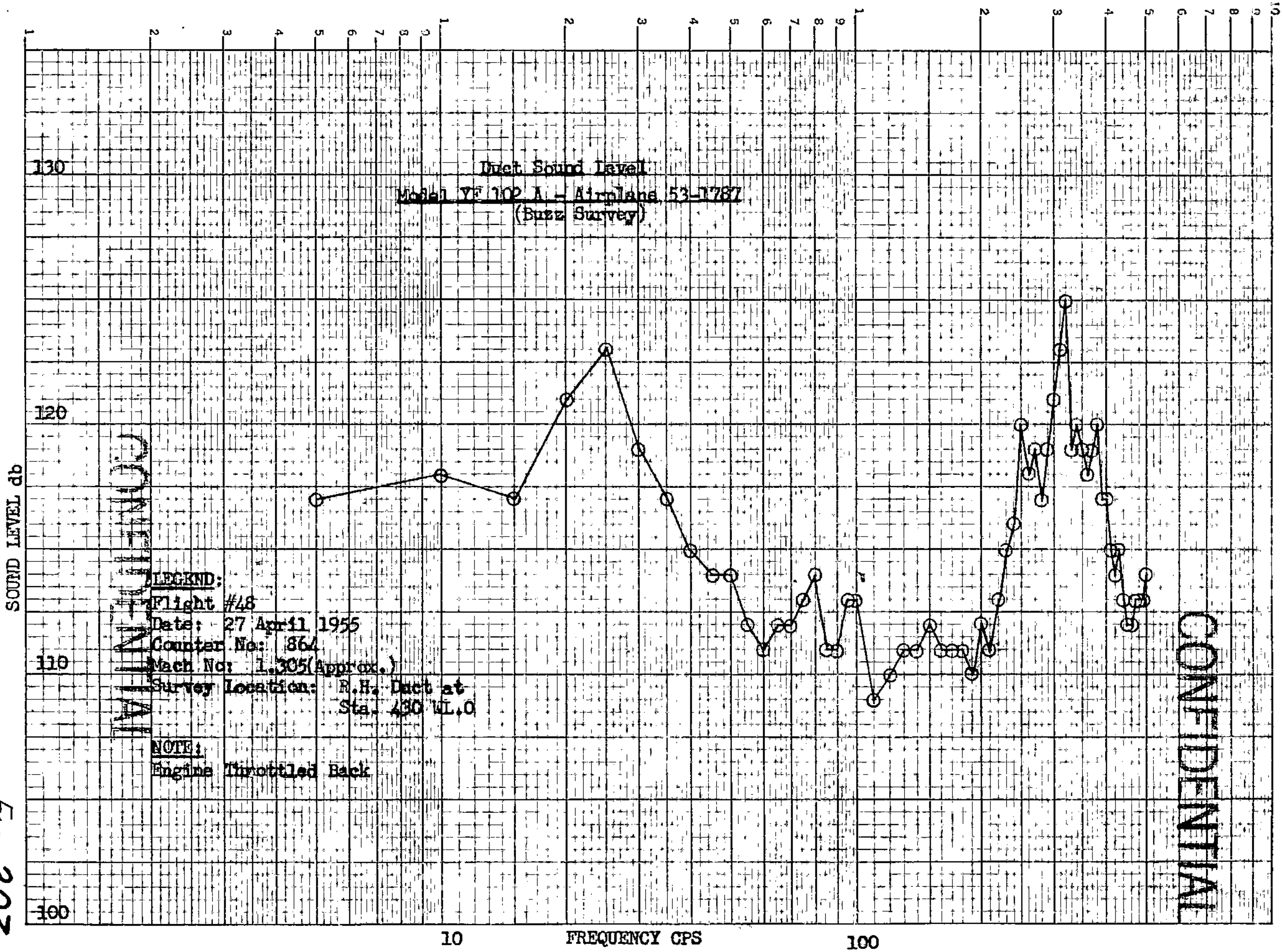
NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 202

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

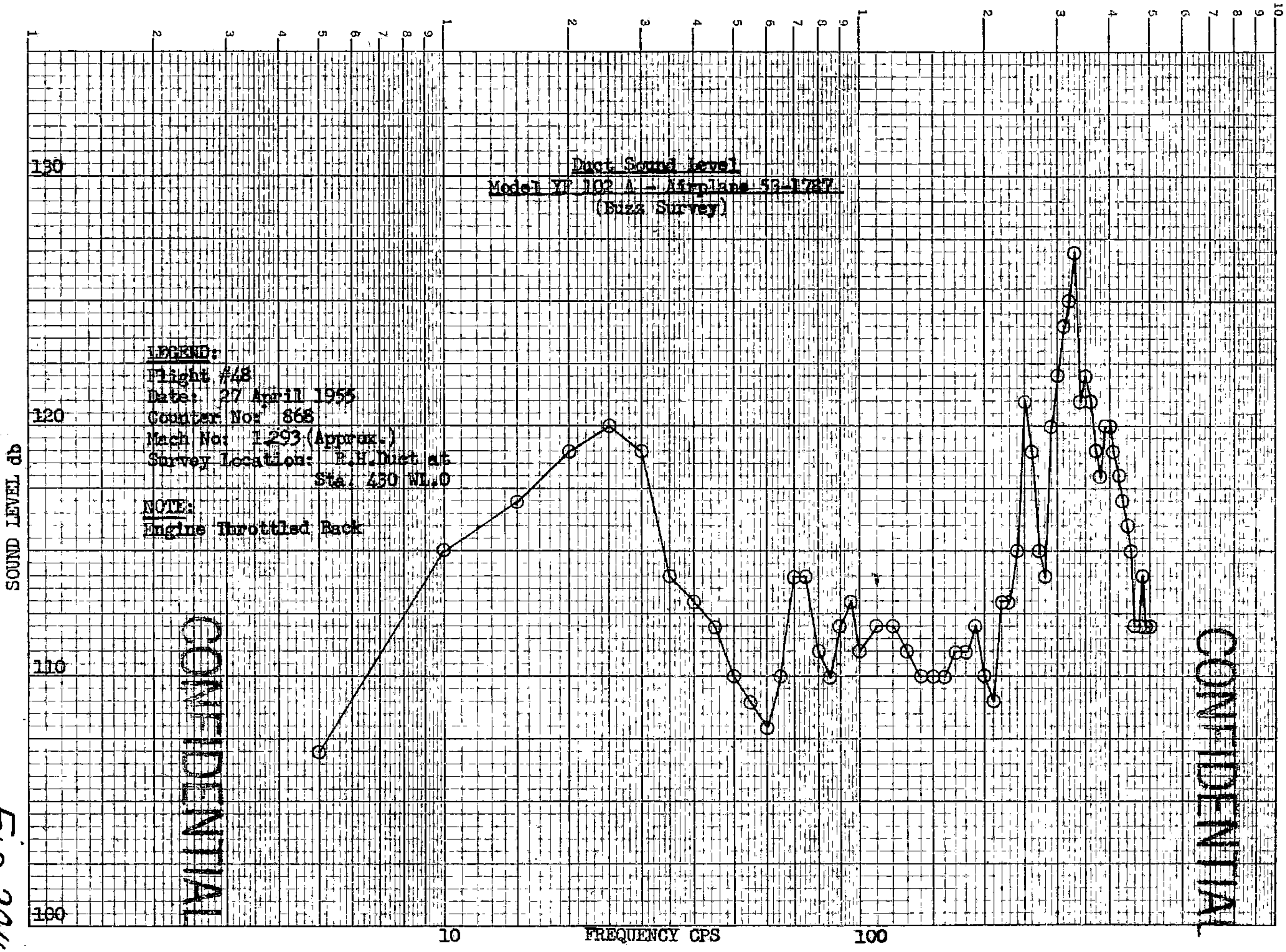
CONFIDENTIAL



CONFIDENTIAL

Fig. 203

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



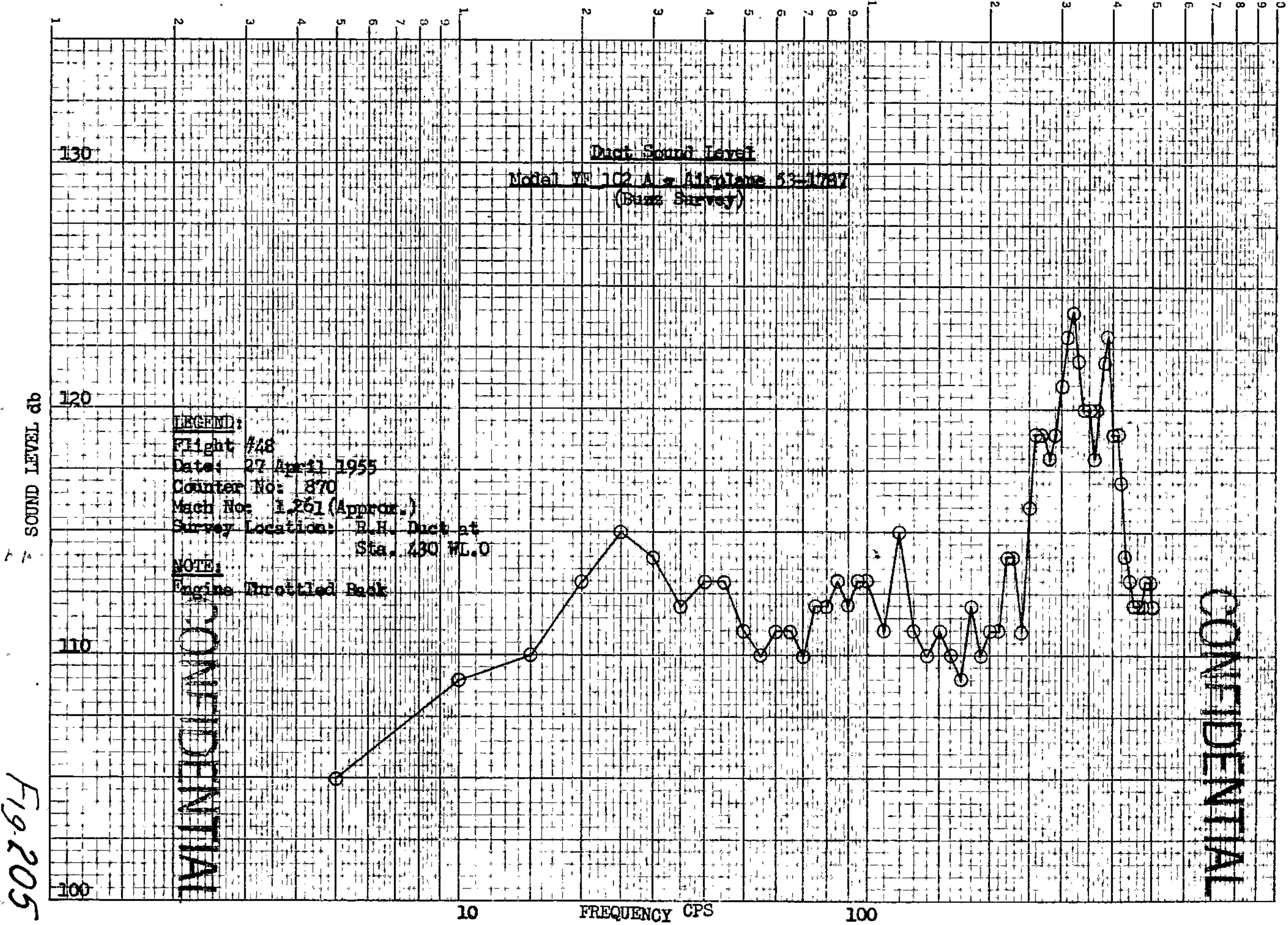
Duct Sound Level
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 868
Mach No: 1.293 (Approx.)
Survey Location: R.H. Duct at
Sta. 430 WL.0

NOTE:
Engine Throttled Back

Fig. 204

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



Duct Sound Level
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

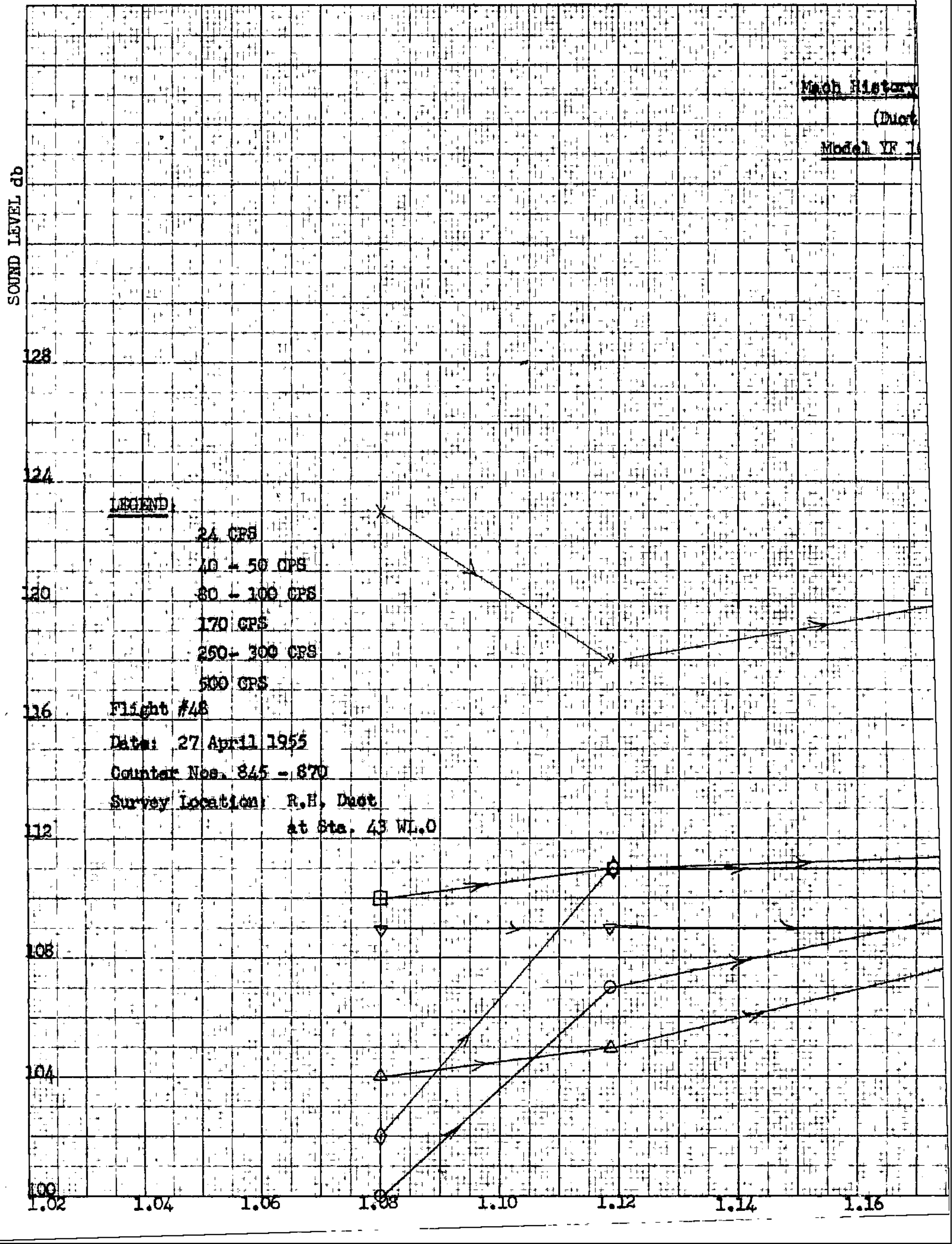
LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 870
Mach No: 1.261 (Approx.)
Survey Location: R.H. Duct at
Sta. 430 WL.0

NOTE:
Engine Throttled Back

CONFIDENTIAL

CONFIDENTIAL

Fig. 205



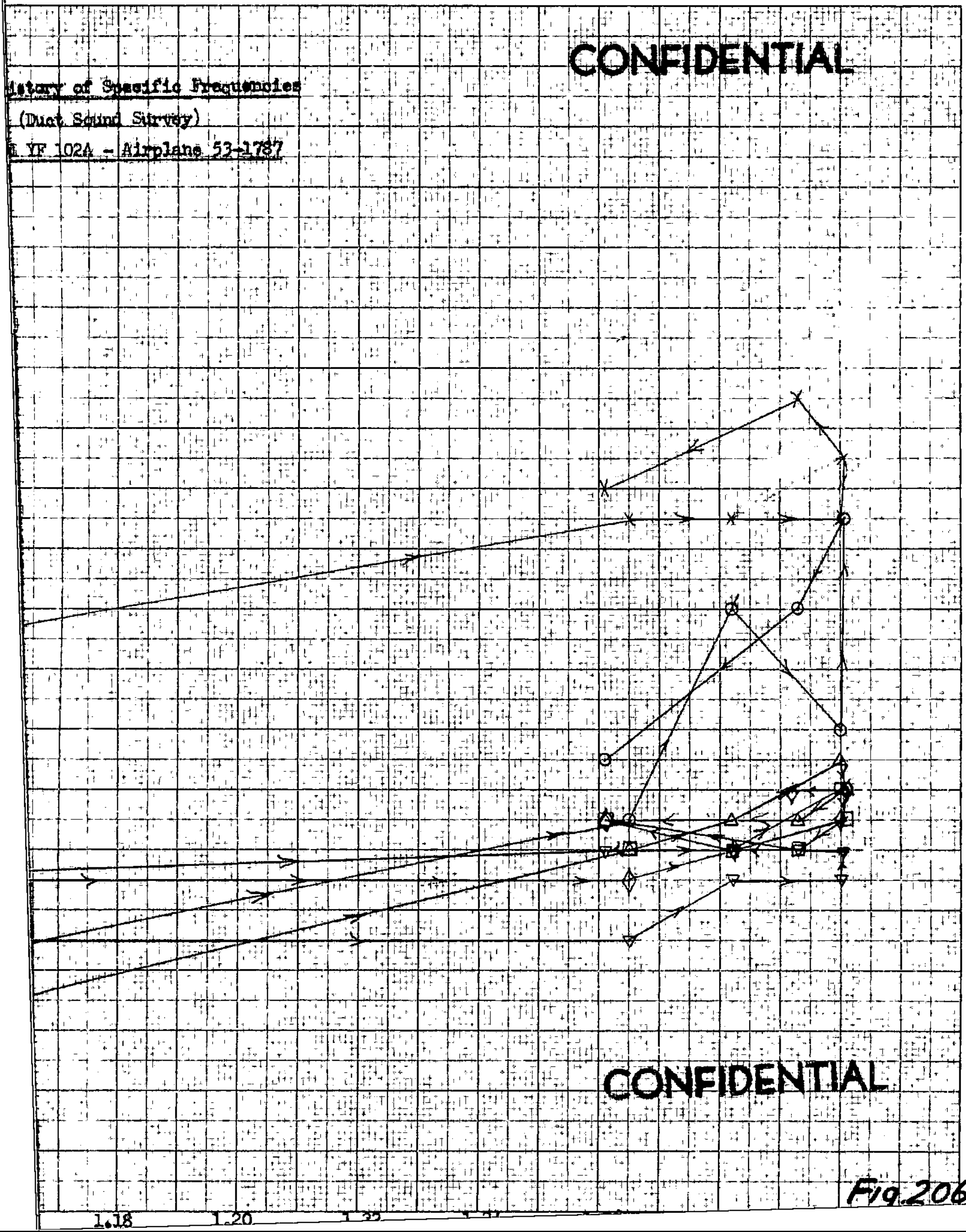
10 X 10 TO THE ½ INCH 359-11L

istory of Specific Frequencies

(Duct Sound Survey)

YF 102A - Airplane 53-1787

CONFIDENTIAL

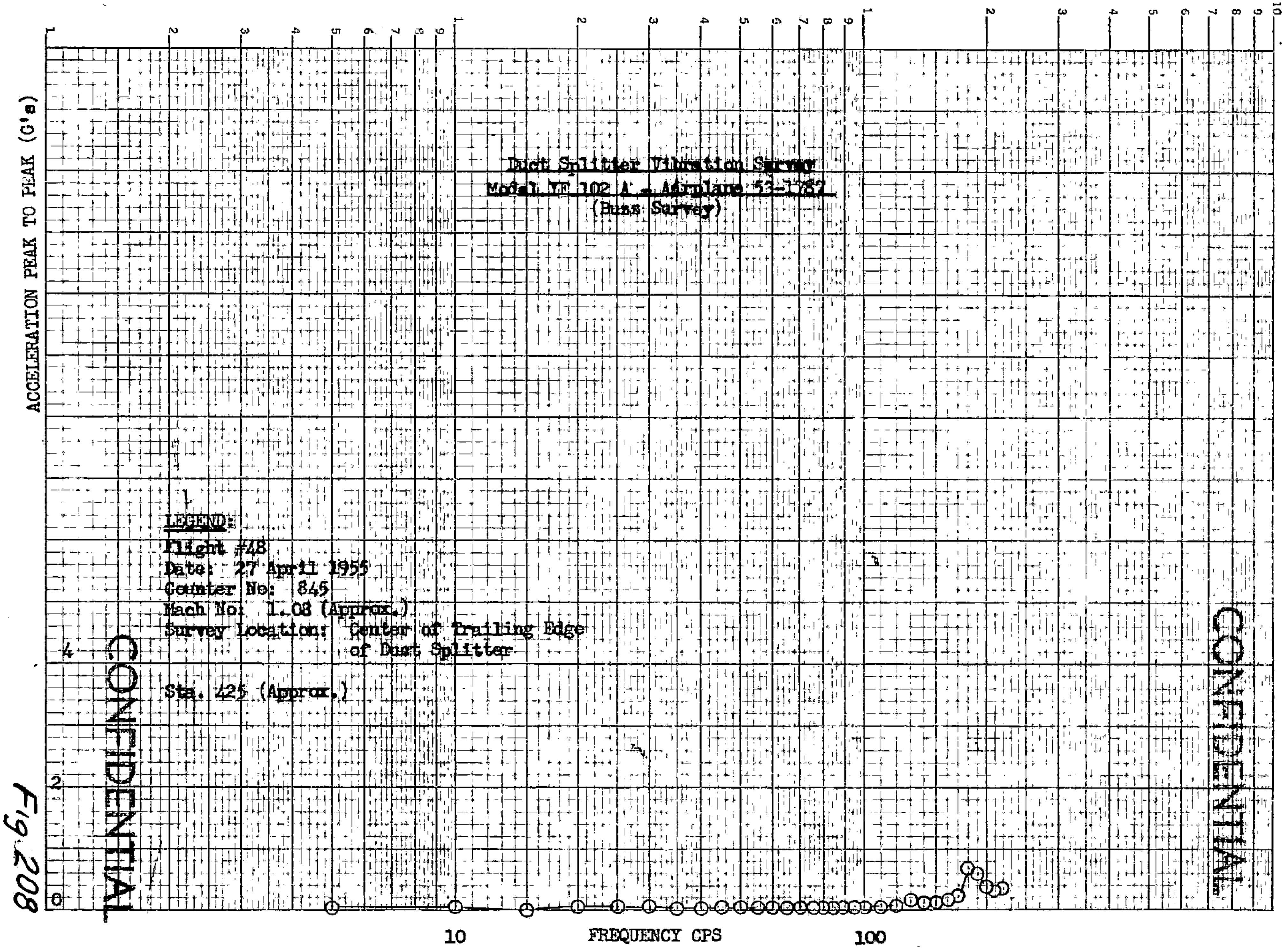


CONFIDENTIAL

Fig. 206

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buss Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 845
Mach No: 1.08 (Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter

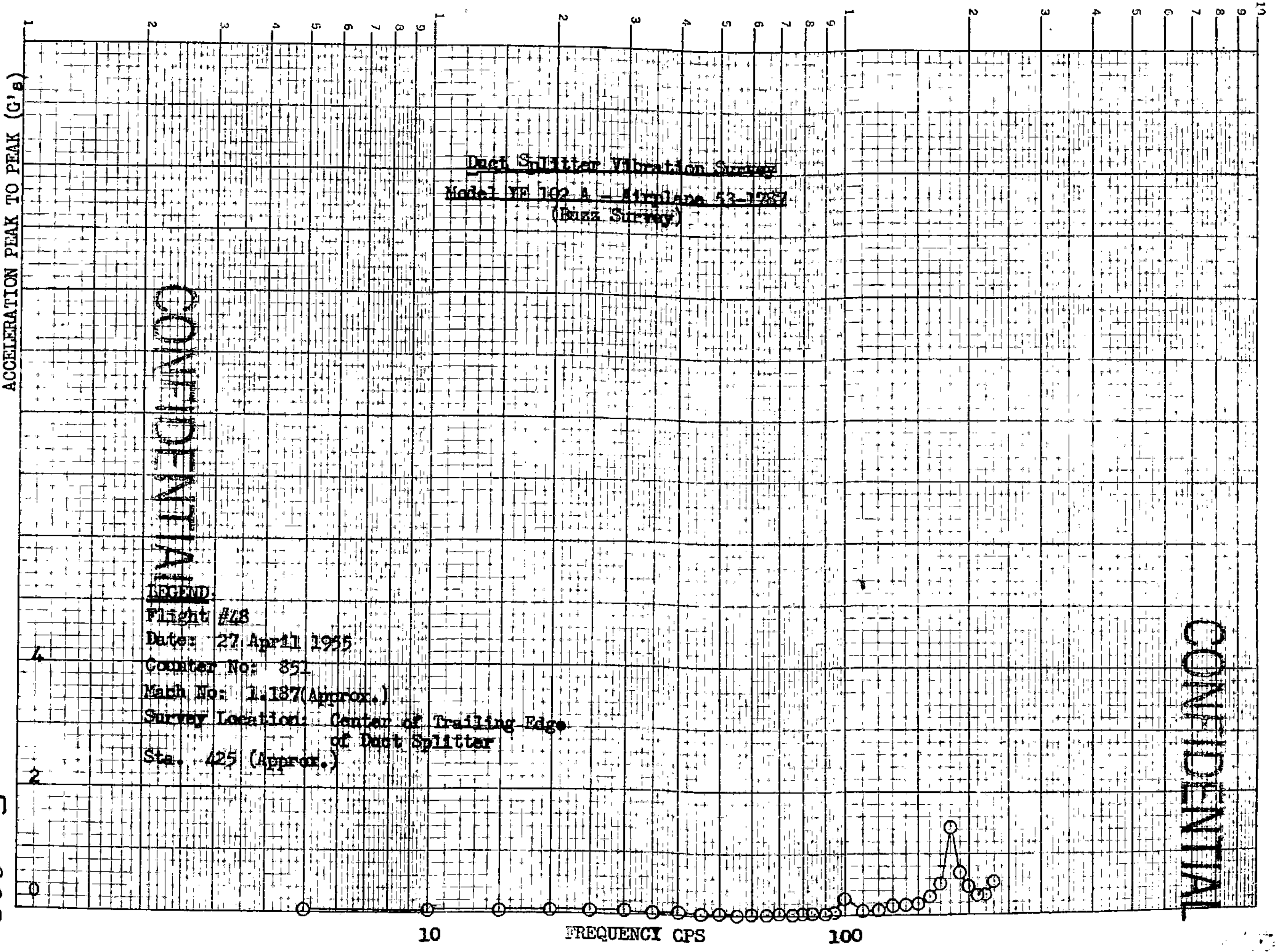
Sta. 425 (Approx.)

CONFIDENTIAL

Fig. 208

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



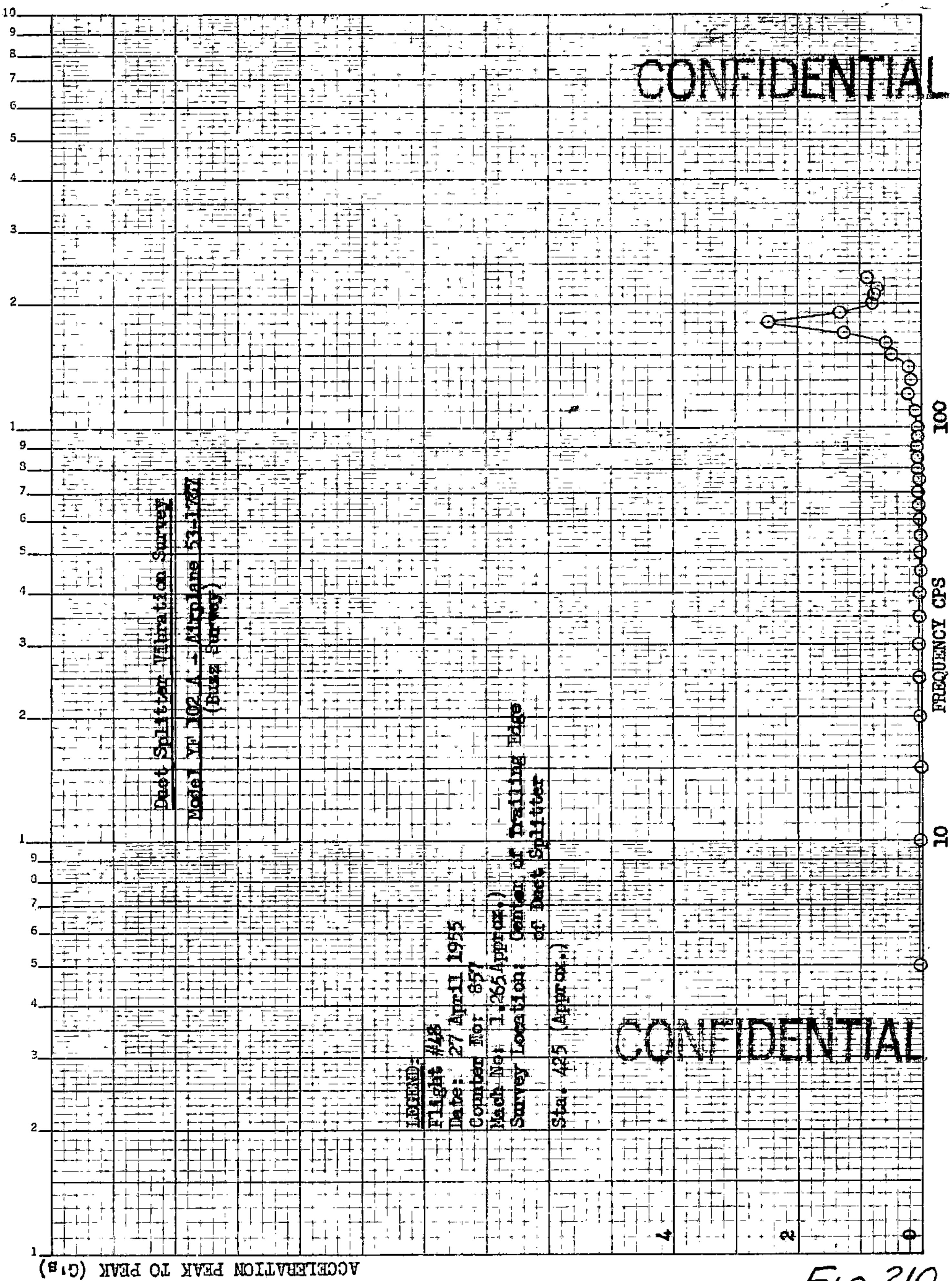
Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 851
Mach No: 1.187(Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta. 425 (Approx.)

CONFIDENTIAL

Fig. 209

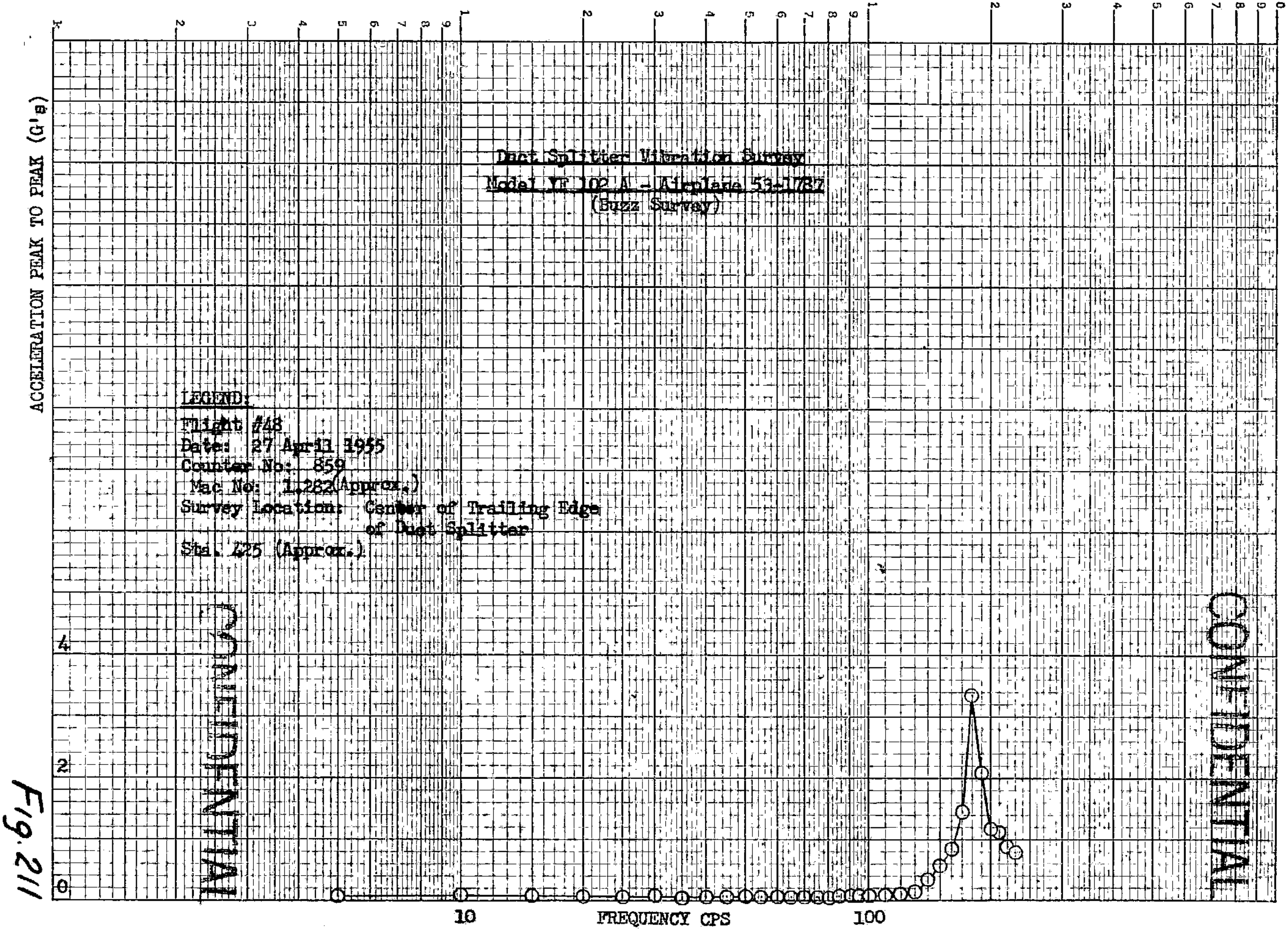
CONFIDENTIAL



CONFIDENTIAL

Fig. 210

CONFIDENTIAL



Duct Splitter Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

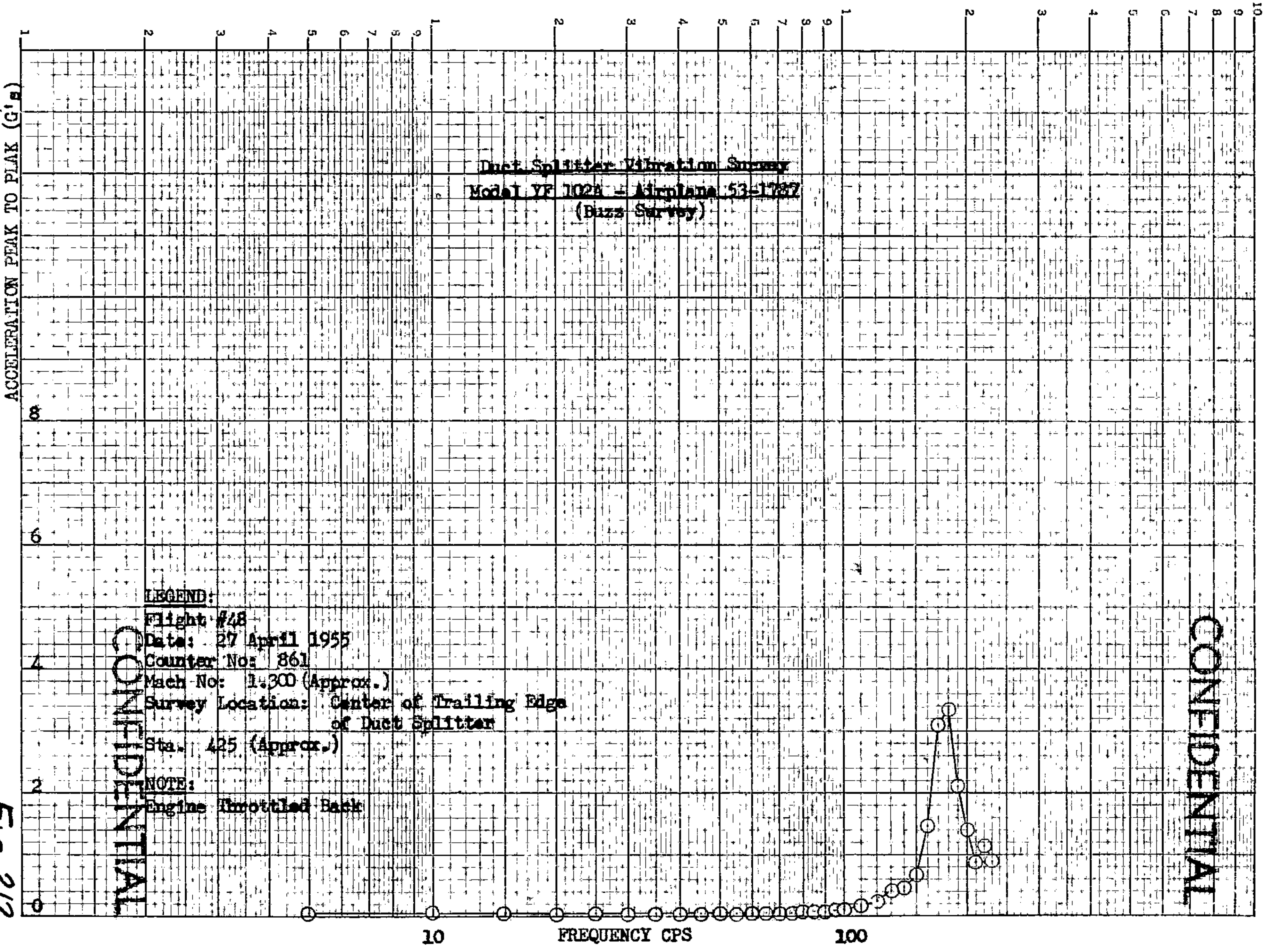
LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 859
Mac No: 1.282(Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta. 425 (Approx.)

CONFIDENTIAL

Fig. 211

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

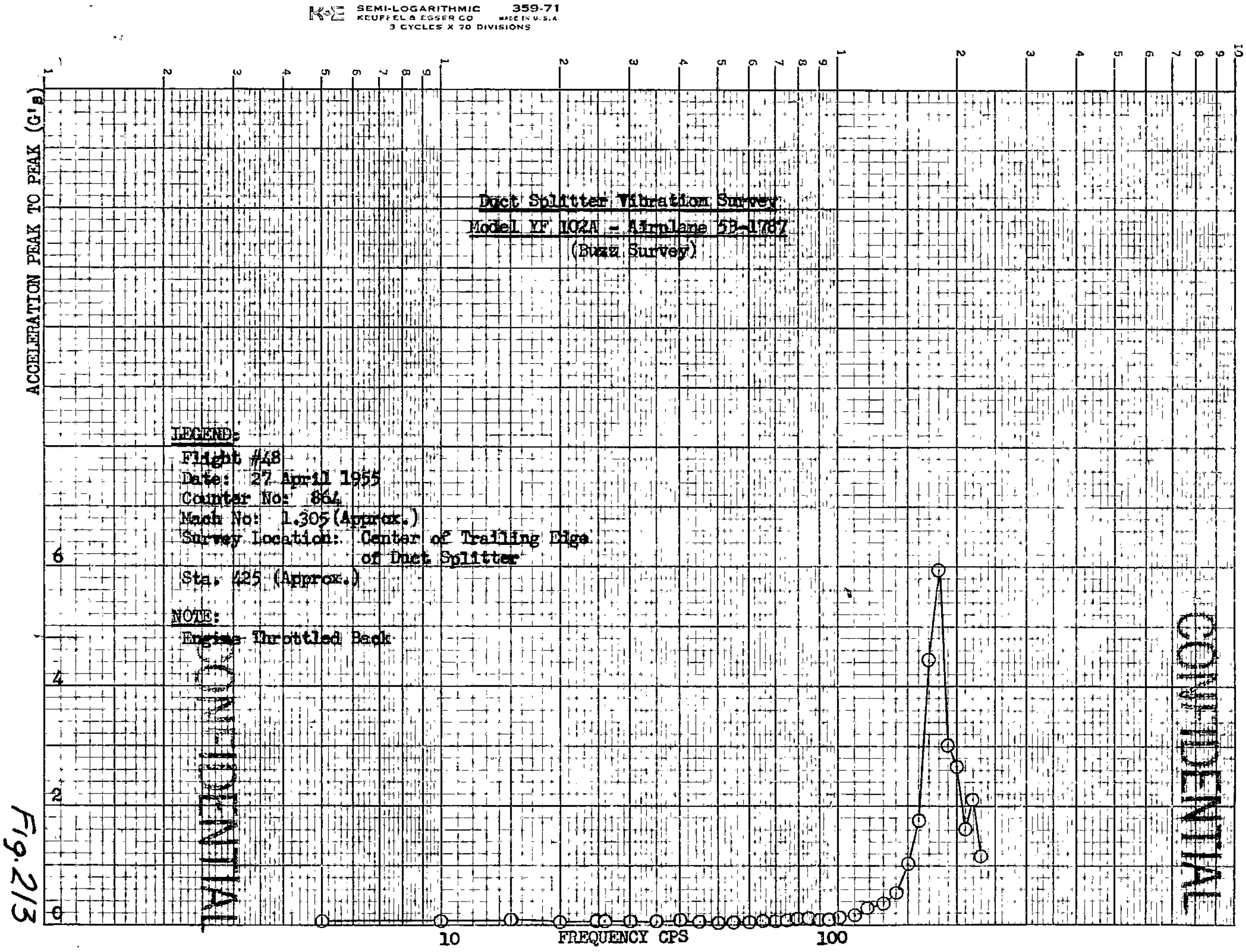
LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 861
Mach No: 1.300 (Approx.)
Survey Location: Center of Trailing Edge of Duct Splitter
Sta. 425 (Approx.)

NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 212

CONFIDENTIAL



Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 864
Mach No: 1.305 (Approx.)
Survey Location: Center of Trailing Edge of Duct Splitter
Sta. 425 (Approx.)

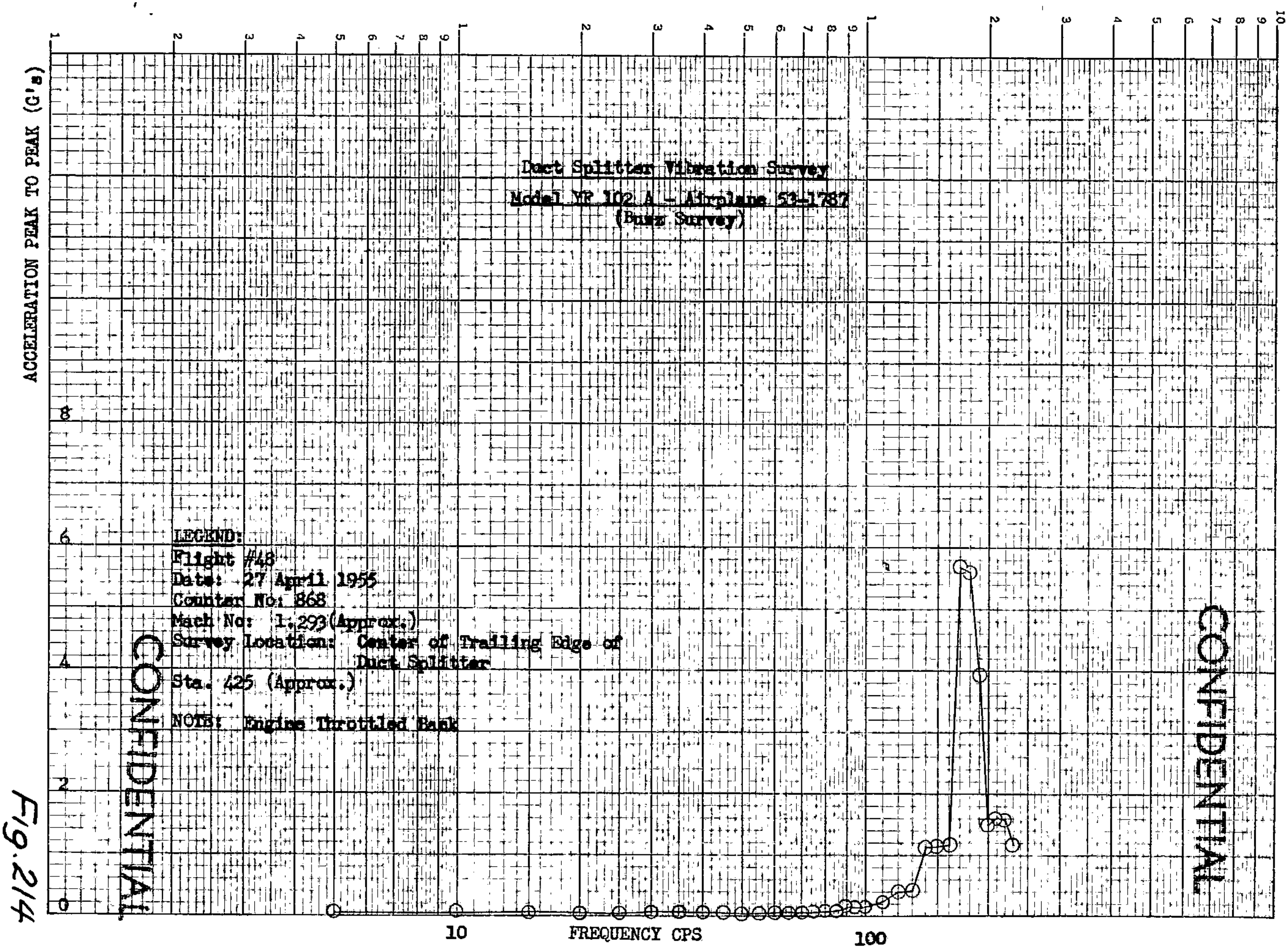
NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 213

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Duct Splitter Vibration Survey

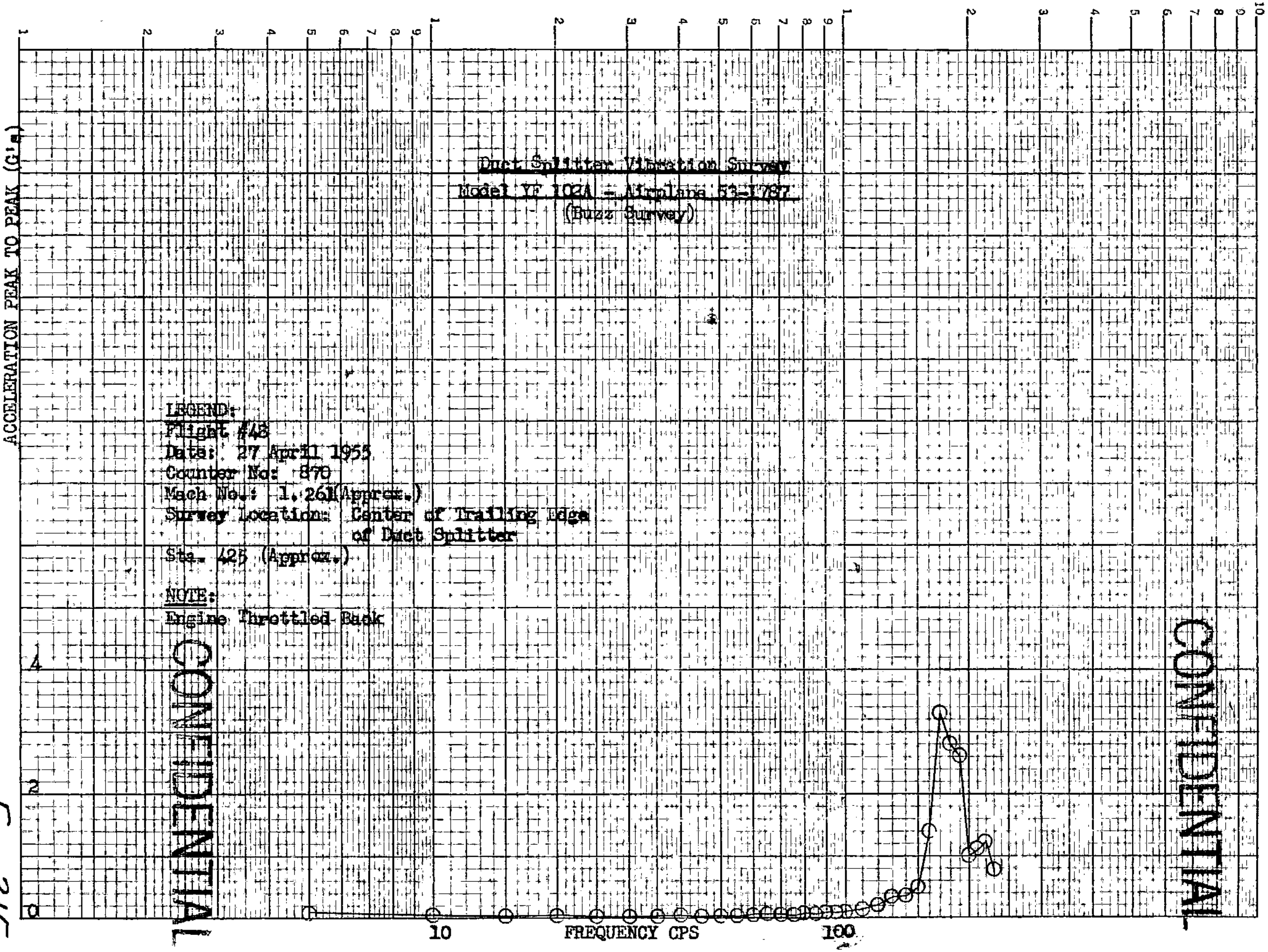
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 868
Mach No: 1.293(Approx.)
Survey Location: Center of Trailing Edge of
Duct Splitter
Sta. 425 (Approx.)
NOTE: Engine Throttled Back

CONFIDENTIAL

Fig. 214

CONFIDENTIAL



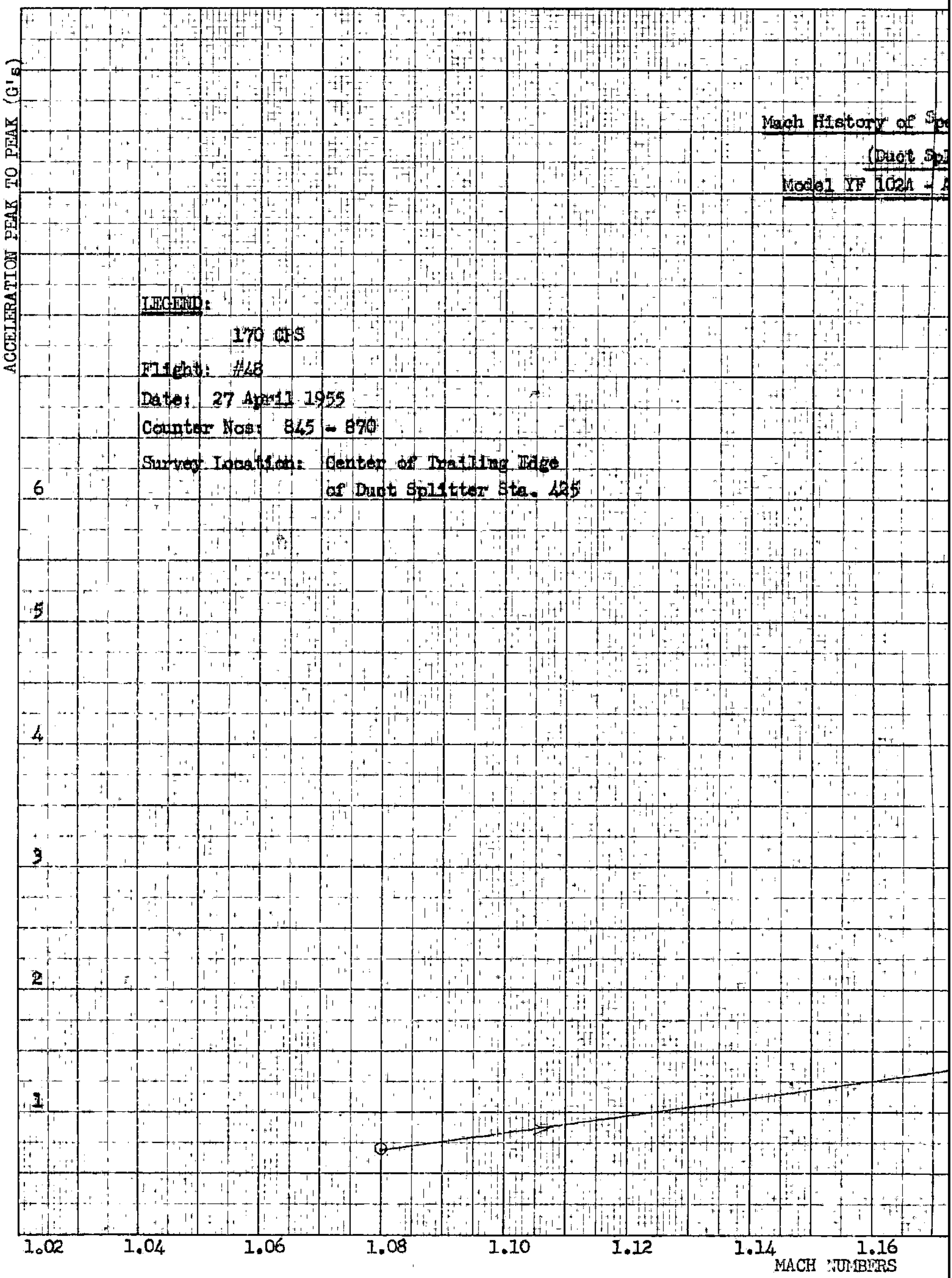
Duct Splitter Vibration Survey
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 870
Mach No.: 1.261(Approx.)
Survey Location: Center of Trailing Edge
of Duct Splitter
Sta. 425 (Approx.)

NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 215



10 X 10 TO THE 1/2 INCH 359-11L

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ry of Specific Frequencies
(Duct Splitter)
102A - Airplane 53-1787

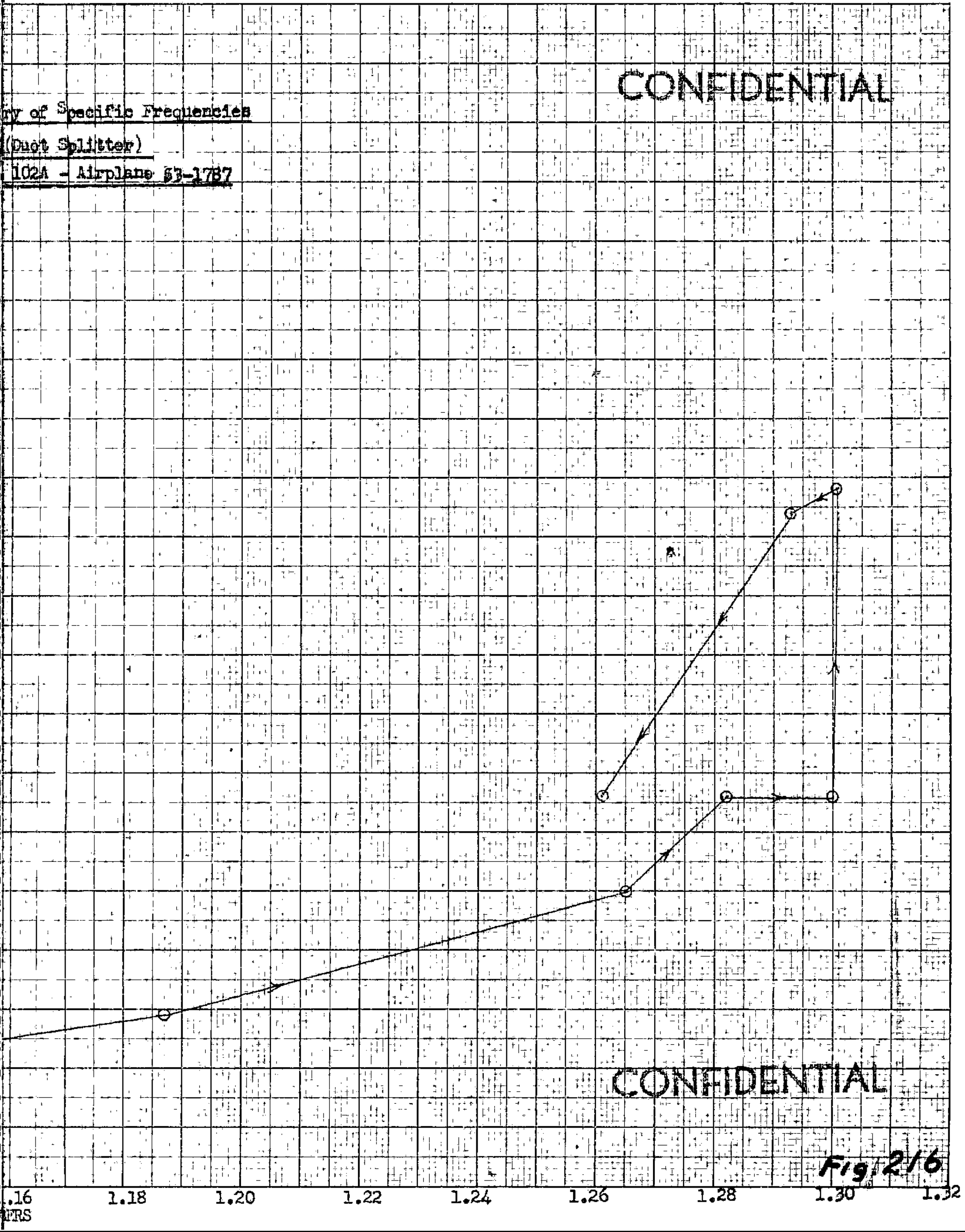

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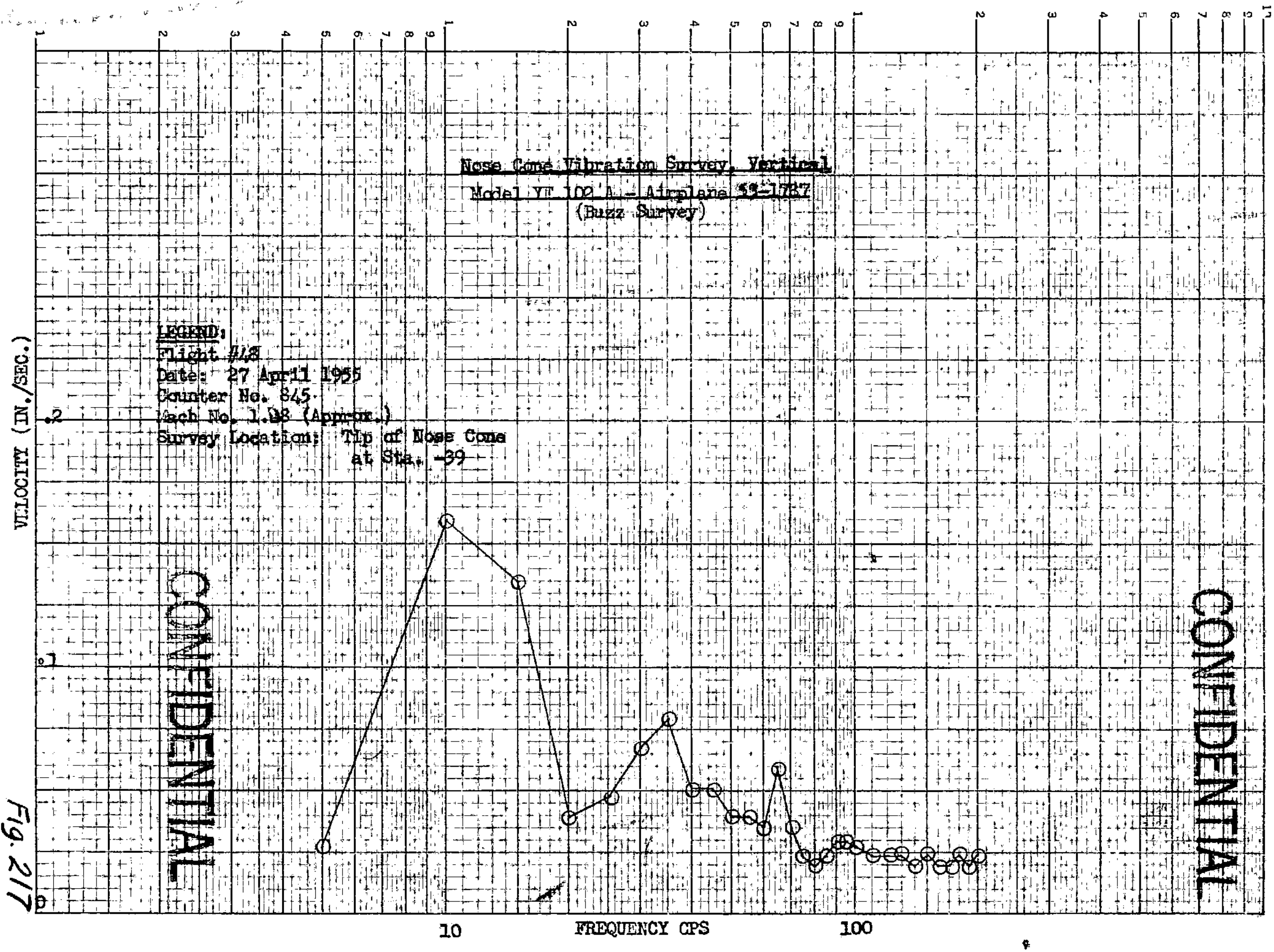
Fig. 2/6

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



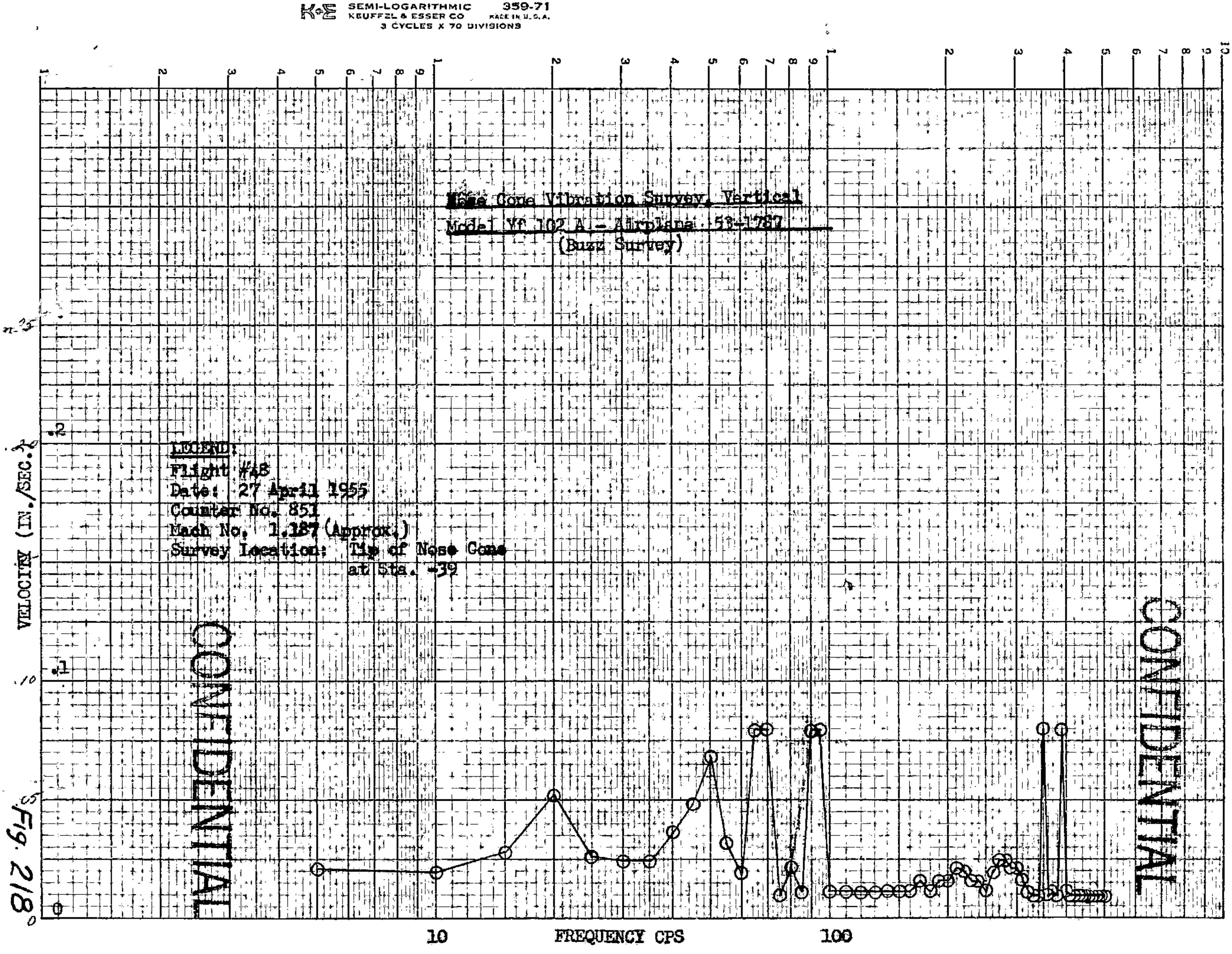
Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No. 845
Mach No. 1.08 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

CONFIDENTIAL

Fig. 217

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

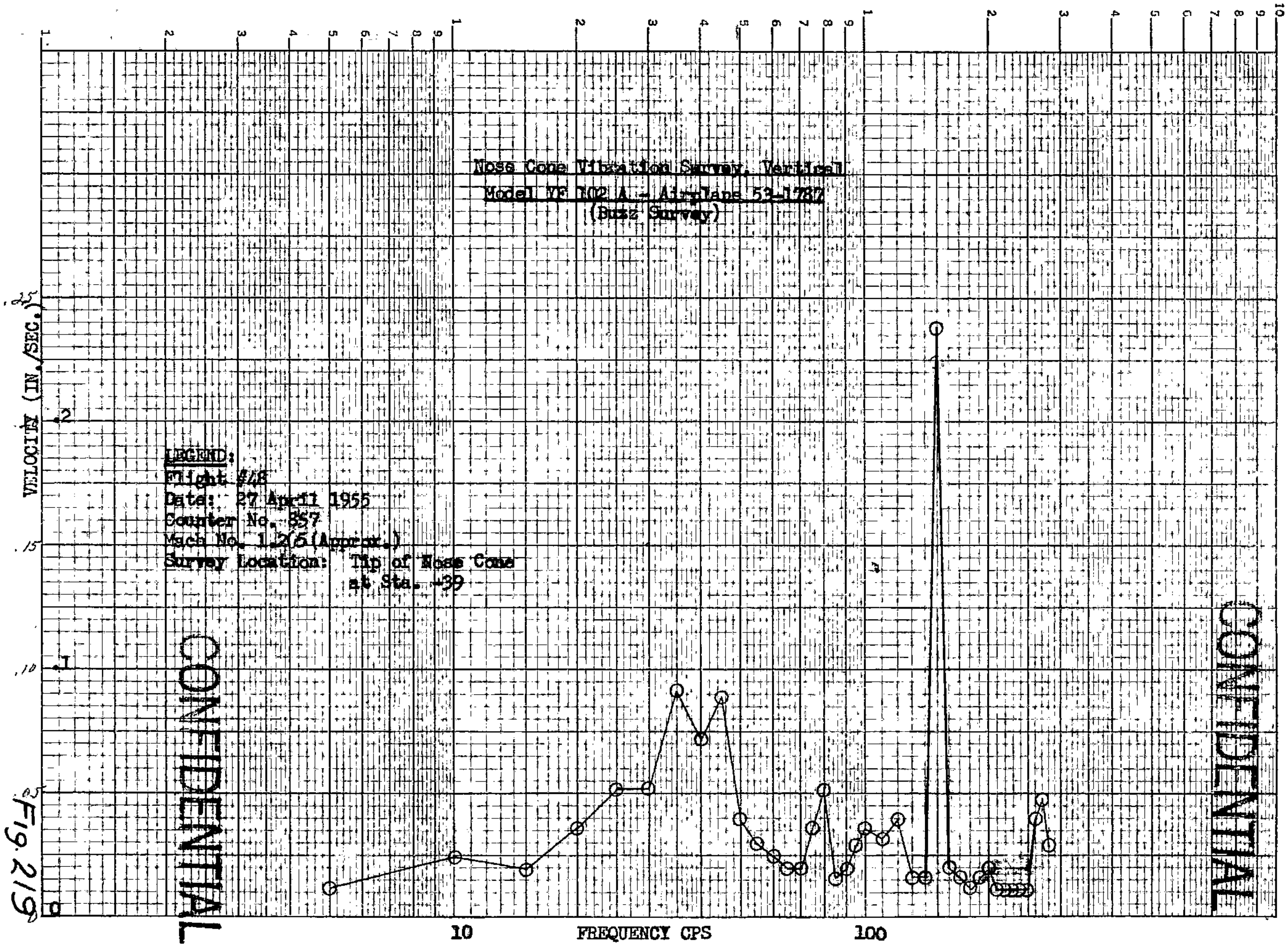
LEGEND:
Flight #48
Date: 27 April 1955
Counter No. 851
Mach No. 1.187 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

CONFIDENTIAL

Fig. 218

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



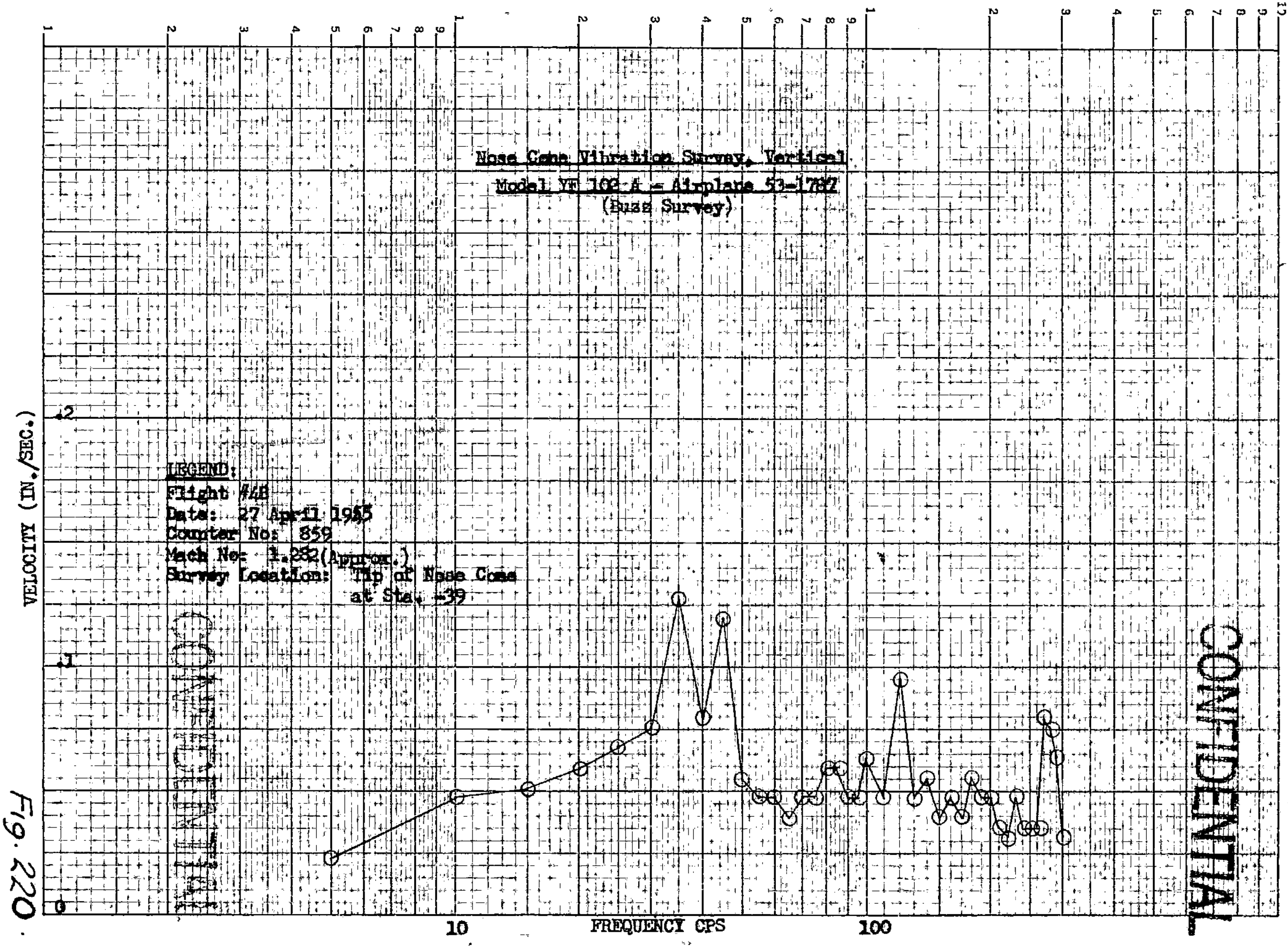
Nose Cone Vibration Survey, Vertical
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No. 857
Mach No. 1.25(Approx.)
Survey Location: Tip of Nose Cone at Sta. -39

Fig. 219

CONFIDENTIAL

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical

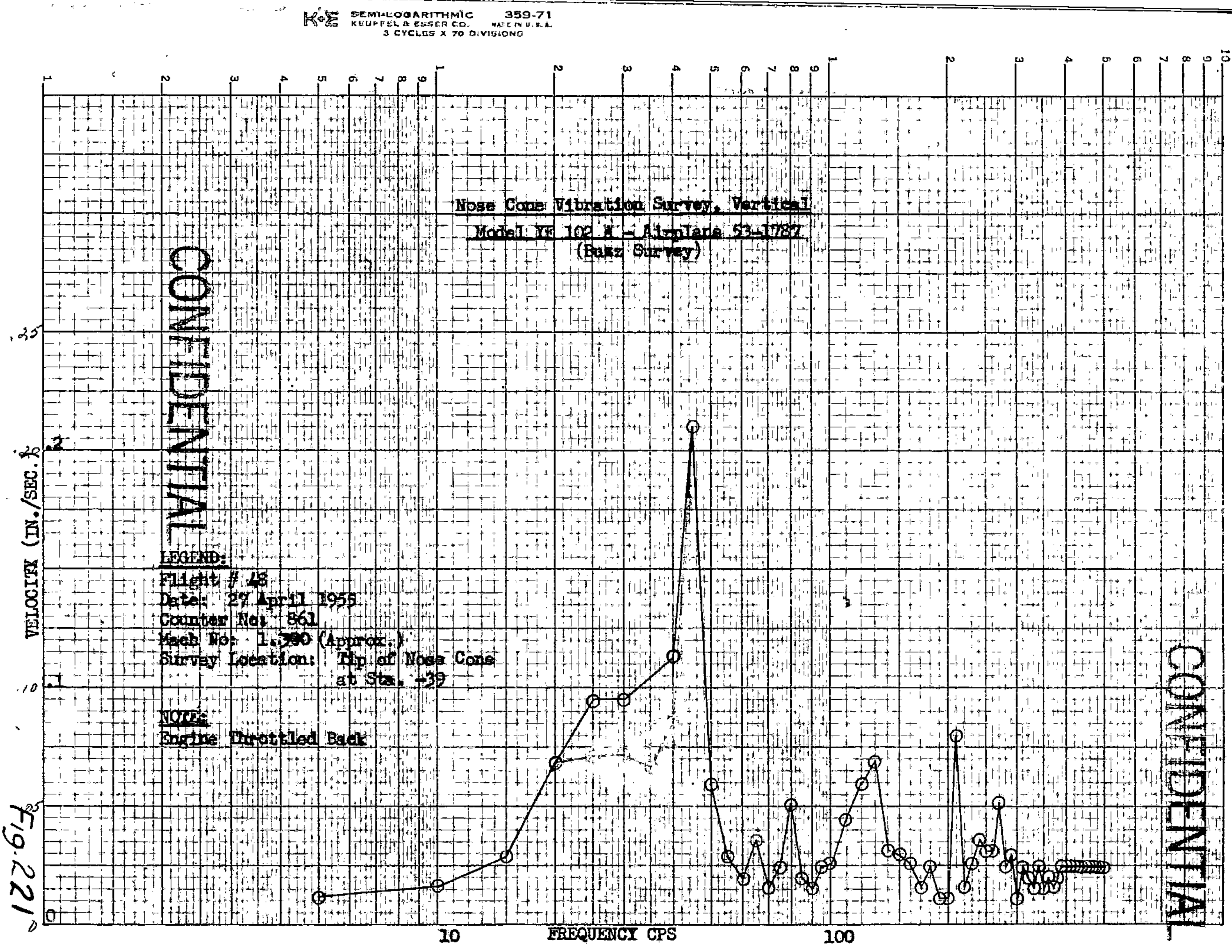
Model YF 102-A - Airplane 53-1787

(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 859
Mach No: 1.282 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

CONFIDENTIAL

Fig. 220.



Nose Cone Vibration Survey, Vertical

Model YF 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND:

Flight # 48
Date: 27 April 1955
Counter No. 561
Mach No. 1.380 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

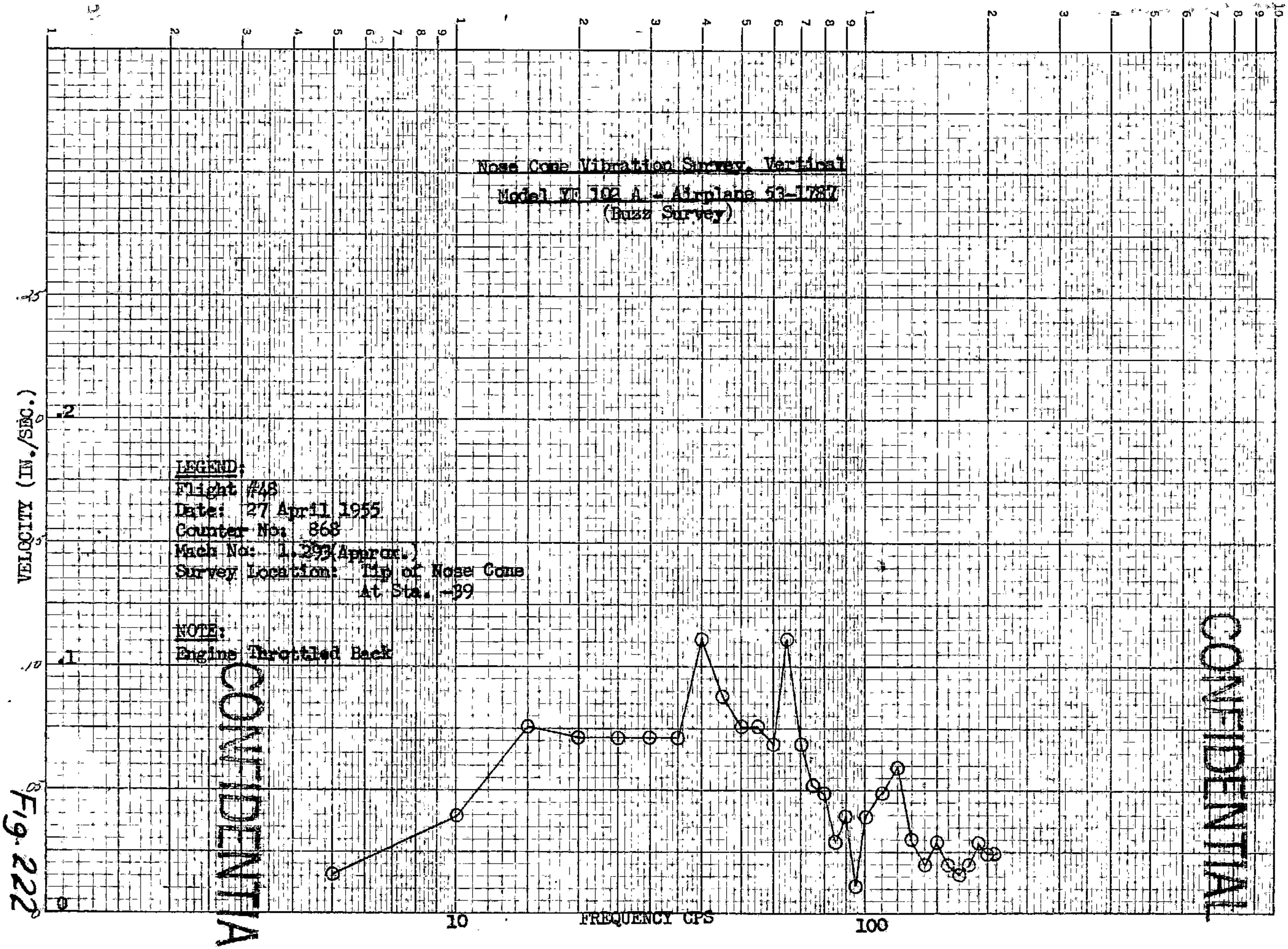
NOTE:

Engine Throttled Back

Fig. 221

K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical

Model YF 102 A - Airplane 53-1787

(Buzz Survey)

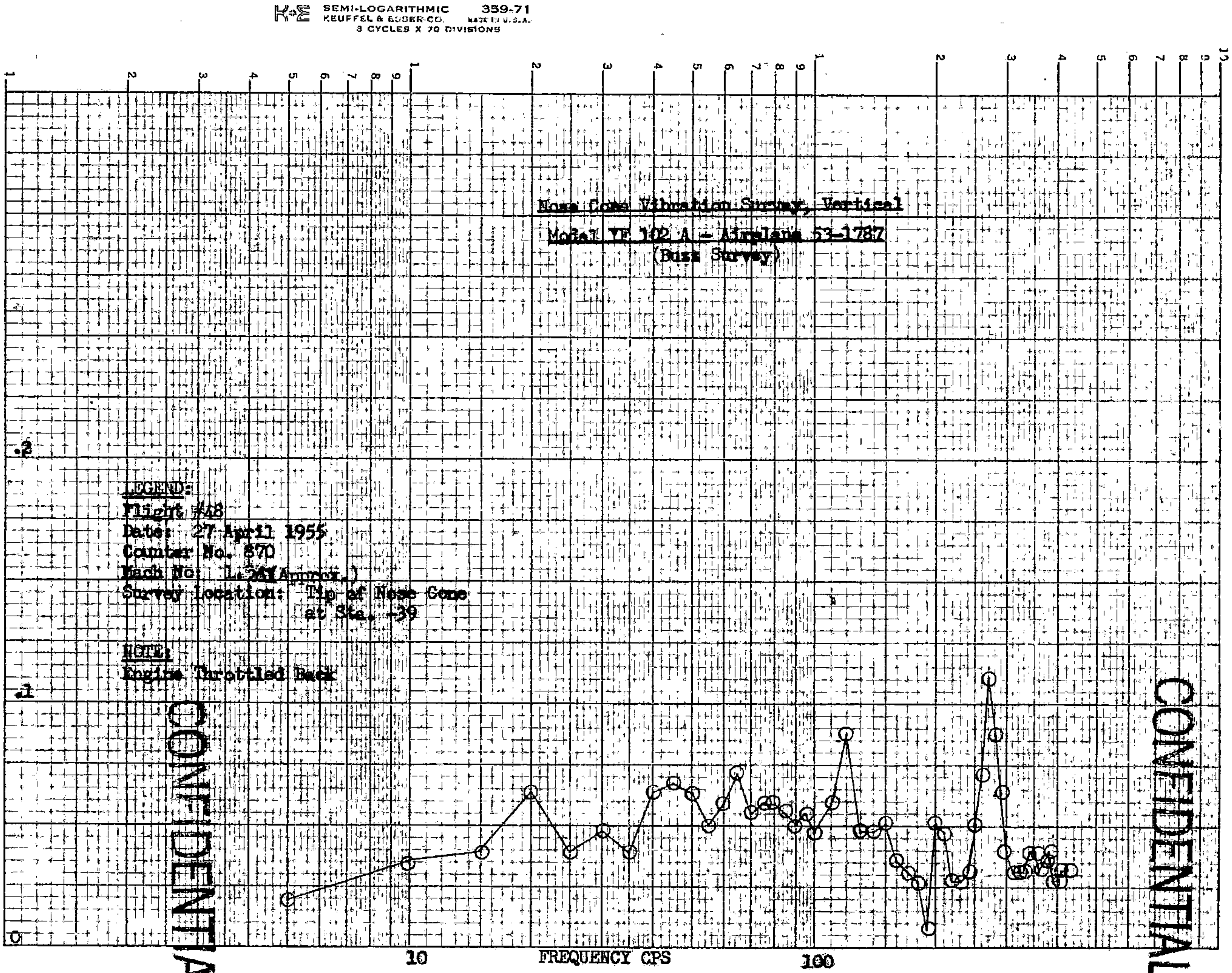
LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 868
Mach No: 1.293(Approx.)
Survey Location: Tip of Nose Cone
At Sta. -39

NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 222

CONFIDENTIAL



Nose Cone Vibration Survey, Vertical

Model YF 102 A – Airplane 53-1787

(Buzz Survey)

LEGEND:

Flight #48

Date: 27 April 1955

Counter No. 870

Mach No: 1.26 (Approx.)

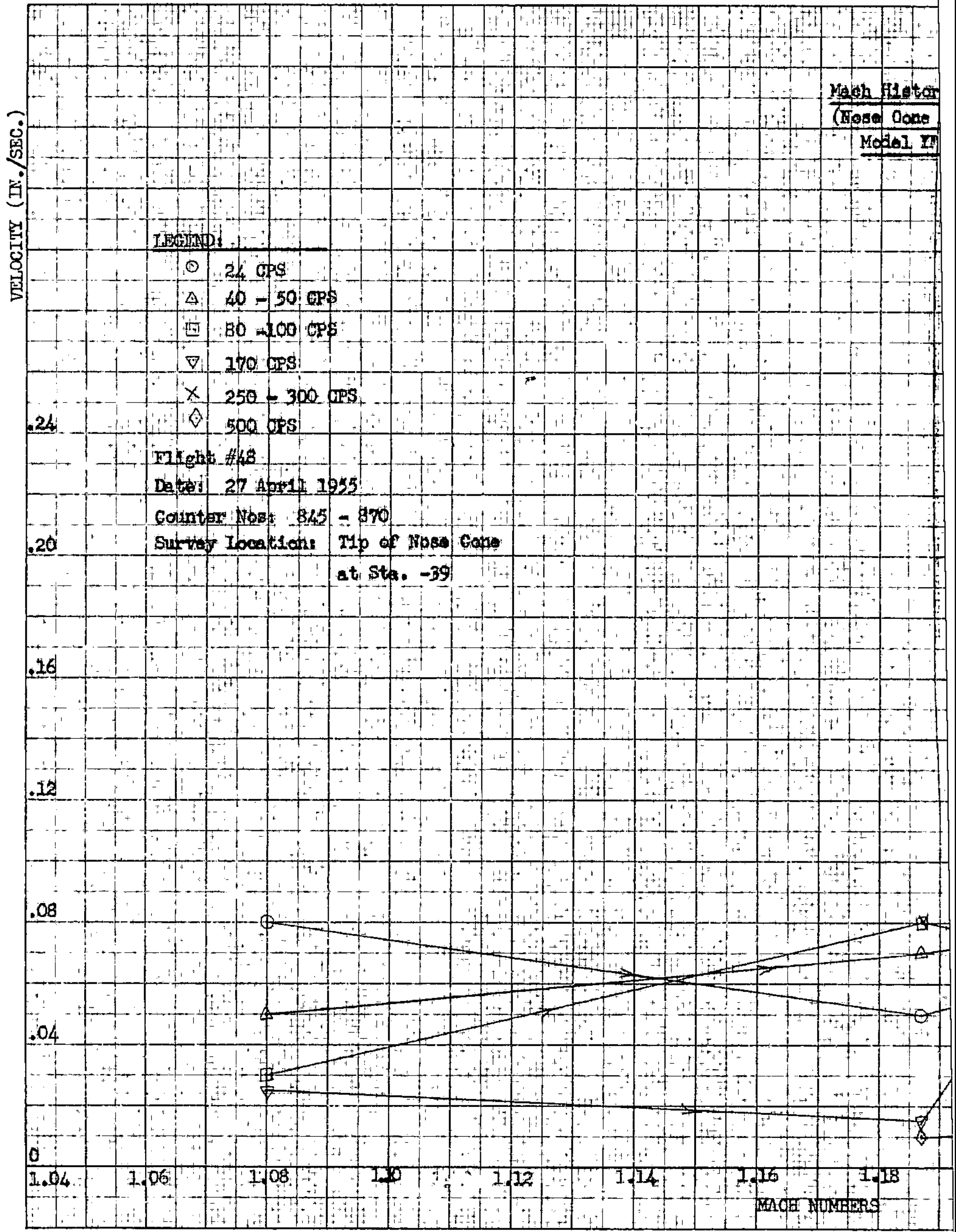
Survey Location: Tip of Nose Cone
at Sta. -39

NOTE:

Engine Throttled Back

CONFIDENTIAL

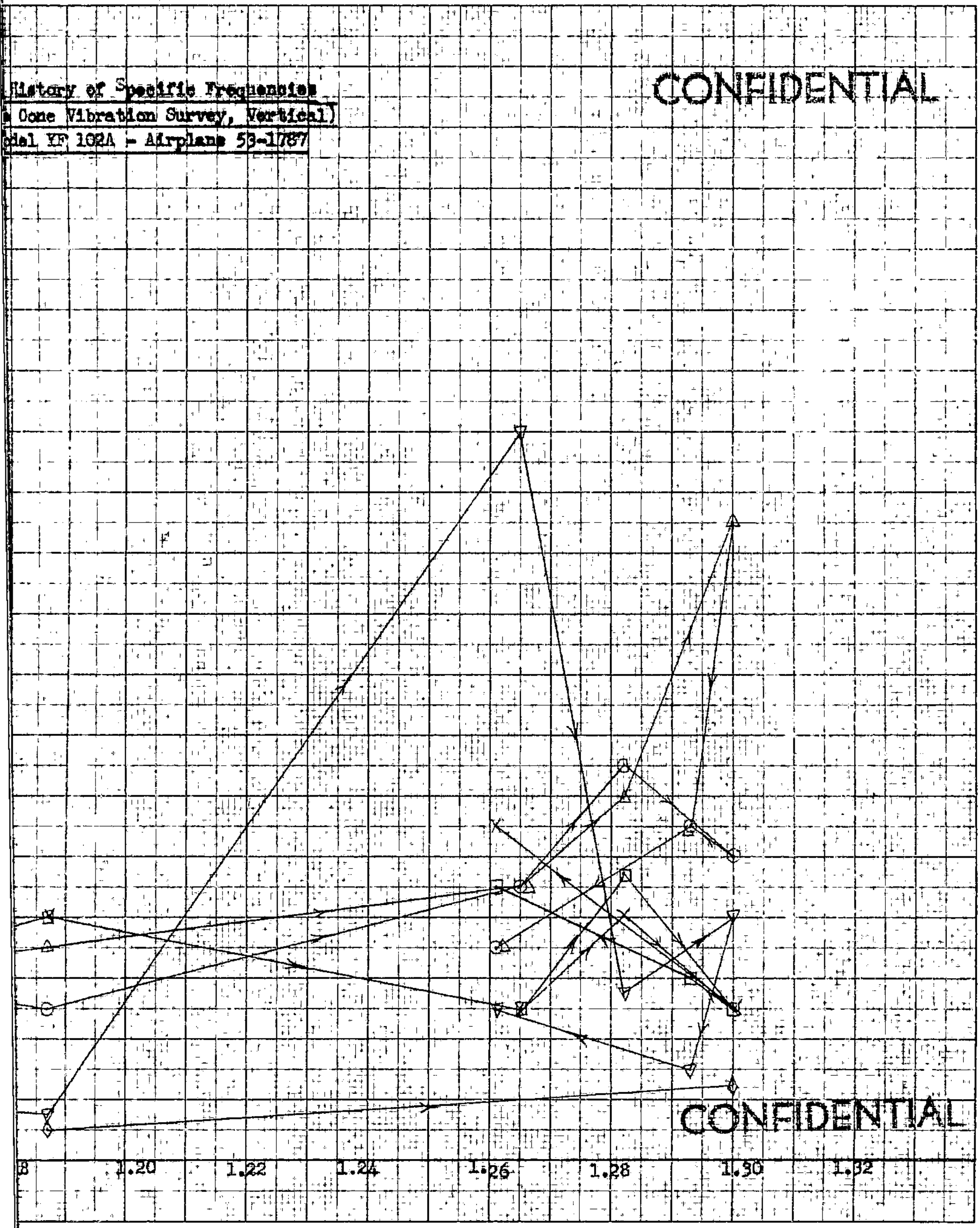
Fig. 223



K&E 10 X 10 TO THE ½ INCH 359-11L
KEUFFEL & ESSER CO.

CONFIDENTIAL

istory of Specific Frequencies
Cone Vibration Survey, Vertical)
odel YF 102A - Airplane 53-1787



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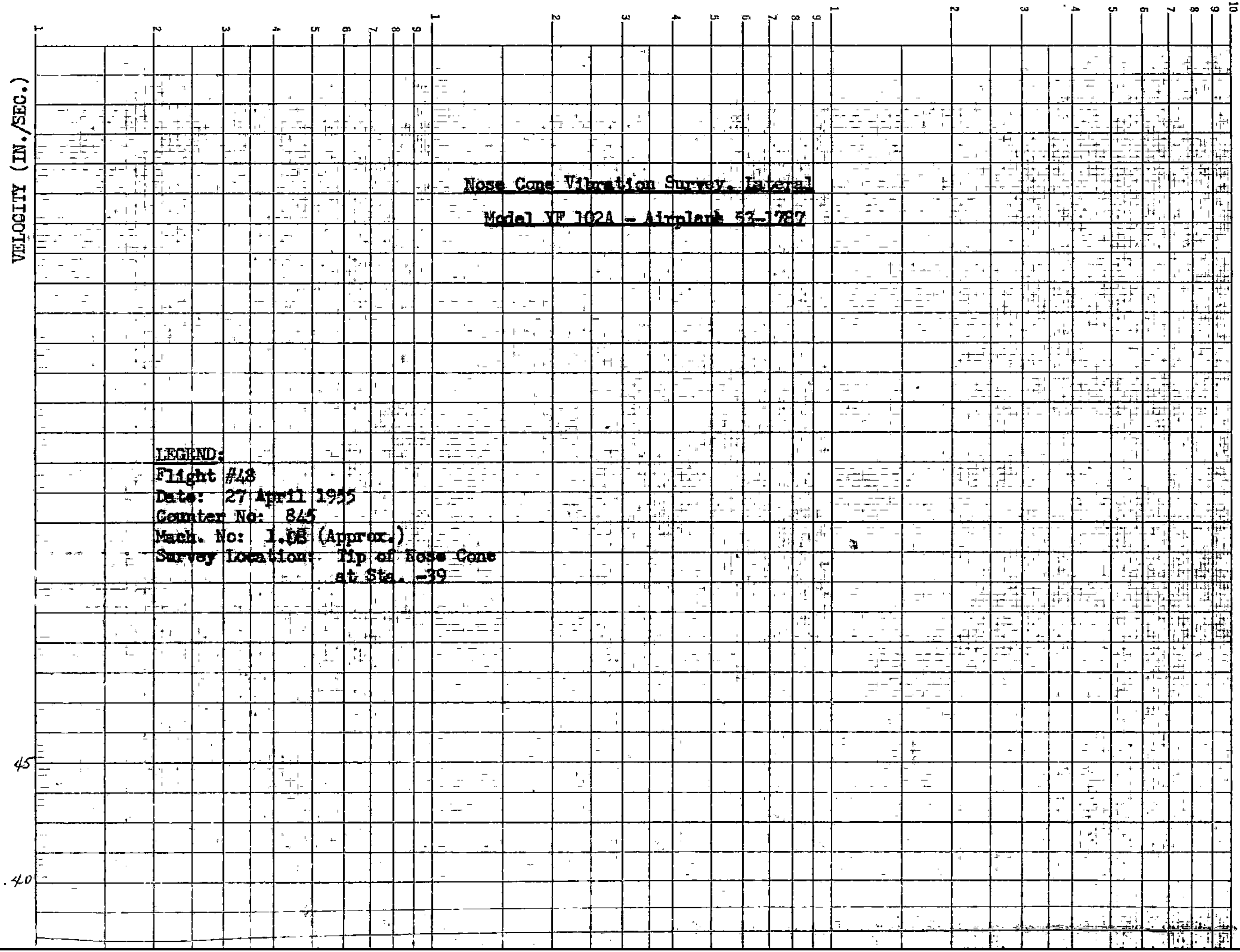
359-73L KEUFFEL & ESSER CO.
Semi-Logarithmic, 3 Cycles × 10 to the ½ inch.

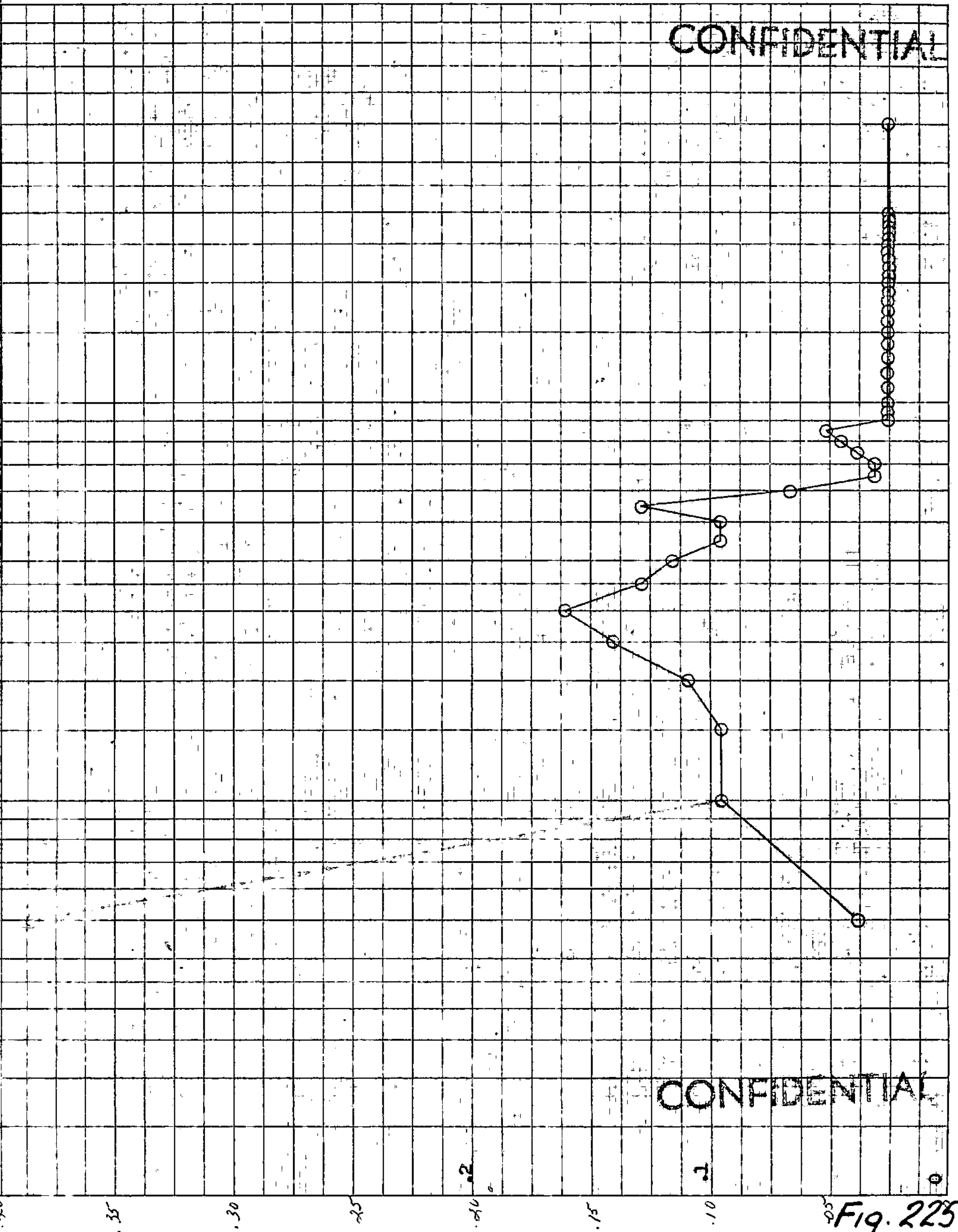
Nose Cone Vibration Survey, Lateral

Model XF 102A - Airplane 53-1787



LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 845
Mach. No: 1.08 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

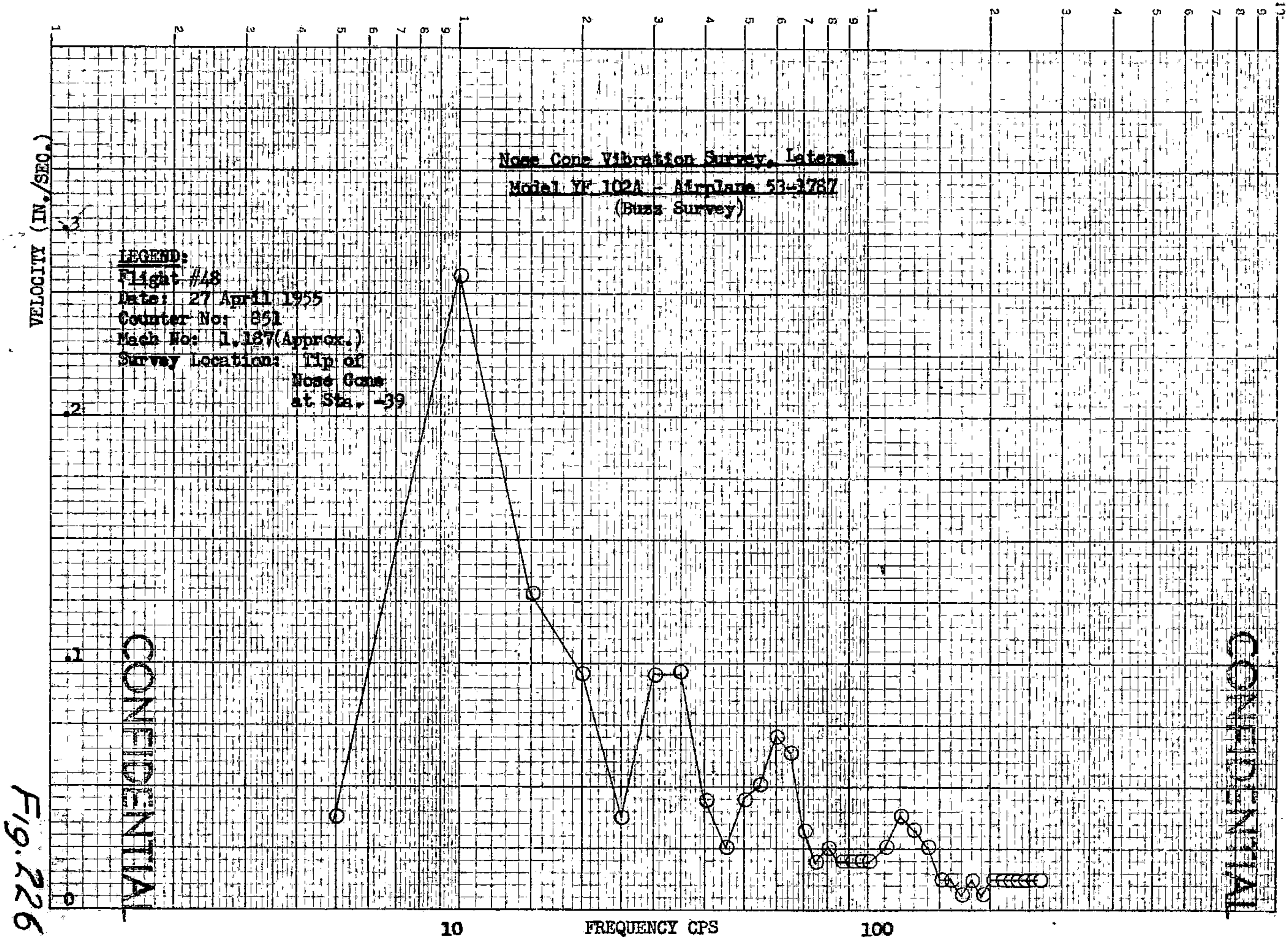
CONFIDENTIAL



CONFIDENTIAL

Fig. 225

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

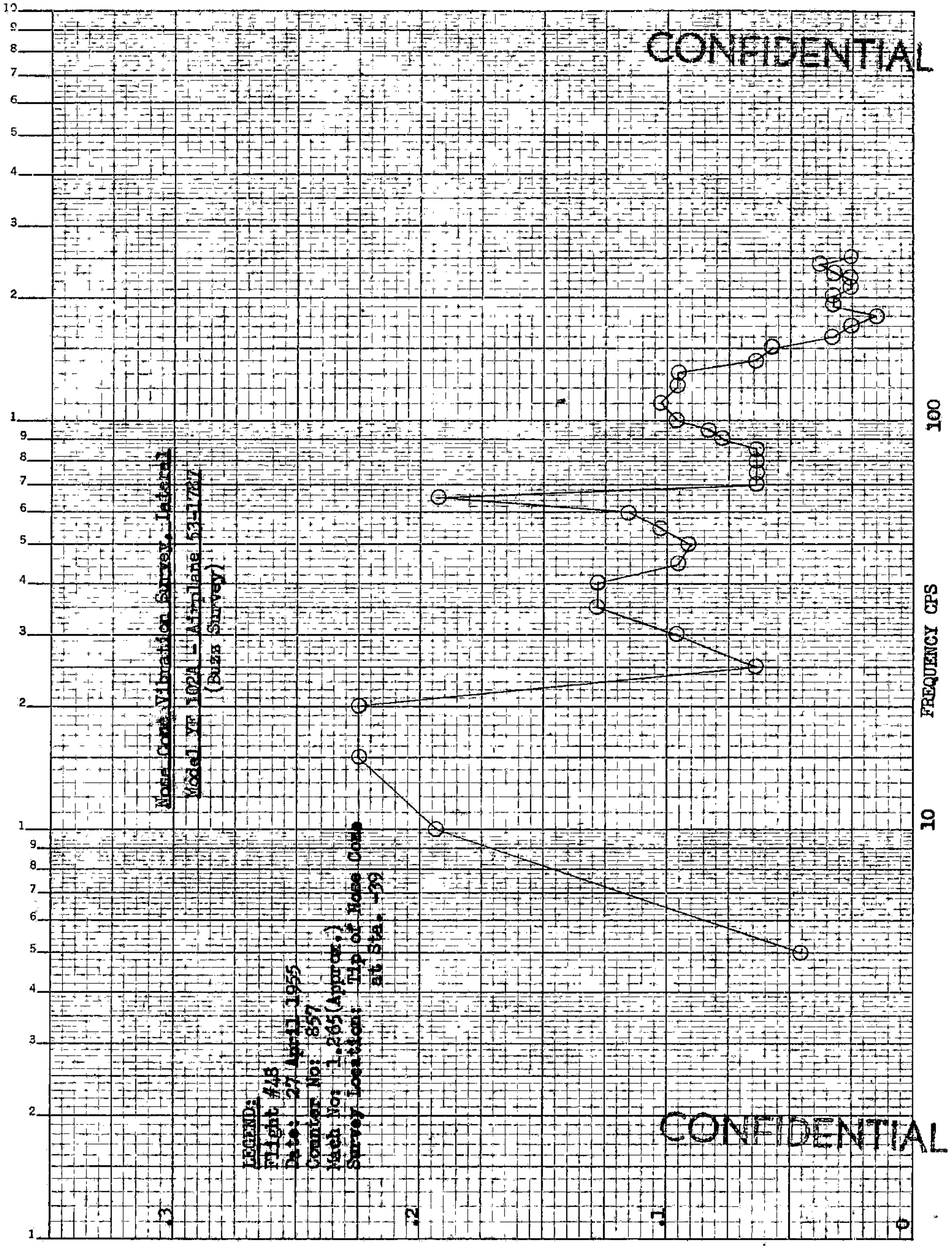


Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 851
Mach No: 1.187(Approx.)
Survey Location: Tip of
Nose Cone
at Sta. -39

Fig. 226

CONFIDENTIAL



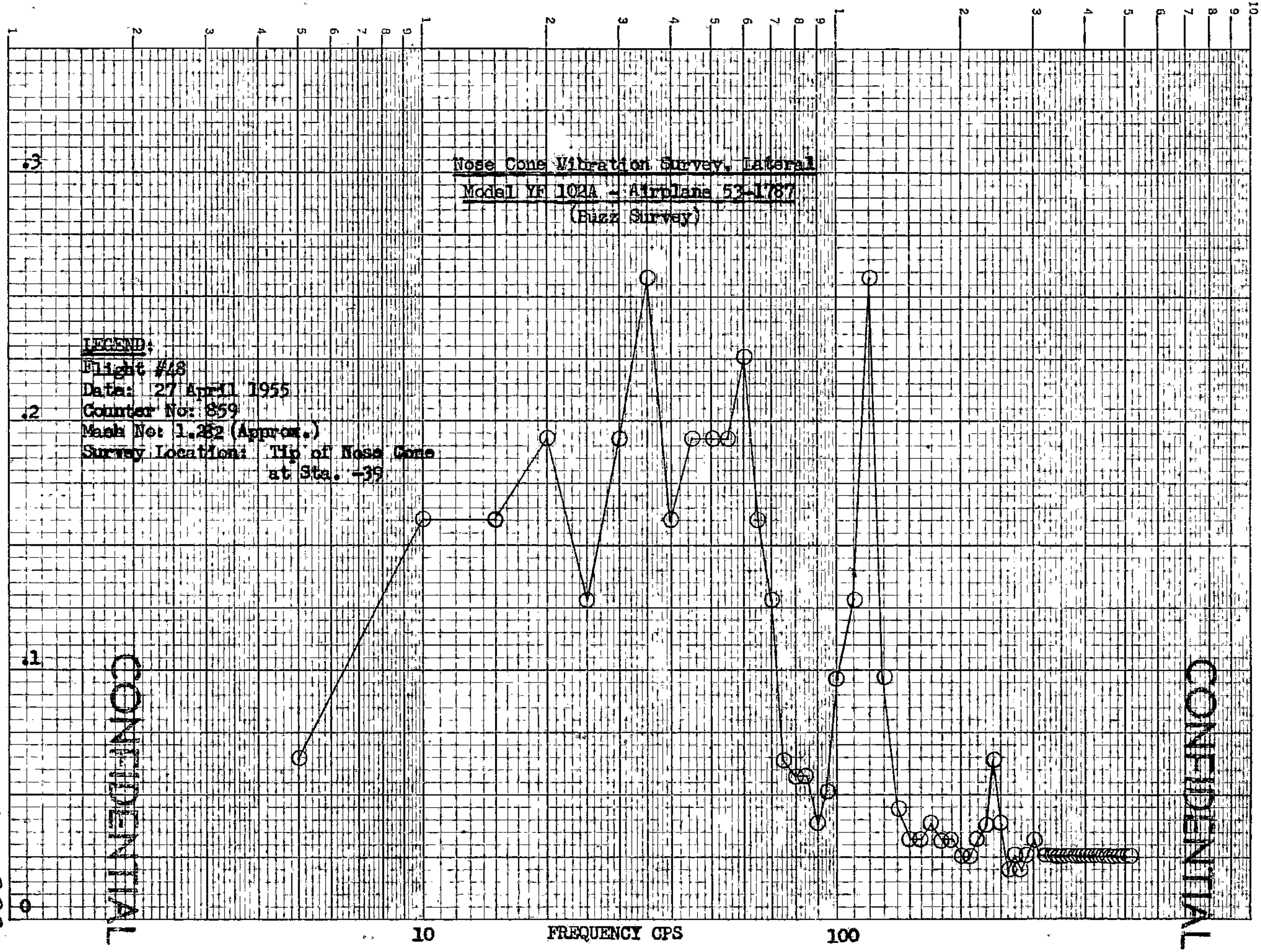
K+E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL

Fig. 227

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL

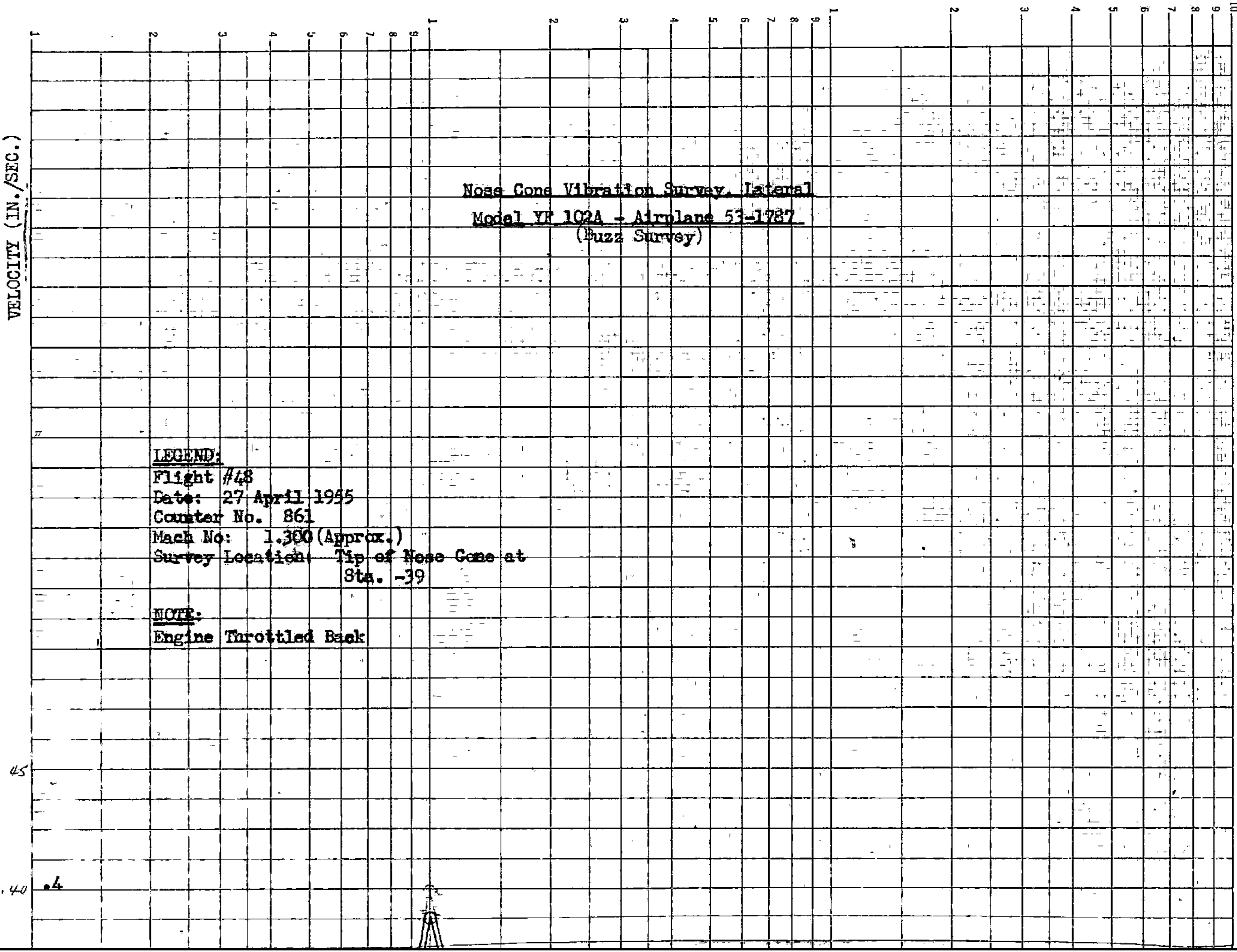

Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 859
Mach No: 1.282 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

CONFIDENTIAL

Fig. 228

359-73L KEUFFEL & ESSER CO
Semi-Logarithmic, 3 Cycles × 10 to the ½ inch.



Nose Cone Vibration Survey, Lateral

Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No. 861
Mach No: 1.300 (Approx.)
Survey Location: Tip of Nose Cone at
Sta. -39

NOTE:
Engine Throttled Back

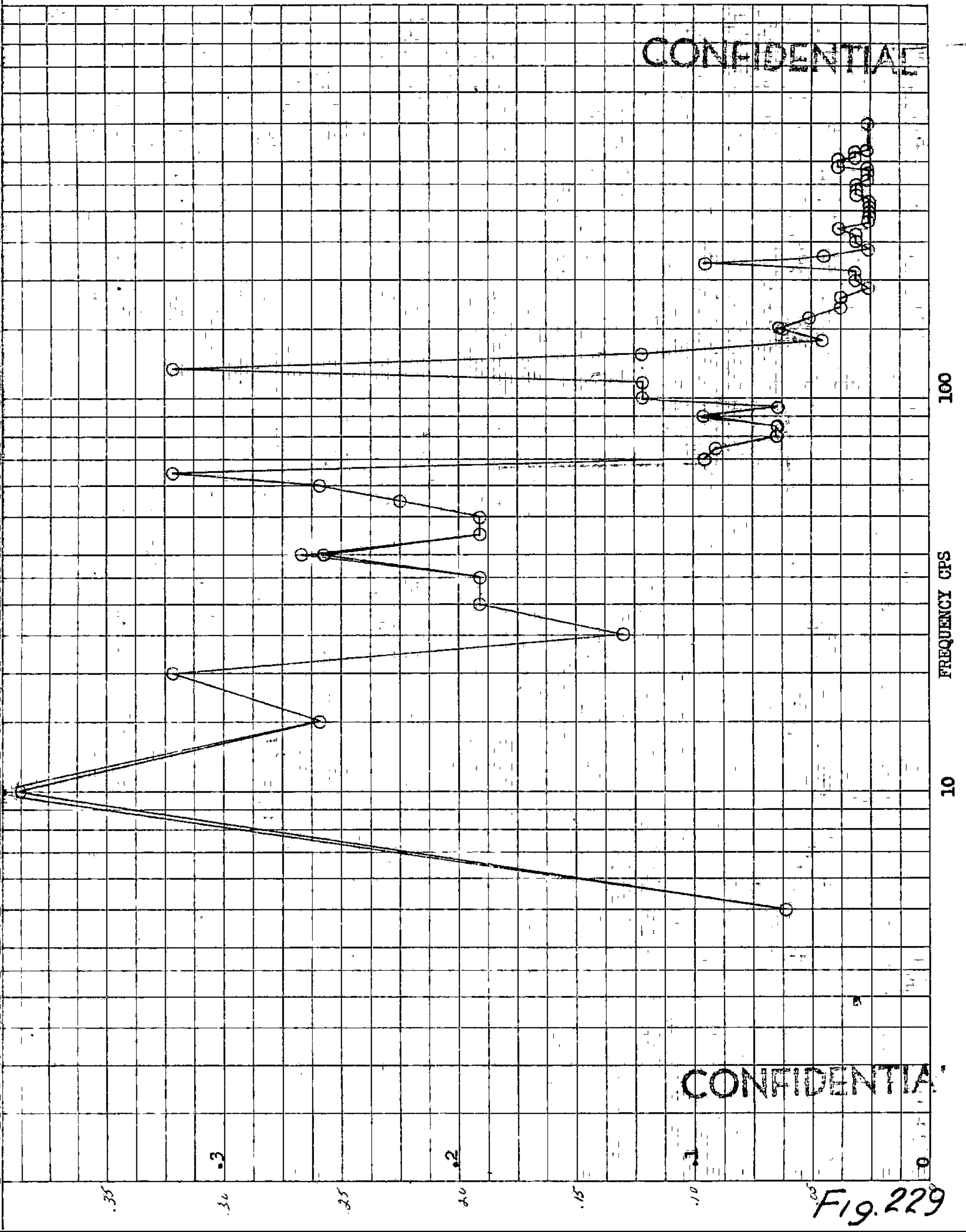
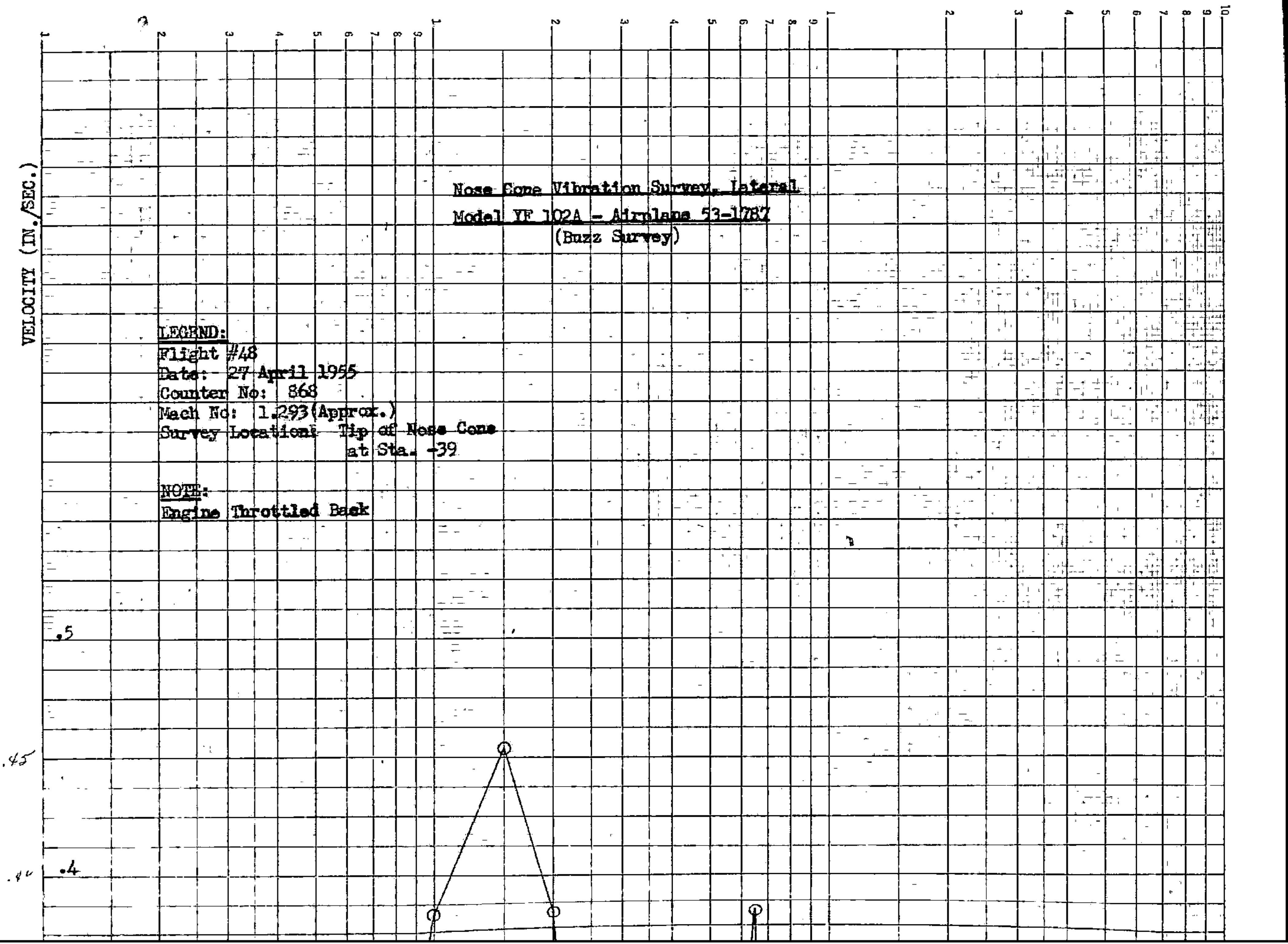


Fig. 229

359-73L KEUFFEL & ESSER CO.
Semi-Logarithmic, 3 Cycles × 10 to the ½ inch.



Nose Cone Vibration Survey, Lateral
Model YF 102A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 868
Mach No: 1.293 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

NOTE:
Engine Throttled Back

CONFIDENTIAL

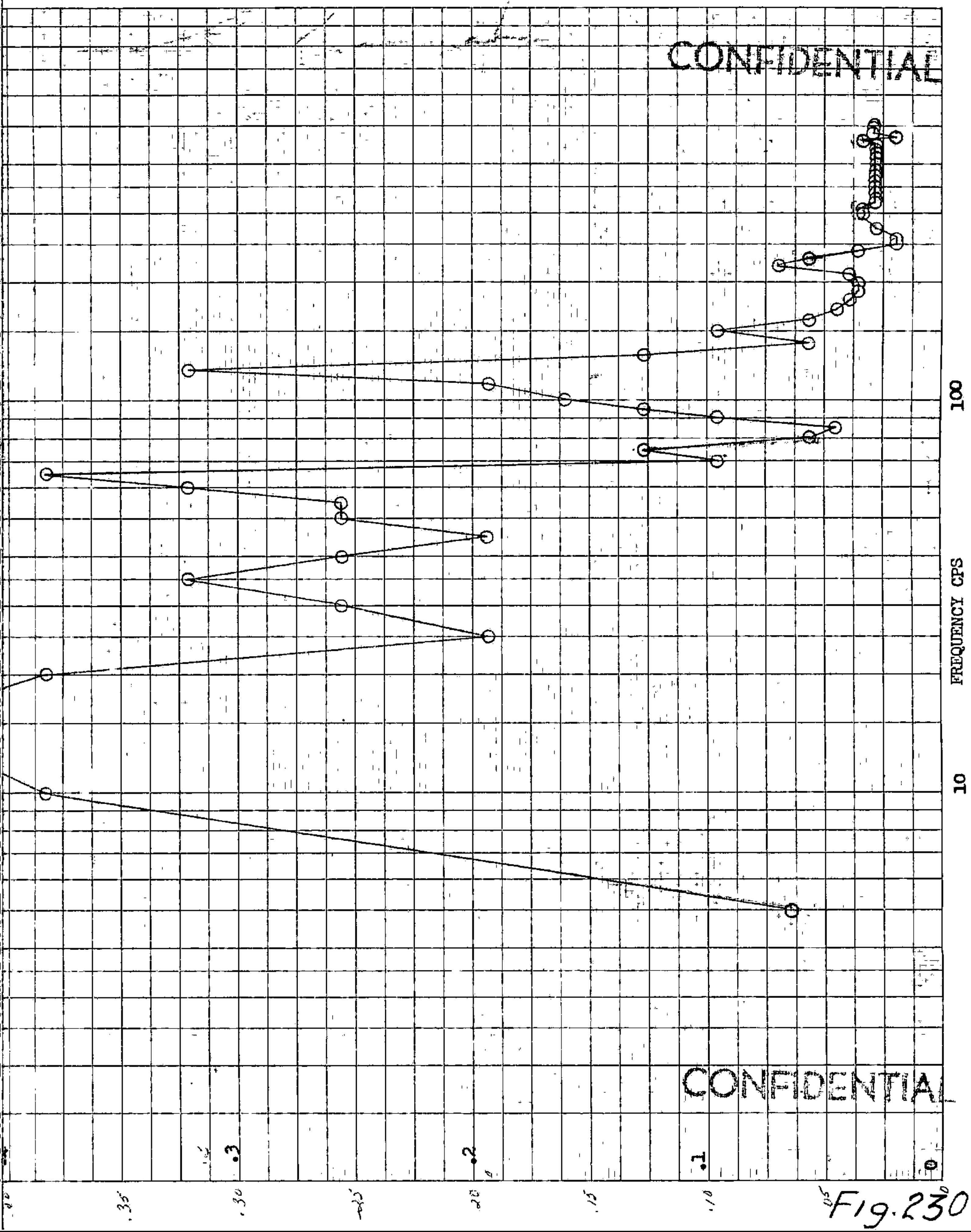


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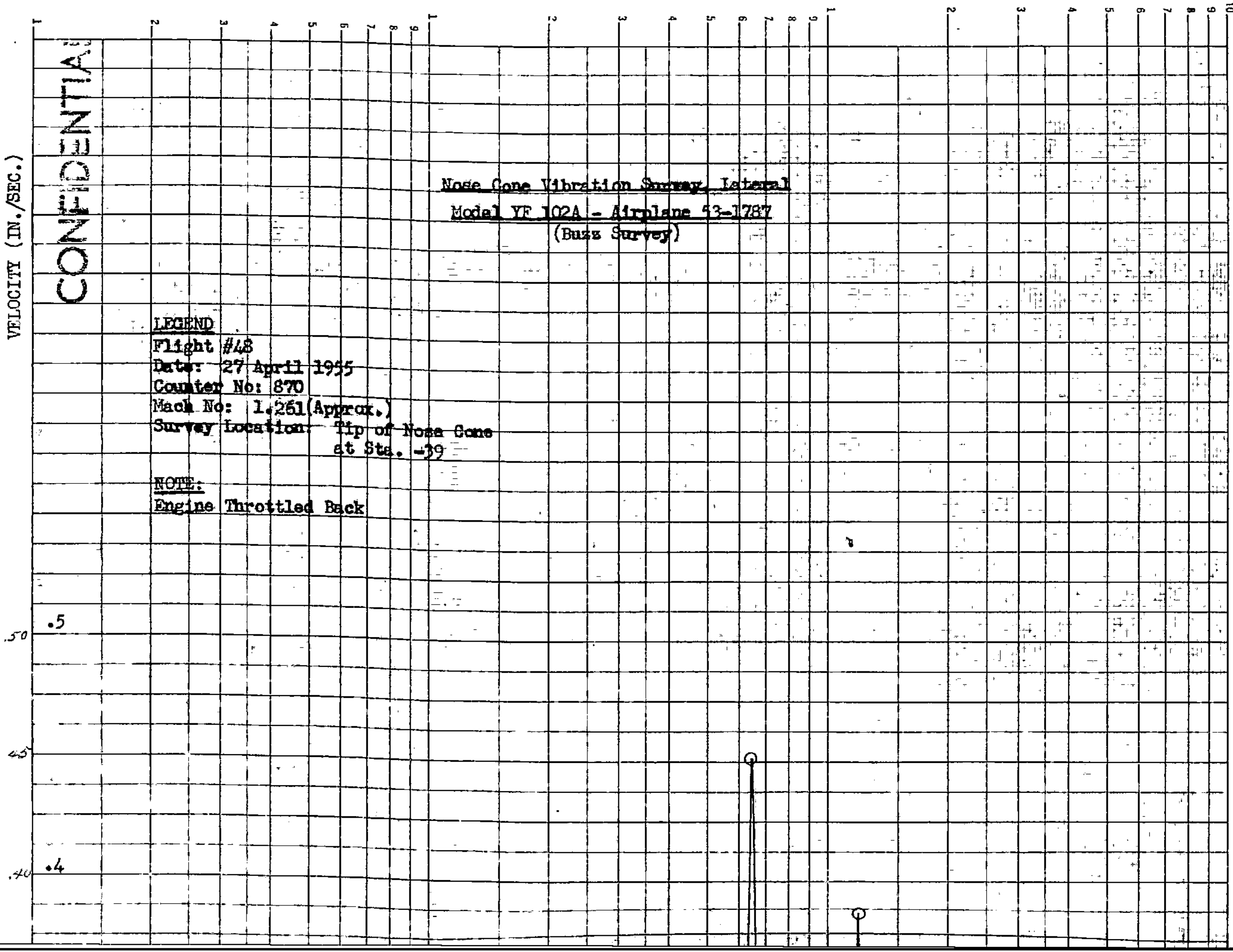
Fig. 230

359-73L KEUFFEL & ESSER CO.
Semi-Logarithmic, 3 Cycles × 10 to the ½ inch



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Nose Cone Vibration Survey, Lateral

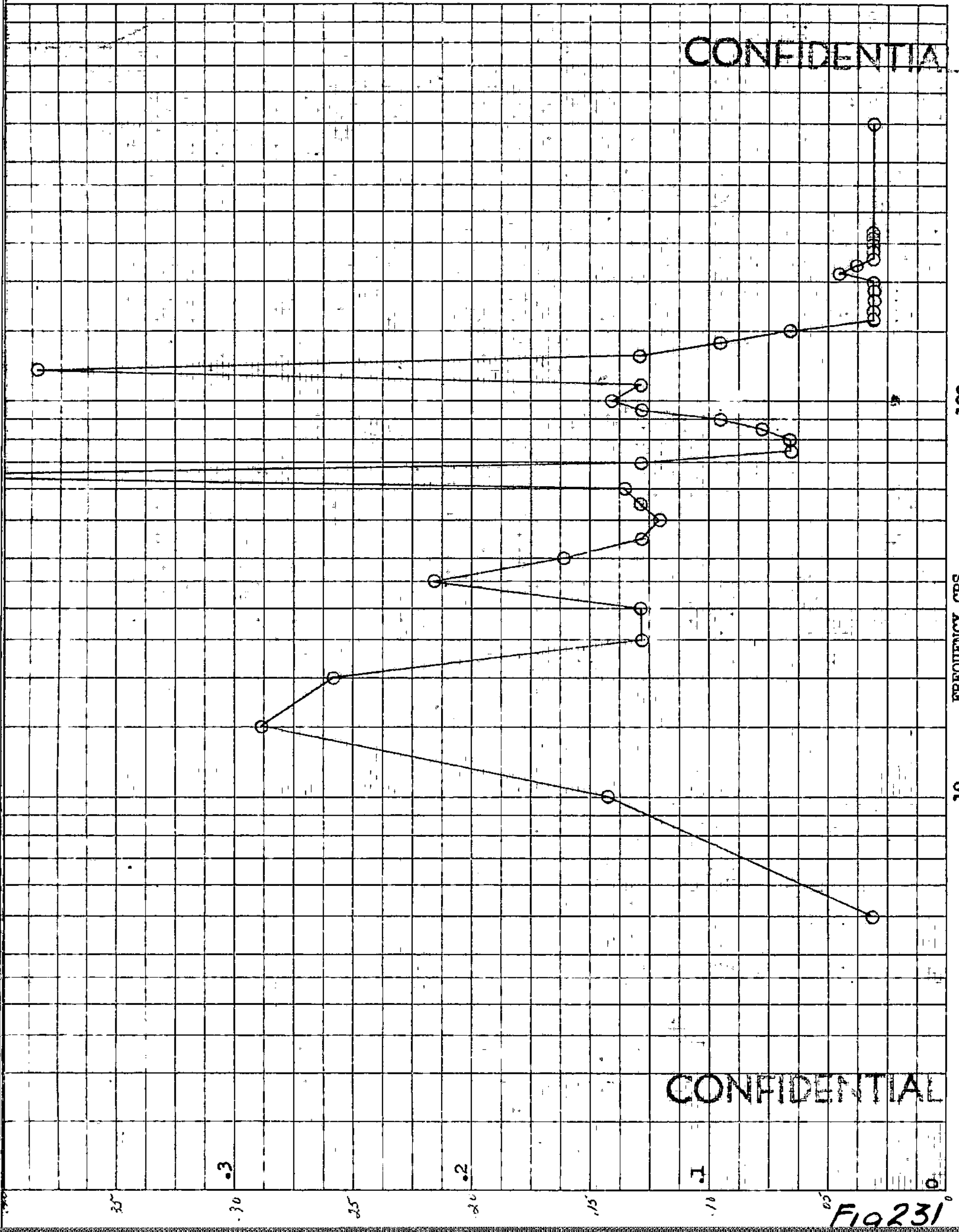
Model YF 102A - Airplane 53-1787

(Buzz Survey)

LEGEND
Flight #48
Date: 27 April 1955
Counter No: 870
Mach No: 1.261 (Approx.)
Survey Location: Tip of Nose Cone
at Sta. -39

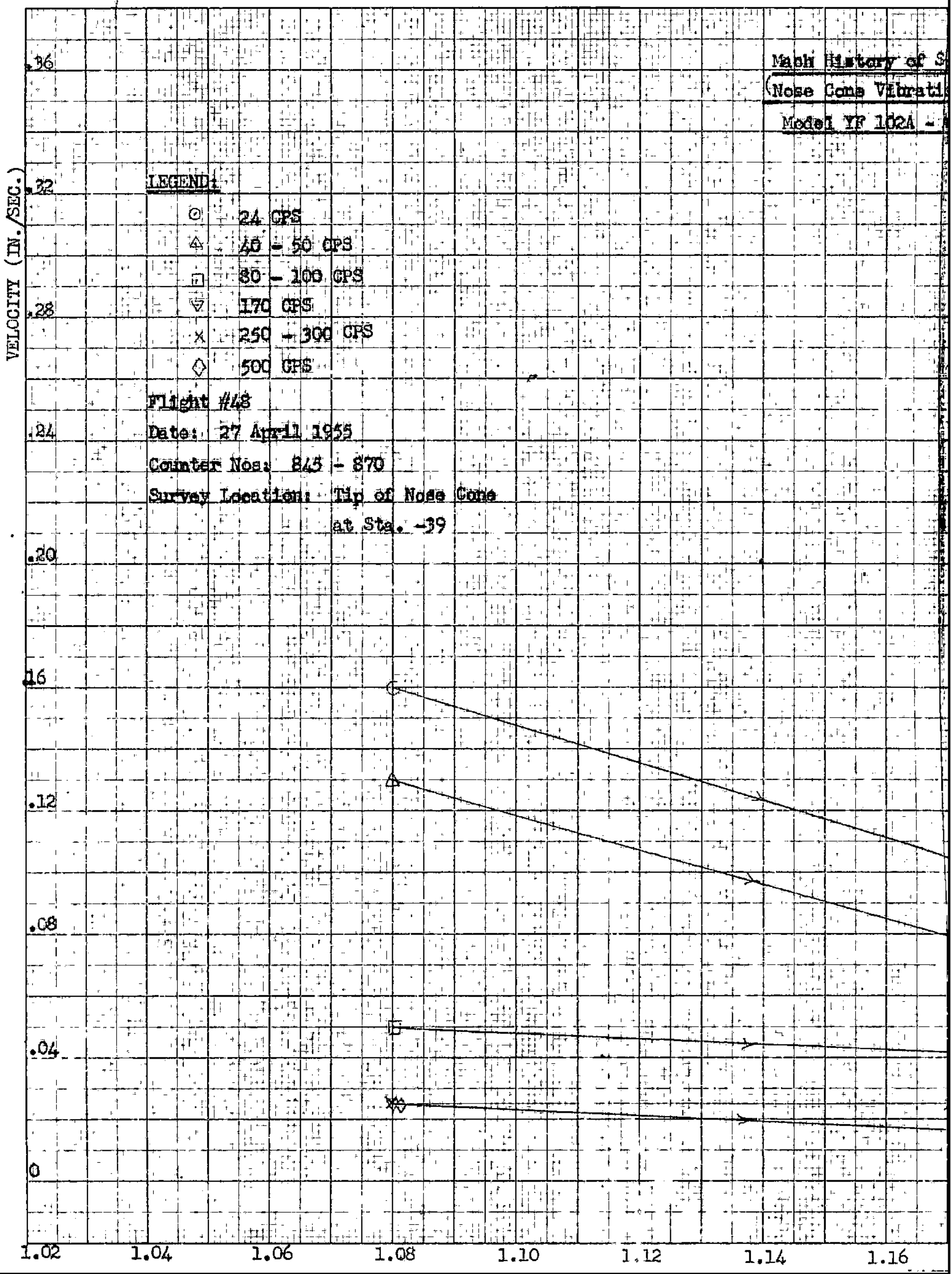
NOTE:
Engine Throttled Back

CONFIDENTIAL



CONFIDENTIAL

Fig 231



10 X 10 TO THE ½ INCH 359-11L
KEUFFEL & ESSER CO.

y of Specific Frequencies

ibration Survey, Lateral)

02A - Airplane 53-1787

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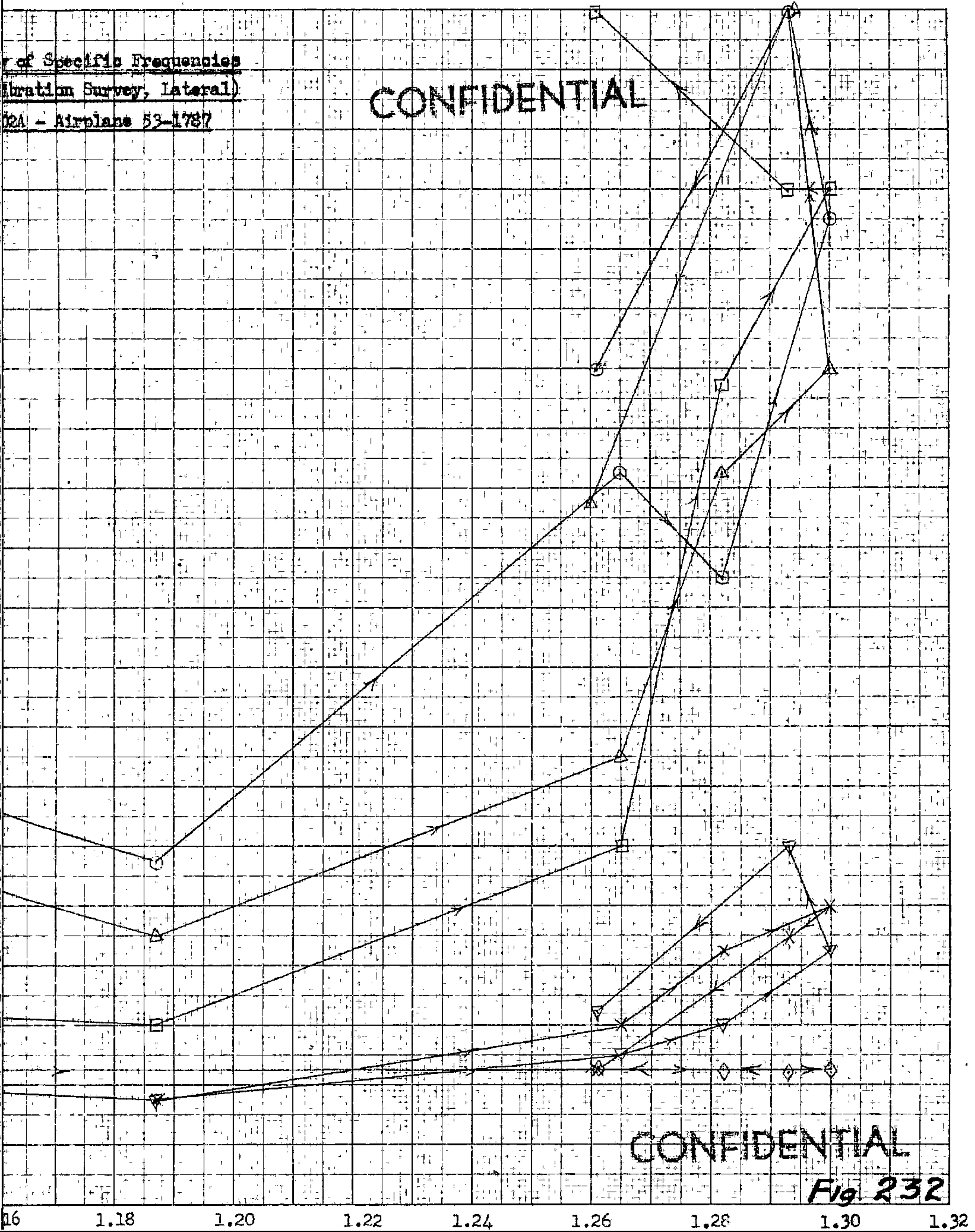

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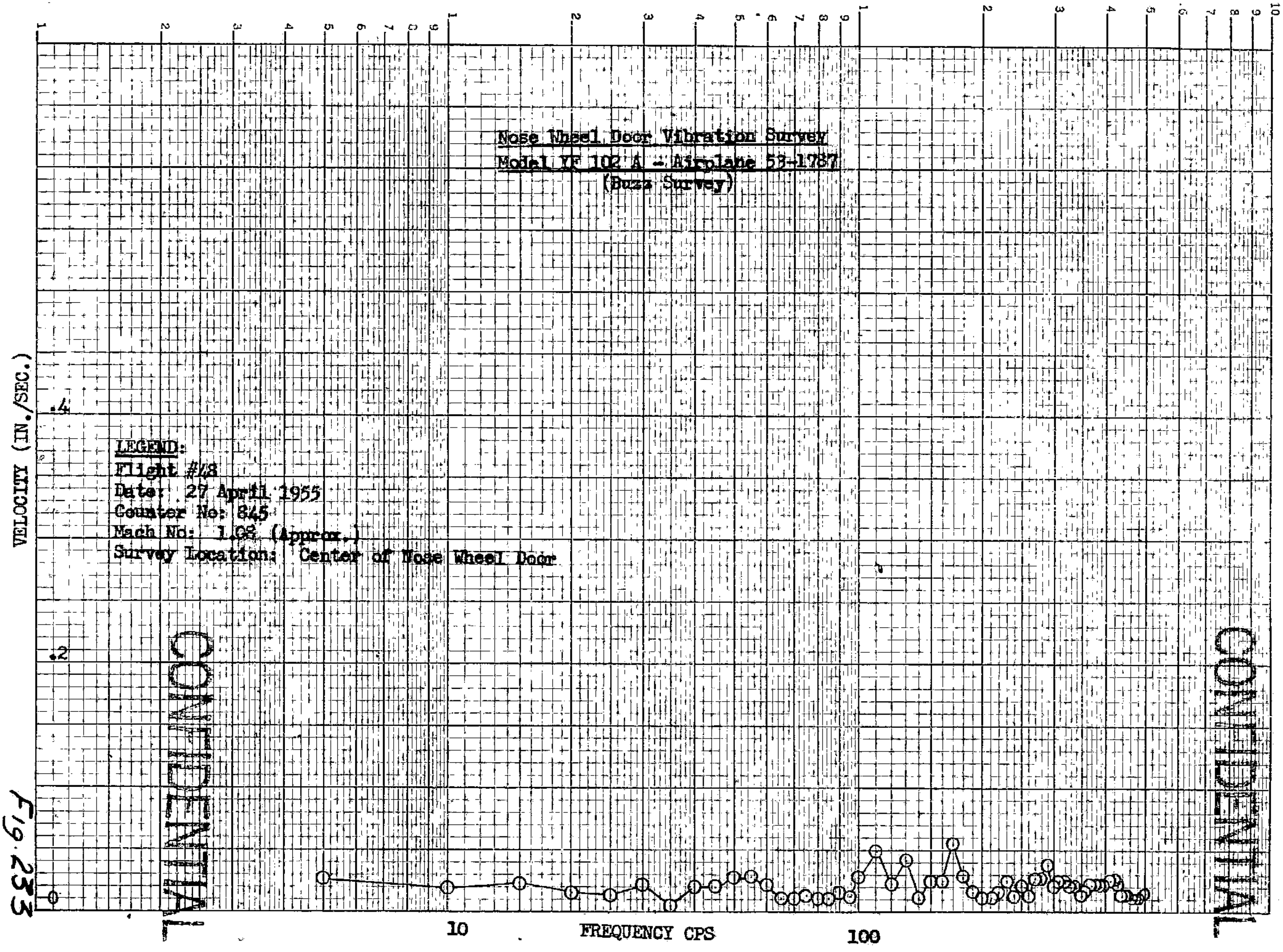
Fig. 232

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 845
Mach No: 1.08 (Approx.)
Survey Location: Center of Nose Wheel Door

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Fig. 233

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

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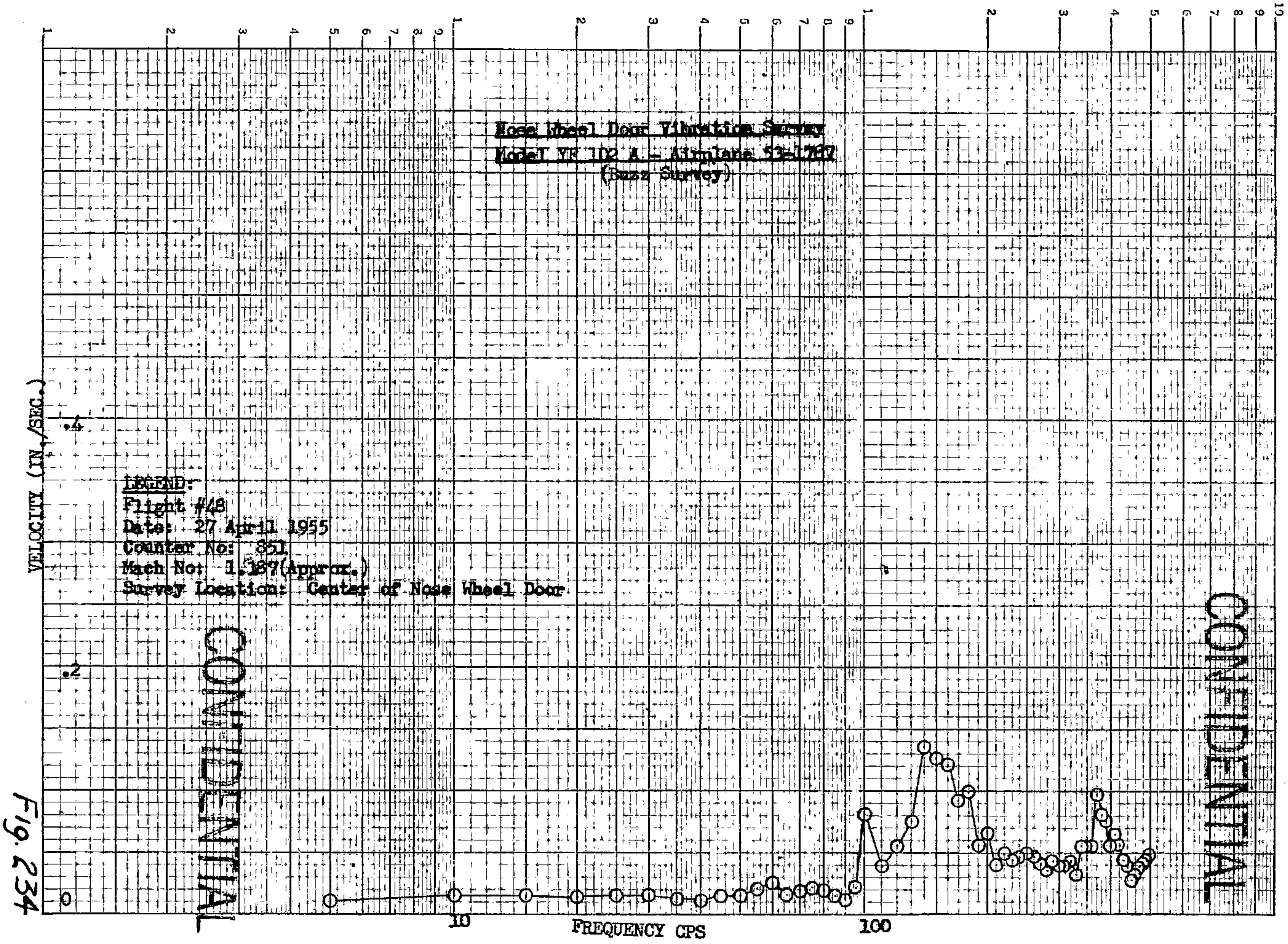

Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 851
Mach No: 1.187(Approx.)
Survey Location: Center of Nose Wheel Door

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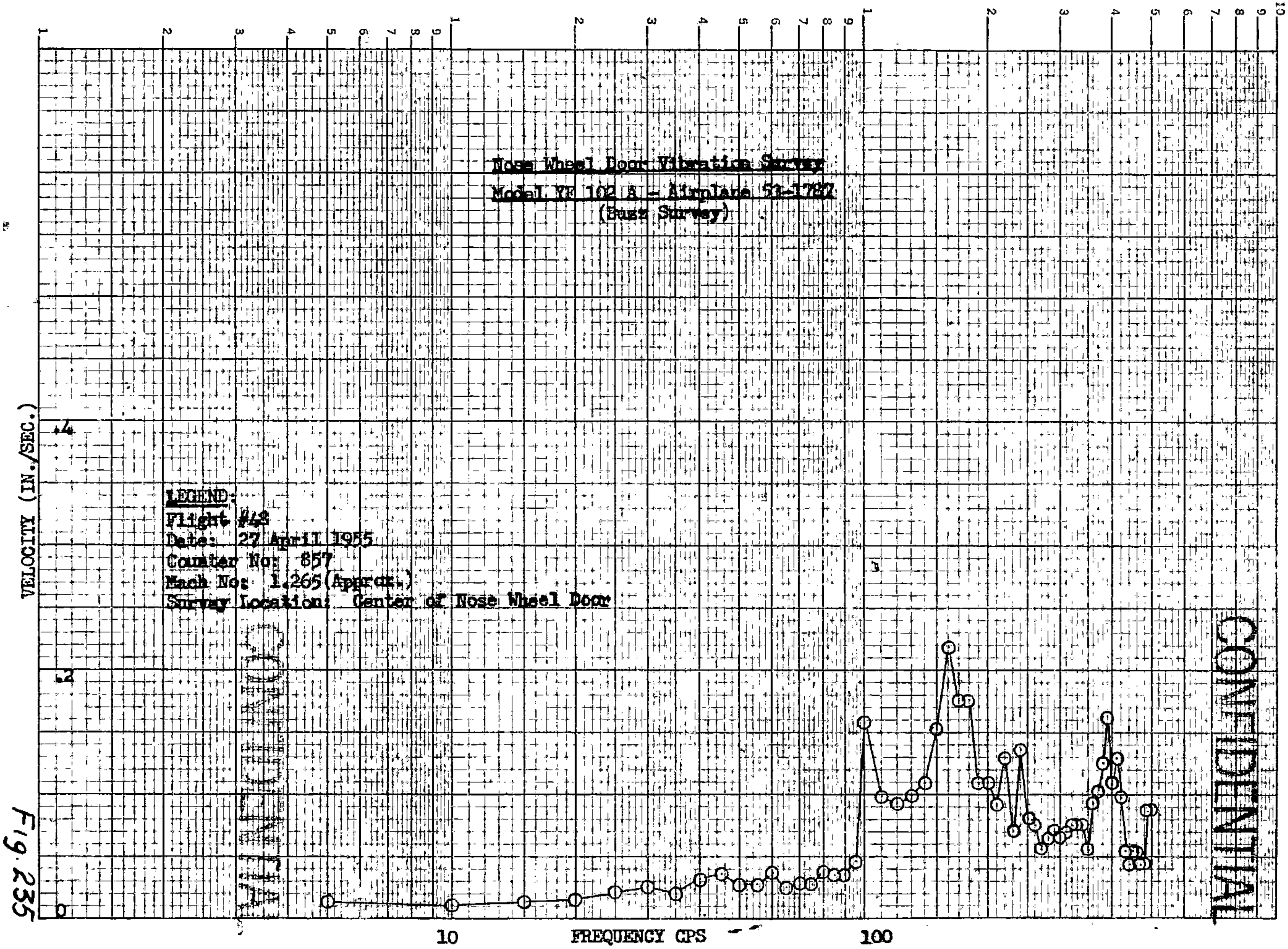
Fig. 234

K&E SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



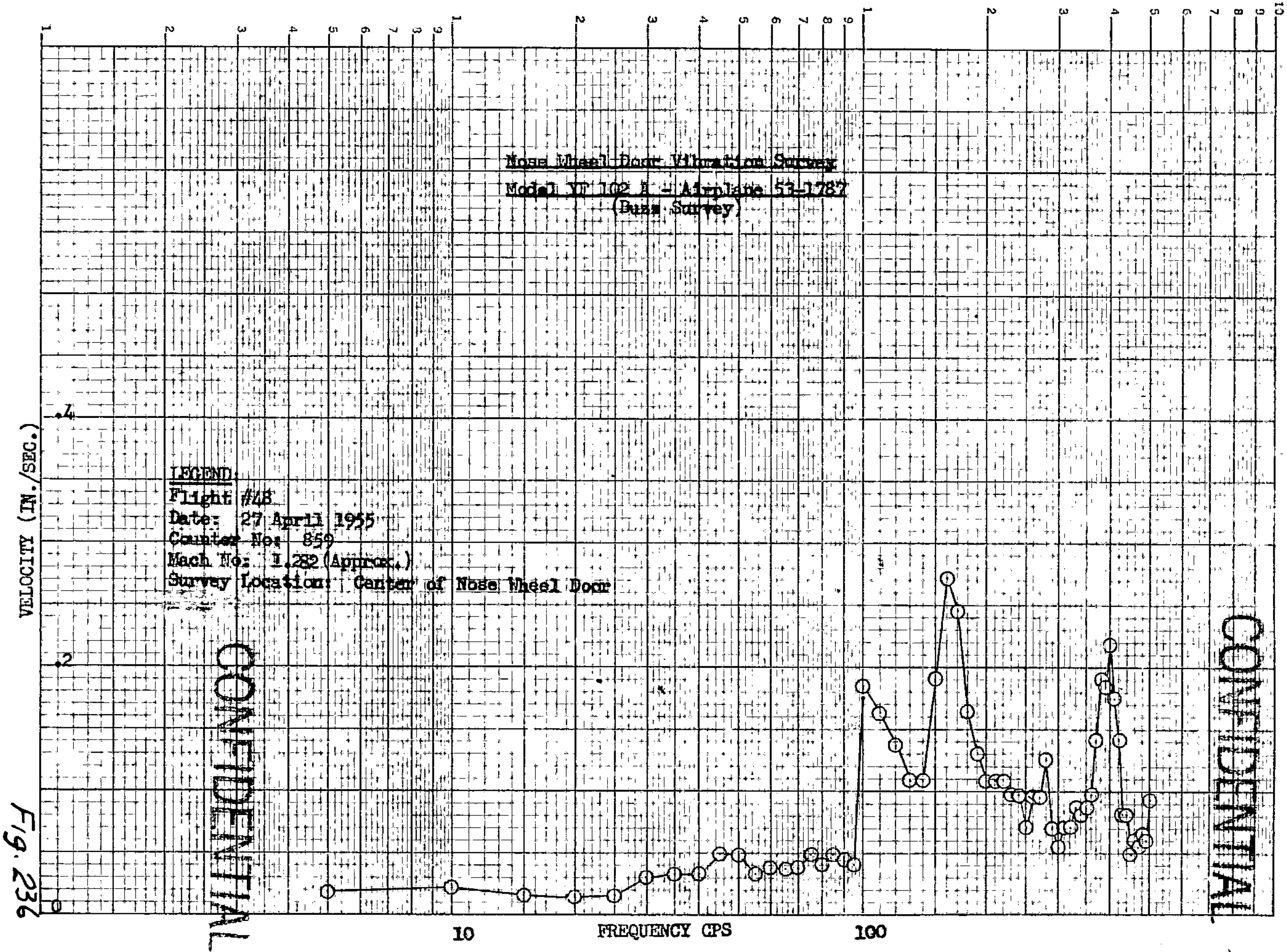
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1782
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 857
Mach No: 1.265 (Approx.)
Survey Location: Center of Nose Wheel Door

Fig. 235

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS

CONFIDENTIAL



Nose Wheel Door Vibration Survey

Model YF 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND:

Flight #48

Date: 27 April 1955

Counter No: 859

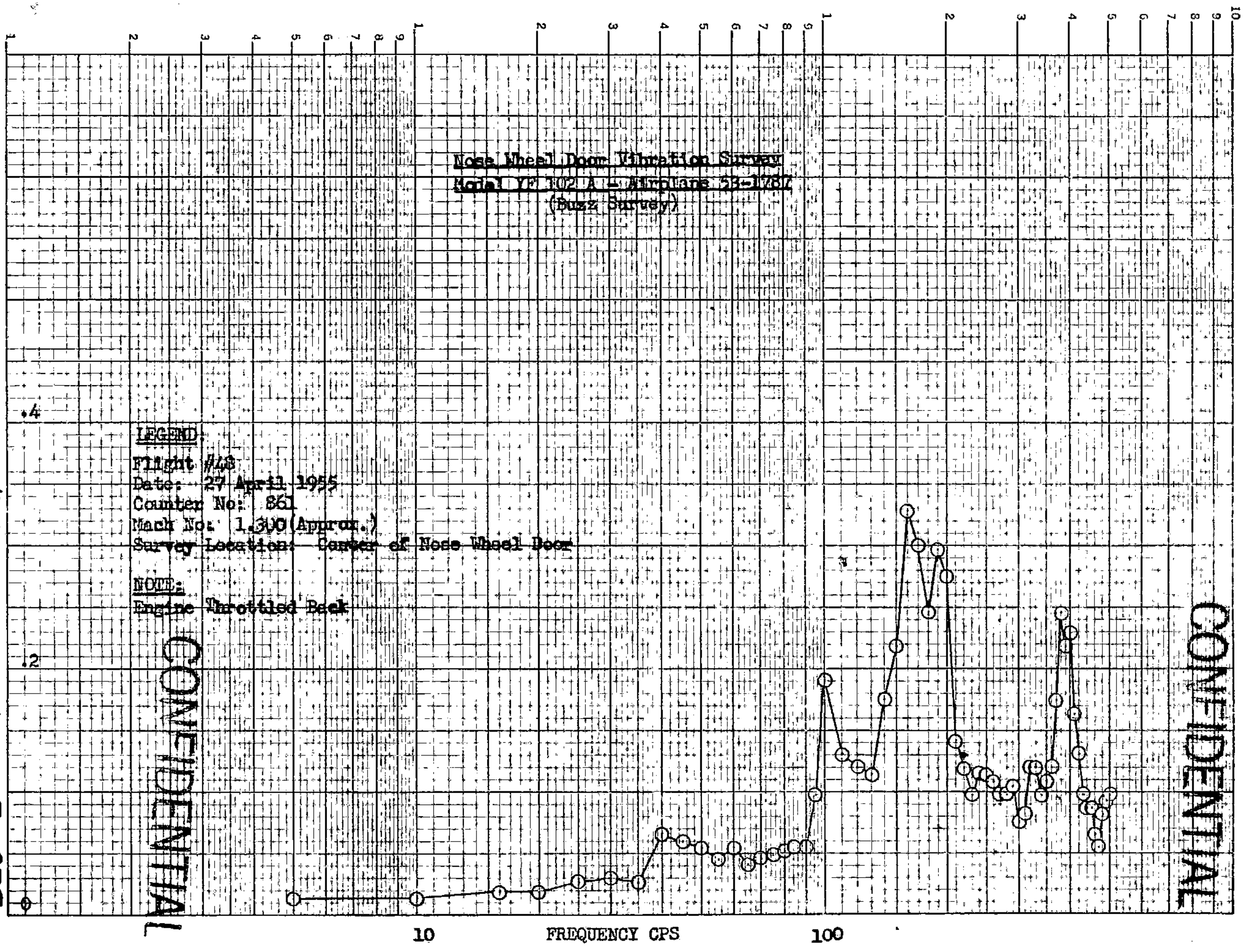
Mach No: 1.282 (Approx.)

Survey Location: Center of Nose Wheel Door

CONFIDENTIAL

Fig. 236

CONFIDENTIAL



Nose Wheel Door Vibration Survey
Model YF 102 A – Airplane 53-1787
(Buzz Survey)

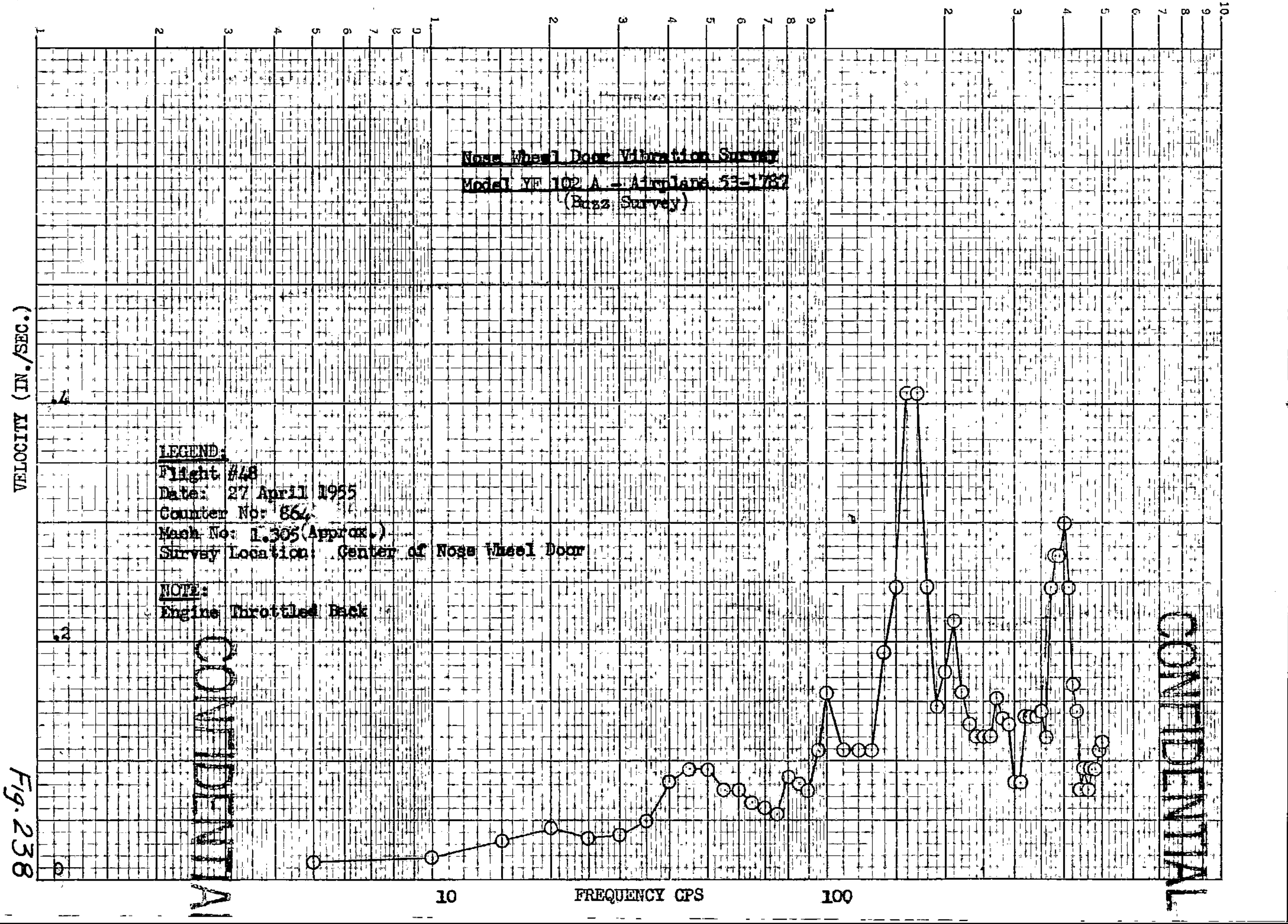
LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 861
Mach No: 1.300 (Approx.)
Survey Location: Center of Nose Wheel Door

NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 237

SEMI-LOGARITHMIC 359-71
KEUFFEL & ESSER CO. MADE IN U.S.A.
3 CYCLES X 70 DIVISIONS



Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 864
Mach No: 1.305 (Approx.)
Survey Location: Center of Nose Wheel Door

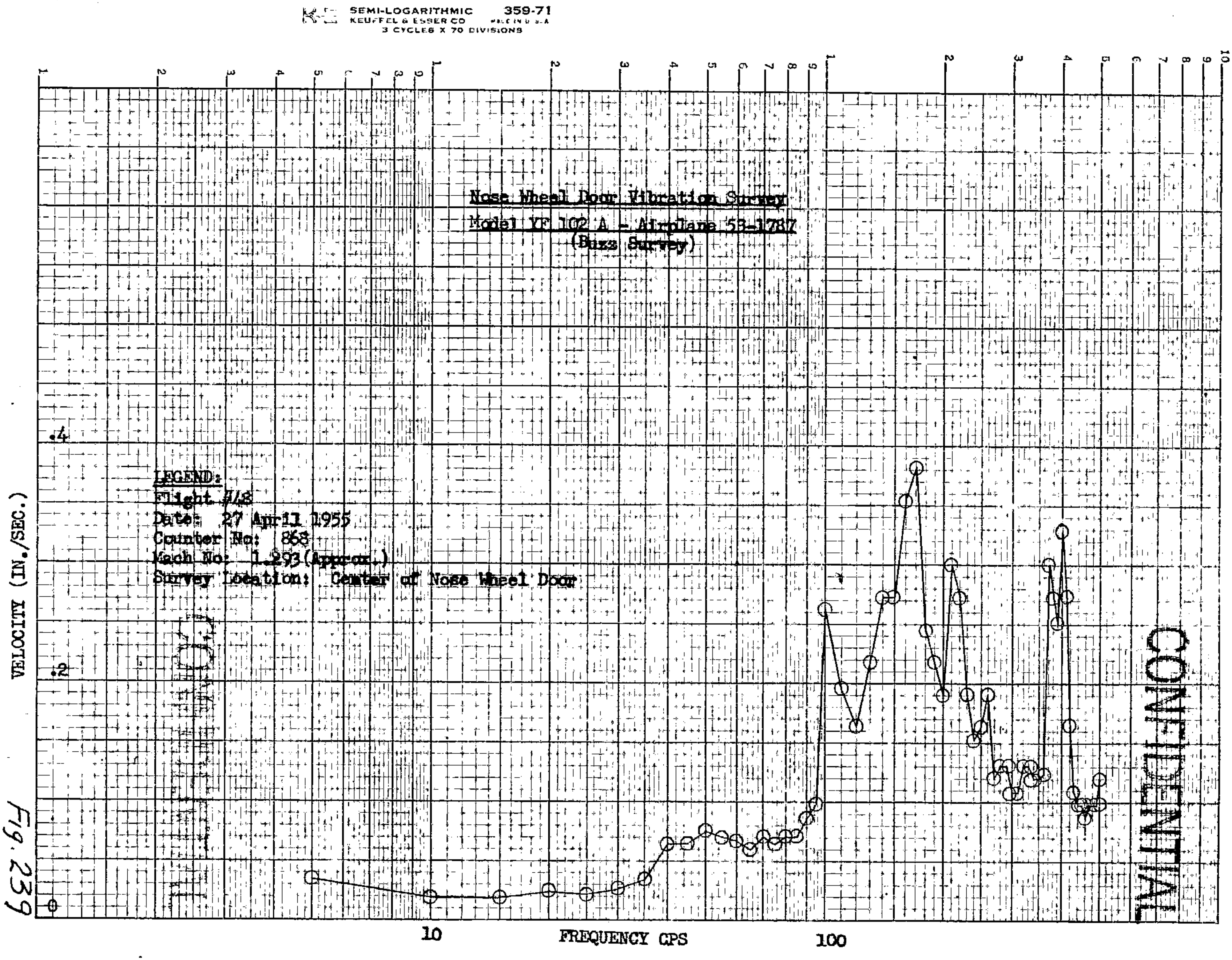
NOTE:
Engine Throttled Back

CONFIDENTIAL

CONFIDENTIAL

Fig. 238

CONFIDENTIAL



Nose Wheel Door Vibration Survey

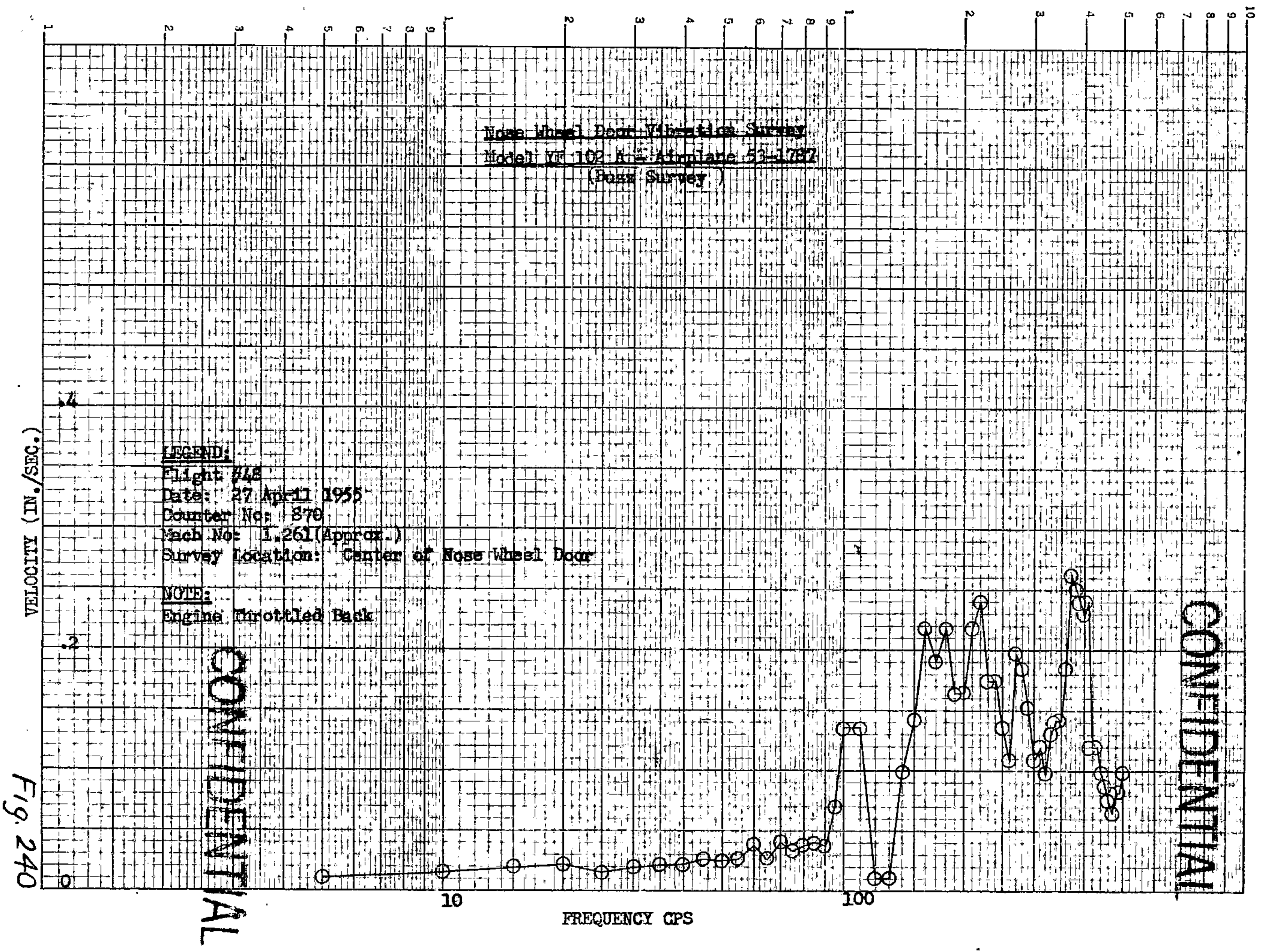
Model YF 102 A - Airplane 53-1787

(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 868
Mach No: 1.293(Approx.)
Survey Location: Center of Nose Wheel Door

Fig. 239

CONFIDENTIAL



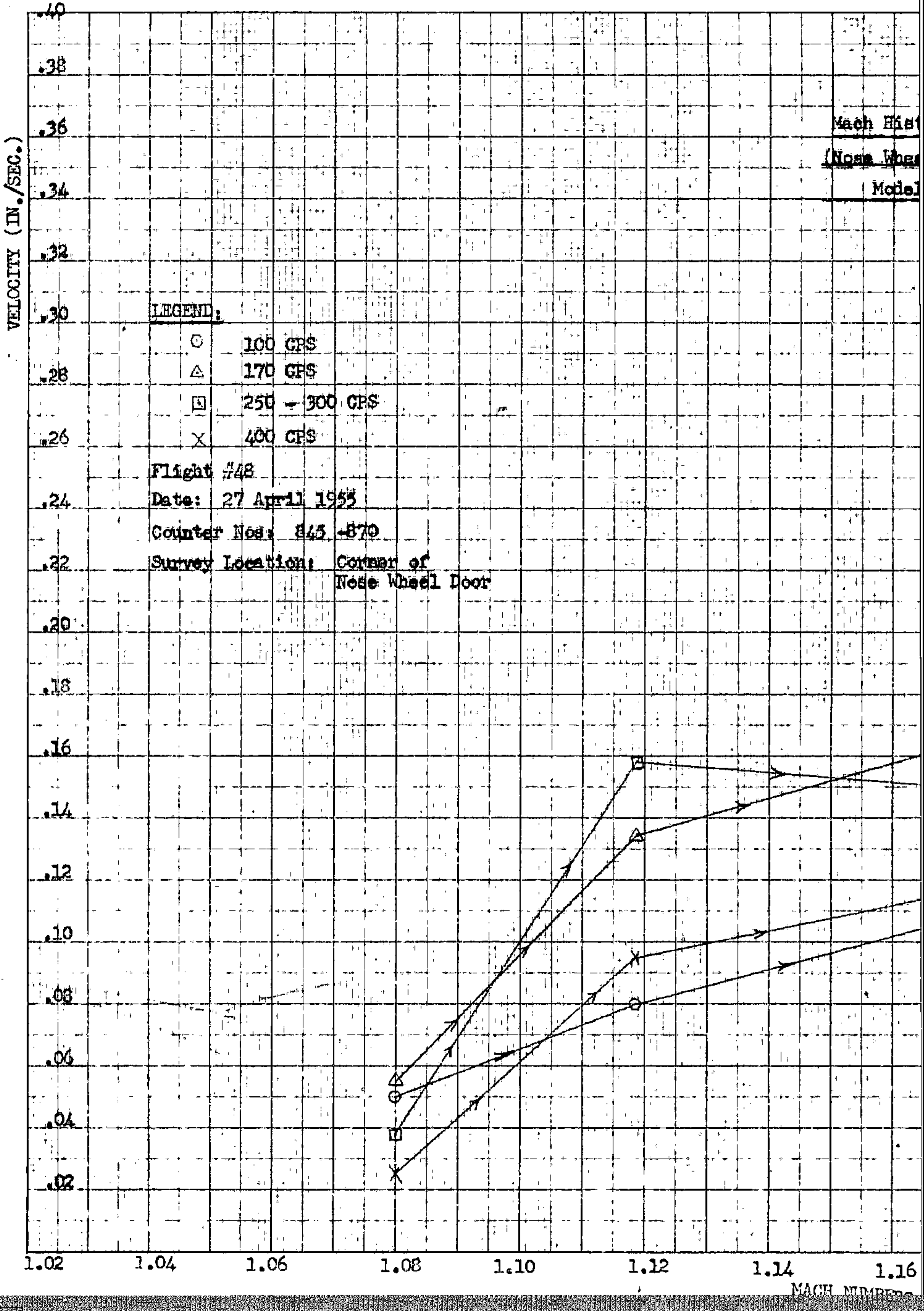
Nose Wheel Door Vibration Survey
Model YF 102 A - Airplane 53-1787
(Buzz Survey)

LEGEND:
Flight #48
Date: 27 April 1955
Counter No: 870
Mach No: 1.261(Approx.)
Survey Location: Center of Nose Wheel Door

NOTE:
Engine Throttled Back

CONFIDENTIAL

Fig. 240



10 X 10 TO THE ½ INCH 359-11L
KEUFFEL & ESSER CO.

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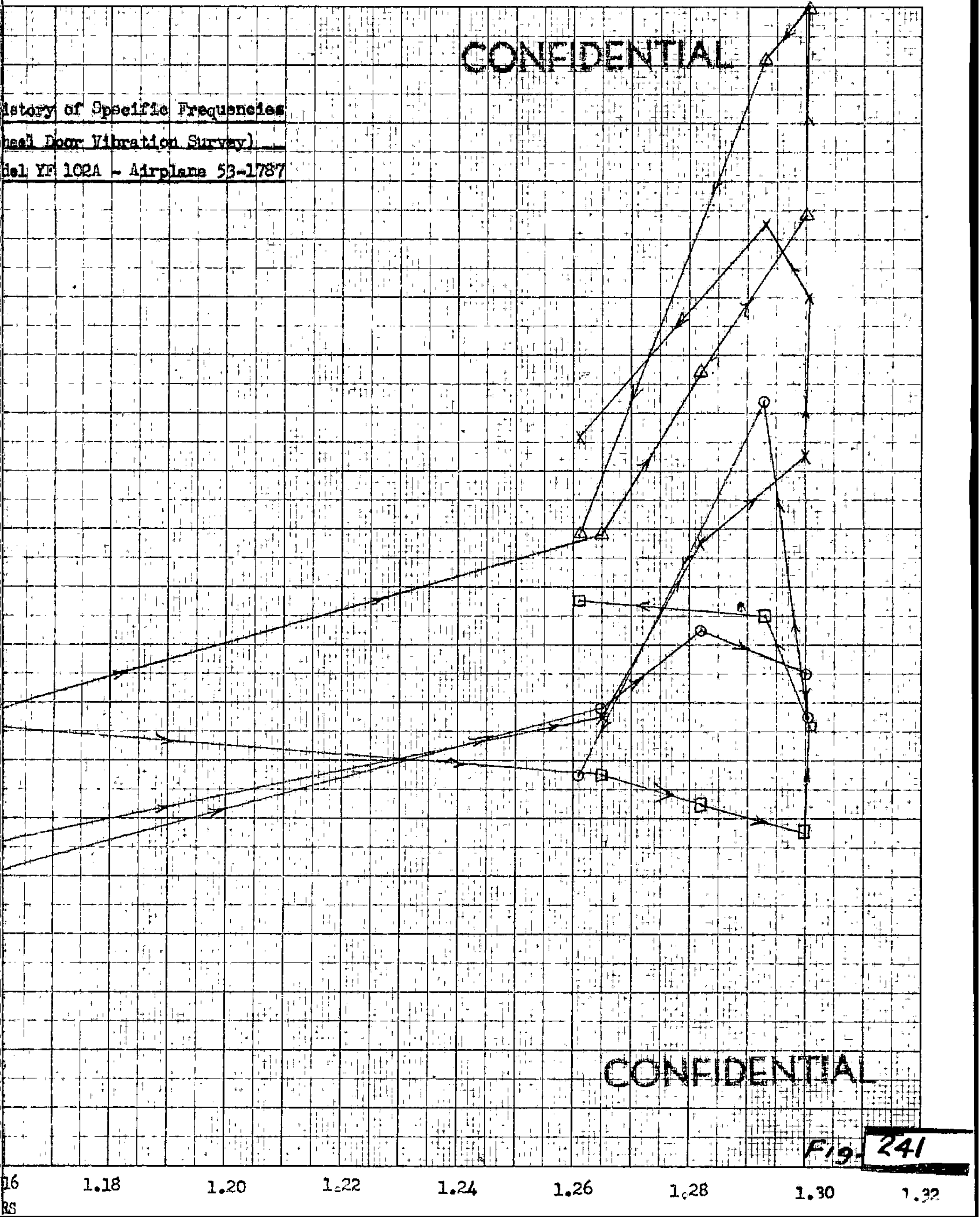
istory of Specific Frequencies

heel Door Vibration Survey)

del YF 102A - Airplane 53-1787



CONFIDENTIAL

Fig. 241

10 X 10 TO THE ½ INCH 359T-11G
KEUFFEL & ESSER CO.
ALBANENE ®

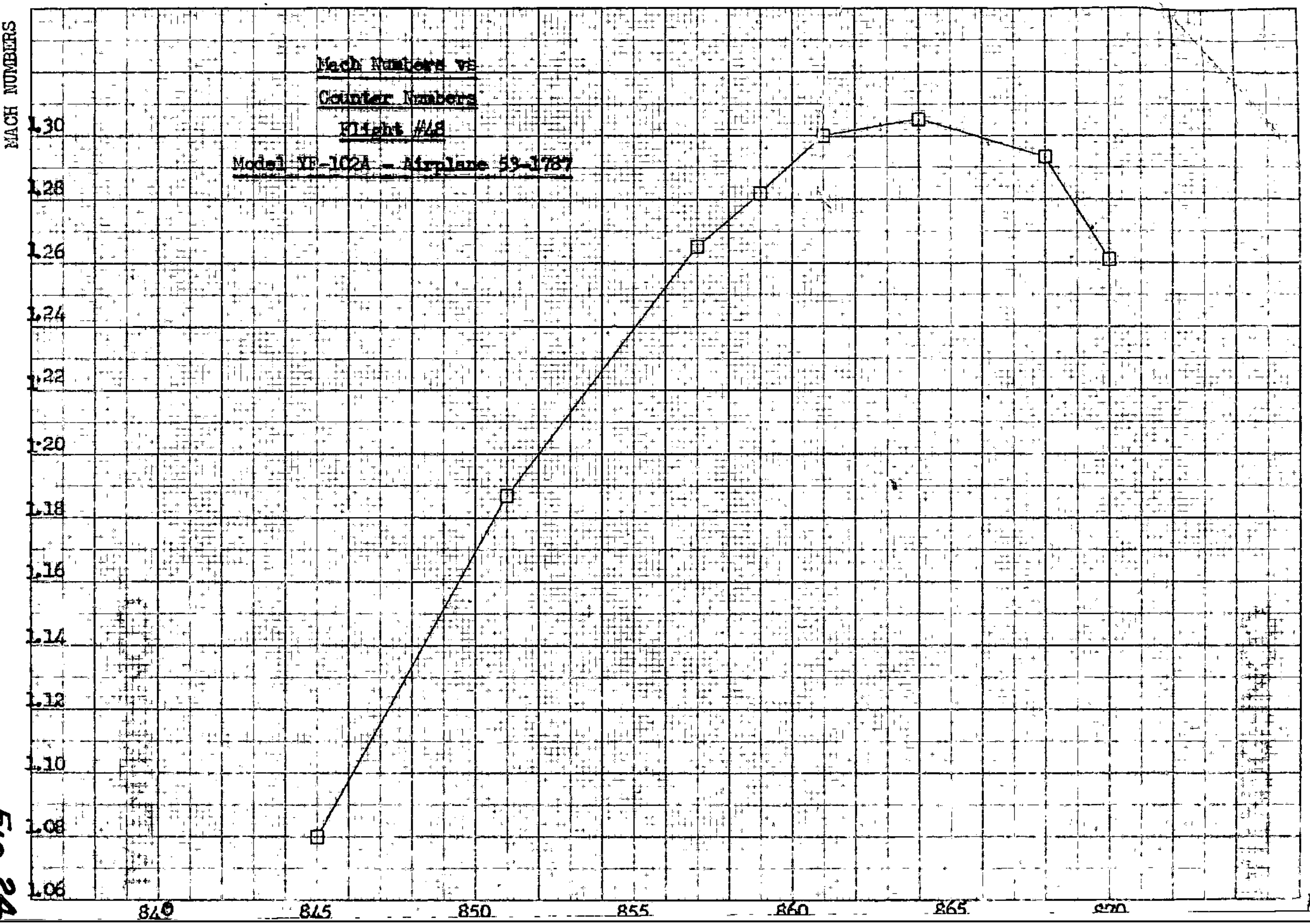


Fig. 24.

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File

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

"B" Revision, 25 Oct. 1954
Deleted fin ram air inlet test
Reason: Test assigned to
AF53-1783.

Report No.: ZC-8-103
Addendum No.: 9-B

"A" Revision:
Revised and rewritten 23 July 54
Reason: To include data for drag
evaluation

Airplane No.: 52-7995

Item Description:

Afterbody Fillet (Yellow Canary), afterburner nozzle actuator, ~~and~~ ~~[illegible]~~

Purpose:

To obtain pressure and/or temperature data for the above items.

Configuration: Noted

Procedure:

A. Yellow Canary Evaluation

1. Install pressure pickups (and one temperature pickup) in accordance with Thermodynamic Memo T-I-178 dated 12 July 1954.

2. Two pressure pickups and a single temperature pickup are located on the inner surface of the yellow canary, aft of the engine EXIT. These should be recorded during one ground engine run up (afterburner included), and flight to maximum altitude with afterburner, in addition to the flights of paragraph A3.

3. All pressure and temperature pickups should be recorded simultaneously under the following flight conditions:

   a. Level flight at 15,000 feet, covering the full Mach Number range.

   b. Level flight at 30,000 feet, covering the full Mach Number range.

   c. Flight at supersonic velocity for at least 30 seconds. It is requested that Mach No. 1.1 or better be maintained. Record during preliminary dive, and during the subsequent transition to subsonic velocity.

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Procedure: (Continued)

4. Record angle of attack and elevon surface position for the above flights. These are expected to affect the pressures.

B. Temperature of Afterburner Nozzle Actuators:

1. Install temperature pickups on upper and lower surface centerline afterburner nozzle actuators. Obtain temperature data before and after shielding, for a ground engine run up (afterburner included) and flight to maximum altitude with afterburner.

NOTE: This is for record only. Tests have been accomplished.

~~[illegible]~~

~~[illegible]~~

~~[illegible]~~

Measurements Required:

Item	Range	Reference
Yellow canary pressures	0-200PSF	Thermo Memo No. T-I-178
Yellow canary temperature	0-1100°F	Thermo Memo No. T-I-178
Temperature at Afterburner Nozzle Actuators (accomplished)	0-1100°F	Instr. Req. No. 82-15
~~[illegible]~~	~~[illegible]~~	~~[illegible]~~
~~[illegible]~~	~~[illegible]~~	~~[illegible]~~
~~[illegible]~~	~~[illegible]~~	~~[illegible]~~

Approved by: [illegible]
[illegible]

Requested by: Thermo Group

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CONVAIR
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(San Diego)

F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 22

Item Description: Vibration and Acceleration Characteristics

PURPOSE

Vibration and acceleration data is necessary to establish design factors based on actual tests, for the Hughes MG-3 system weight saving study.

AIRPLANE CONFIGURATION

Elongated nose.

PROCEDURE

1. Data shall be recorded over a full range of flight conditions, until a satisfactory pattern is established, with special emphasis on the analysis of conditions such as buffeting, gusts, landing and taxi.

2. At least one landing should be made with as much impact as pilot feels is reasonably safe.

3. Special flights shall be made only in cases where a study of the data indicates that more exploration is necessary in certain areas.

MEASUREMENTS REQUIRED

1. Acceleration, vertical and lateral, at approx. Sta. 47. Analysis to include double amplitude and frequency.

2. Vertical acceleration at C.G.

3. Basic flight control measurements.

NOTES

1. Instrument Requirements No. 82-30.

Requested by Project Office
F. B. Wozniak

Approved by T. L. Maloy
T. L. Maloy

TLM:JS:gh

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

XF-102
FLIGHT TEST ADDENDUM ITEM

28 January 1955
Report No. ZC-8-103
Addendum No. 12-B

Airplane No.: 52-7995

ITEM DESCRIPTION:

Performance of modified ram air turbine emergency hydraulic power system.

This Addendum Item supercedes Add. 12A and 13A of this report (ZC-8-103)

PURPOSE:

1. To determine power output of the ram air turbine.

2. To determine temperature rise of the hydraulic fluid during ram air turbine operation.

3. To determine ram air turbine door acceleration and snubbing cylinder pressure.

NOTE: These tests are required to provide additional data on the ram air turbine operation with modifications to the original system installed on airplane 52-7995 as shown below.

CONFIGURATION:

Tests are to be accomplished using:

1. Modified primary hydraulic system reservoir TA38 (ACA 8-7837).

2. By-pass primary and ram air turbine hydraulic fluid around the fuel-oil cooler (use 5/8" O.D. Tubing).

3. Priority valve installed in ram air turbine latching cylinder line (ACA 8-8130)-A

PROCEDURE:

1. Perform ground tests to check operation and insure proper functioning of all components. It is desirable that data be obtained during ground operation on snubbing cylinder pressure and door acceleration.

2. The flight test program shall consist of the following:

   a. Extend the RAT at approximately 8,000 feet and M = 0.4 and perform a normal landing. (Previous data obtained at 10,200 feet and M = 0.47). Primary and Secondary hydraulic systems should be operating normally.
   b. Extend the RAT at 15,000 feet and M = 0.6 and perform a normal landing (Previous data obtained at 15,000 feet and approximately M = 0.6). Primary and Secondary hydraulic systems should be operating normally.
   c. Extend the RAT at 30,000 feet and M = 0.9. (Previous data obtained at 27,500 feet and M = 0.86). With both the primary and secondary hydraulic systems shut off and the pressure "pumped down" perform a simulated approach and landing at altitude using RAT power only. Total RAT operating time should be approximately 20 minutes. Restore primary and secondary hydraulic power to make actual landing.

NOTE: 1. Maximum temperature should not exceed 250°F. at any time during the above tests.

2. It is desirable that the data specified in paragraph 2a, b and c above be obtained as closely as possible to the specified Mach No. and altitude since the data will supplement previous data at the same conditions.

3. Record the serial numbers of the priority valve installed under ACA8-8130 and the flow control valve in the RAT outlet line.

MEASUREMENTS REQUIRED:

1. RAT Hydraulic pump pressure.
2. Primary accumulator hydraulic pressure (1)
3. Secondary accumulator hydraulic pressure (1)
4. Primary hydraulic reservoir fluid temperature.
5. RAT pump outlet temperature (2)
6. RAT flow rate - at pump inlet.
7. RAT inlet fluid temperature
8. RAT pump inlet pressure - suction.
9. Rudder and elevon surface position.
10. Angle of Attack
11. Angle of Yaw
12. Airspeed
13. Altitude
14. Time

NOTE: 1. Instrumentation items 2 and 3 above require relocation of present pressure pickups from the main primary and secondary hydraulic pump side of the line shut off valves to the primary and secondary hydraulic system accumulators respectively.

2. RAT pump outlet temperature should be recorded on an oscillograph and indicated on a gage in the pilots instrument panel. This item will require additional instrumentation. See Instrument Requirement 82-35.

for E. D. Shannon
Chief of Experimental Flight

NWR

EDS:DD:bh

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File

C O N V A I R
A Division of General Dynamics Corporation
(San Diego)
YF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Sheet 1 of 2

28 October 1954
Report No. ZC-8-103
Addendum No. 25-A

Airplane No.: 52-7995

ITEM DESCRIPTION: Adverse yaw and directional characteristics.

PURPOSE:

1. Adverse yaw tests are required to determine the effect of changes in yaw damper gyro orientation upon gyro output, during rolling maneuvers. Adverse yaw characteristics have been experienced during aileron alone rolls with damper on.

2. Directional characteristics tests are to determine the effect of the rudder fairing on directional trim. Present experience indicates that approximately 1° of left rudder trim is required throughout the normal speed range up to M = .93. At M = .93 a sudden and undesirable directional trim change occurs.

AIRPLANE CONFIGURATION: Noted Below.

PROCEDURE A (Adverse Yaw):

Record yaw conditions during rolls with aileron alone at M = .9 and 35,000 feet, with and without damper, and acquire a roll rate of at least 40°/second, for each of the following gyro configurations:

a. Tilt gyro down 3 to 5 degrees, so that the yaw rate output causes rudder deflections which are less adverse during roll maneuvers.

b. Return gyro to normal, except reverse the normal direction of rotation (See Note 1).

c. Restore normal direction of rotation of gyro.

PROCEDURE B (Directional Characteristics):

1. Install a dummy actuator fairing on the opposite side of the fin from the actuator fairing, to provide symmetry.

2. Test trim characteristics through the normal speed range, including Vmax at 15,000 feet.

MEASUREMENTS REQUIRED:

1. Airspeed.

2. Pressure Altitude.

3. Outside Air Temperature.

4. Rudder Surface Position

5. R.H. Elevon Surface position.

"A" Revision, 3 Nov. 1954:
Record pressures on both R.H. and L.H. side of afterbody fillet (Instrument Requirements 82-32)
REASON: To determine whether or not the afterbody pressures are symmetrical, as this may affect directional characteristics.

Approved by:
D. H. Bennett
for E. D. Shannon
Chief of Experimental Flight

CONFIDENTIAL

28 October 1954
Report No. TC-3-10[illegible]
Addendum No. 25-A

MEASUREMENTS REQUIRED - Continued

6. L.H. Elevon surface position.

7. Nose boom yaw angle.

8. Yaw rate gyro output.

9. Roll rate gyro output.

10. Normal acceleration at C.G.

NOTES:

1. Sensing gyro of Par. A1 may be reversed by either rotating the gyro case 180° or by reversing the direction of rotation of the gyro. If gyro rotation is reversed the input voltage leads to the sensing potentiometer must also be reversed.

Requested by Dynamics

Approved by
E. D. Shannon
EN Chief of Experimental Flight

EDS:JNS:gh

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File

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

YF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 26

ITEM DESCRIPTION: Buffet characteristics and Wing Fence Evaluation.

PURPOSE: To determine buffet characteristics for various airplane configurations, and to determine the possibility of weight saving through elimination or reduction of wing fences.

AIRPLANE CONFIGURATION:

A. Install dummy rudder actuator fairing (Accomplished)

B. Install dummy fin tip antenna.

C. Remove all wing fences.

PROCEDURE:

1. Conduct flights at altitudes of 15,000 to 35,000 feet in the speed range from M = .7 to M = 1.0 to establish buffet boundary patterns. Separate and successive patterns shall be established for each of the above configurations, for comparison purposes.

2. An attempt should be made to obtain buffeting by dumping cabin pressure, and the counter number noted. Airplane configuration is optional.

3. The buffet boundary should be established for making coarse corrections and fine corrections while tracking at high altitude. It is suggested that this be obtained as a by-product of tracking flights for other purposes.

4. Stick Position/g shall be obtained after removal of the wing fences (configuration C) for normal load factors of 2g and 3g. This is in addition to the buffet pattern, which is ordinarily based on smaller load factors. The wing fences should be restored or altered if necessary after aerodynamic analysis.

5. Directional characteristics tests in accordance with Addendum 25 should be performed after removal of the wing fences.

MEASUREMENTS REQUIRED:

1. Elevon Surface Position
2. Rudder Surface Position
3. Longitudinal Stick Position
4. Vertical Velocity Pickup, Sta. 532, BL 174.
5. Lateral Velocity Pickup, Sta. 566, WL 101.3
6. Normal and Lateral Acceleration at C.G.
7. Vertical and lateral acceleration at Sta. 47.
8. Yaw rate Gyro output.
9. Nose boom yaw angle.

Approved by D. H. Bennett
for E. D. Shannon
Chief of Experimental Flight

EDS:JNS:gn
Requested by:
Dynamics & Aero

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file

F-102 FLIGHT TEST INSTRUMENT REQUIREMENT AND FLIGHT TEST REQUEST

Model F-102 — Requirement No. 82 19

Ser. No. AF 52-7995 — 24 June 1954

TITLE DESCRIPTION: Effect of landing gear selector valve operation on secondary system hydraulic pressure.

TEST PURPOSE: To investigate a possible cause of secondary system hydraulic pressure fluctuations as indicated by existing flight test data.

MEASUREMENTS REQ.

CANCELLED

Reason: Fluctuations traced to hydraulic pumps.
5 November 1954

1. Nose gear door selector valve position.

2. Nose gear selector valve position.

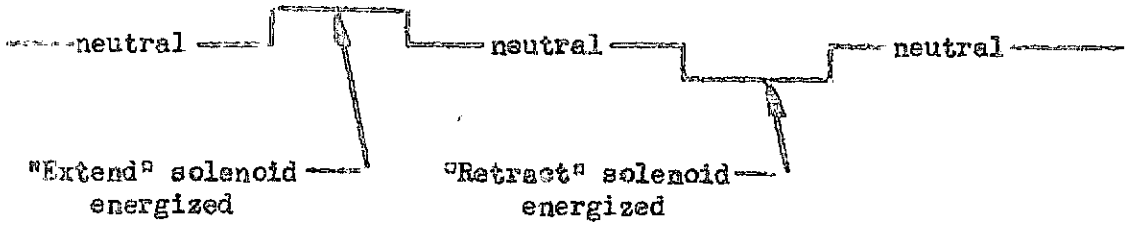
3. Main gear selector valve position.

4. Secondary system hydraulic pressure (0-3500PSI).

5. Primary system hydraulic pressure (0-3500PSI).

NOTES

1. Each selector valve trace to indicate thus:



. Record all measurements simultaneously on the same oscillograph.

Frequency response 40 cps minimum.

rding to be continuous during takeoff and landing. Obtain om several flights (special flights not required).

e instrumentation and tests for earliest practicable date.

request confirms advance letter to V. Allwardt, dated 21 June 54, H. Wright and T. Chadwick. No separate addendum item shall be red.

Checked by N W Reed
N. W. Reed

Approved by W J Schack
W. J. Schack

Approved by H Cheney
H. Cheney

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San Diego

C O N F I D E N T I A L

File

F-102

ZC-8-105
Sheet 1 of 2

FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Addendum No. 7-A

"A" Revision: 26 Oct. 1954
Added Sheet 2 (Pump Temperature Tests).

Item Description:

Primary hydraulic system oil temperatures.

Purpose:

To determine the necessity of an oil cooler for the primary hydraulic system by measuring the designated oil temperatures with (1) the oil cooler operative, and (2) the oil cooler by-passed.

Configuration:

Optional.

Procedure:

Make measurements at convenient intervals (not to exceed 10 seconds) during a typical performance flight.

Note: Temperature pick-ups, if insulated from ambient air, may be installed on the outside of the hydraulic oil lines.

Measurements Required:

Hydraulic Oil Temperatures *		**References**	
		ZC-8-163	ZC-8-052
1. Reservoir	C - 160°F	Page 64 #20	Page II - 8 #148
*2. Pump Outlet	C - 180°F		
*3. Oil into Hydraulic Oil Cooler	C - 300°F	Page 64 #16	
*4. Oil out of Elevon Servo Valve	C - 280°F	Page 64 #10	
*5 [illegible] Speed Brake [illegible] Line	C - 280°F	Page 64 #13	
Oil Out of [illegible]	C - 280°F	Page 64 # 3	

[illegible] Instr. [illegible]

Requested by Hydraulics Group

Approved by T. L. Maloy
T. L. Maloy
Group Engineer
Experimental Flight

TLM:lp

C O N F I D E N T I A L

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San Diego

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ZC-8-103
Sheet 2 of 2

Addendum No. 7-A

1ST REVISION:

PURPOSE:

The purpose of this revision is to require additional testing to determine the cause of the high temperature conditions which were experienced at the hydraulic pumps after the deletion of the hydraulic fluid coolers. Data analysis shall include determination of the following:

A. Amount of heat added to the pump by engine radiation.

B. Amount of heat carried across the flanges by conduction.

C. Amount of heat added by internal leakage within the pump.

PROCEDURE:

Measurements shall be recorded during several typical flights, and maintained for at least two minutes after shutdown.

MEASUREMENTS:

In addition to Items 1, 2 and 4 of Sheet 1, the following temperatures shall be recorded, for the Primary System:

A. Lower side of pump flange.

B. Lower side of mating engine flange.

C. Top aft end of pump case.

D. Pump internal by-pass (Case drain line).

NOTE: Instrument Requirements No. 82-31.

Requested by General Design

Approved by D. H. Bennett
for E. D. Shannon
Chief of Experimental Flight

EDS:JHS:gh

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file

Date: 1 Oct., 1954

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 23

ITEM DESCRIPTION: Cable Tension Values

PURPOSE:

To obtain control cable tension values from tests for design purposes.

PROCEDURE:

Elevator, Aileron and rudder cable tension variations versus time should be recorded at the following conditions.

1. A flight at high altitude and minimum power setting to obtain maximum cold conditions.

2. A flight at 20,000 feet and maximum Mach number with afterburner to obtain a high heat condition. (Level Flight)

MEASUREMENTS REQUIRED:

1. Rudder cable tension.

2. Elevator cable tension.

3. Ailerson cable tension.

NOTES:

1. Instrument Requirement No. 82-29 (ref.).

2. Install during the next engine change. Instruments should be removed for use on AF53-1780 after the flight plan of this addendum has been accomplished.

Requested by Controls Group

Approved by D. H. Bennett
for E. D. Shannon
Chief of Experimental Flight

EDS:JFS:gh

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Model No. F102
Report No. ZC-8-103
Addendum 15-A
Date - 19 July 1954

"A" Revision:
Added Two (2) Flts. with blanket and extended shroud.

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CONVAIR
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(San Diego)

FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No: YF102 S/N52-7995

ITEM DESCRIPTION:

Temperature of engine outer wall and nozzle actuators.

PURPOSE:

To determine the temperature of the engine outer wall and the nozzle actuators in order to ascertain the amount of insulation necessary to relieve the heat damage to the nozzle actuators.

PROCEDURE:

Record temperatures continuously from take-off to service ceiling with airplane at maximum power plus afterburner.

Reference AVO dated 7-13-54 signed by F.A. Stephenson. See following table for airplane use.

Airplane S/N	No. Flts. Req'd.	Configuration		
		Shields	Blanket	Blanket & Extended Shroud
52-7995	2	X		
52-7995	2		X	
52-7995	2			X

Test requested by Thermo.

Approved by T. L. Maloy
T. L. Maloy

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E.D. [illegible]annon
J.H.B.

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 22

Item Description: Vibration and Acceleration Characteristics

PURPOSE

Vibration and acceleration data is necessary to establish design factors based on actual tests, for the Hughes MG-3 system weight saving study.

AIRPLANE CONFIGURATION

Elongated nose.

PROCEDURE

1. Data shall be recorded over a full range of flight conditions, until a satisfactory pattern is established, with special emphasis on the analysis of conditions such as buffeting, gusts, landing and taxi.

2. At least one landing should be made with as much impact as pilot feels is reasonably safe.

3. Special flights shall be made only in cases where a study of the data indicates that more exploration is necessary in certain areas.

MEASUREMENTS REQUIRED

1. Acceleration, vertical and lateral, at approx. Sta. 47. Analysis to include double amplitude and frequency.

2. Vertical acceleration at C.G.

3. Basic flight control measurements.

NOTES

1. Instrument Requirements No. 82-30.

Requested by Project Office
F. B. Wozniak

Approved by T L Maloy
T. L. Maloy

TLM:JS:gh

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"A" Revision

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

YF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Date:	21 July 1954	Report No.:	ZC-8-10[illegible]
Airplane No.:	52-7995	Addendum No.:	16-A

ITEM DESCRIPTION: J57-P-11 Compressor Stall Test

PURPOSE:

To isolate the conditions resulting in compressor stall characteristics as have been experienced in recent flights at EAFB.

SCHEDULE:

Requested target completion date: 13 August 1954.

PROCEDURE:

Perform one or more flights as required to obtain the reported engine pulsation and stall conditions. Operations shall include the following:

(1) Steady state (power lever fixed) flights with climb to altitude where most pulsations occur. Obtain continuous data recordings from 20,000 feet and up. More than one pulsation should be recorded at each time, I.E., do not retard power lever immediately after first occurrance, but continue to permit engine to pulsate for at least three times.

(2) Perform what the pilot considers a normal engine acceleration (transient condition) at the following altitudes:

a. Maximum attainable altitude
b. 40,000 feet
c. 35,000 feet
d. 30,000 feet

MEASUREMENTS REQUIRED:

(a) Altitude
(b) Airspeed
(c) O.A.T.
(d) T_{T2} (Total inlet temperature)
(e) T_{T7} (Turbine discharge temperature)
(f) N_1 (RPM)
(g) N_2 (RPM)

"A" Revision

1. Perform several turn and bank manuevers during the flights of this Addendum.
REASON: USAF request in order to complete the inlet duct pressure survey initiated by Addendum 2.

2. Added measurements as shown on Sheet 2.

3. Perform tests with both old and new type fuel control units:
Old unit part No. 190075-13
New unit part No. 190255-11

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YF-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

MEASUREMENTS REQUIRED: (Continued)

*(h) P_{T2} (Total inlet pressure). Use a minimum of 4 (four) rakes each containing 4 (four) pressure pickups. A minimum of one rake to have dynamic pickups with oscillograph recording in addition to photo panel recordings.
NOTE: Engine P_{T2} instrumentation probes may be used on an interim basis, if necessary, pending completion of rakes.
(i) One total pressure pickup in each intake duct upstream of splitter.
*(j) P_{T4} (Compressor discharge total pressure).
*(k) P_{S4} (A/B meter signal pressure).
*(l) $P_{S4.5}$ (Burner pressure).
*(m) P_{T7} (One position minimum on oscillograph).
(n) P_{T0} (Airspeed total pressure).
(o) Main engine fuel flow.
(p) A/B fuel flow
*(q) A/B fuel manifold pressure.
*(r) Primary fuel pressure at pressurizing valve.
*(s) Secondary fuel pressure at pressurizing valve.
*(t) Eng. fuel pump inlet pressure.
(u) Compressor bleed valve positions.
(v) A/B nozzle position.
(w) PLA (power lever angle).
*(x) Holley bleed valve governor, case temp.
*(y) Fuel temp. between fuel pump exit and fuel control.

* Denotes items which should be recorded by oscillograph for more accurate study of sequencing of events and dynamic characteristics.

REFERENCE:

Instrument Requirement No. 82-23, 82-24, 82-25, 82-28.

Requested by: Engineering Power Plant Group Memo dated 15 July 19[illegible]

Approved by T L Maloy
T. L. Maloy
Group Engineer
Experimental Fl[illegible]

TLM:JS:gh

Add the Following to Measurement Requirements:
*1. P_{T3} (P&W Kit)
*2. Holley governor pressure bias
3. Holley governor temp. bias
4. Fuel pump inlet fuel temp.
5. Static press. in vicinity of fuel control
6. Fuel inlet press. (piezometer)

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F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Date: 4 August 1954

Report No. ZT-8-103
Addendum No. 17

Airplane No.: S/N 52-7995

Item Description:

Engine Lubricating Oil Temperatures with and without Air-Oil Cooler.

Purpose:

To determine whether or not the engine lubricating oil can be cooled adequately with only the fuel-oil cooler.

Configuration:

Completely by-pass air-oil cooler.

Procedure:

Measure the following oil and fuel temperatures throughout a typical high altitude flight.

Measurements Required:

1. Temperature of engine lubricating oil from engine.
2. Temperature of engine lubricating oil into reservoir.
3. Temperature of engine lubricating oil into engine.
4. Temperature of fuel out of cooler.
5. Temperature of fuel out of pumps.
6. Temperature of fuel in tank.
7. Engine fuel flow to nozzles.

NOTE: Fuel to eng. should not exceed 225°F as it tends to "coke" system. Advancing throttle and decreasing altitude should lower temp. Pilot indicator needed.

Requested by Thermo Group

Approved by
For T. L. Maloy

TLM:MC:gh

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F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Date: 4 August 1954

Report No. ZC-8-103
Addendum No. 17

Airplane No.: S/N 52-7995

Item Description:

Engine Lubricating Oil Temperatures with and without Air-Oil Cooler.

Purpose:

To determine whether or not the engine lubricating oil can be cooled adequately with only the fuel-oil cooler.

Configuration:

Completely by-pass air-oil cooler.

Procedure:

Measure the following oil and fuel temperatures throughout a typical high altitude flight.

Measurements Required:

1. Temperature of engine lubricating oil from engine.
2. Temperature of engine lubricating oil into reservoir.
3. Temperature of engine lubricating oil into engine.
4. Temperature of fuel out of cooler.
5. Temperature of fuel out of pumps.
6. Temperature of fuel in tank.
7. Engine fuel flow to nozzles.

NOTE: Fuel to eng. should not exceed 225°F as it tends to "coke" system. Advancing throttle and decreasing altitude should lower temp. Pilot indicator needed.

Requested by Thermo Group

Approved by
For T. L. Maloy

TLM:MT:gh

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F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Date: 4 August 1954

Report No. ZC-8-103
Addendum No. 18

Airplane No: 52-7995

Item Description:

Aft section cooling test.

Purpose:

To investigate the effects of the distribution duct and auxiliary air-oil cooler on after section cooling air flow.

Configurations to be tested:

1. Manifold louvers and oil cooler "open".
2. Manifold louvers "open" and oil cooler "closed".
3. Manifold louvers "closed" and oil cooler "open". (Oil cooler shutter controllable from pilot's compartment, louvers to be closed by clamps lined with asbestos for ease of removal and duct to be plugged with stopper that does not damage duct.)

Procedure:

1. Record pressure, temperature, altitude and Mach number for an entire typical flight -- Maximum power climb to ceiling, one minute at maximum power level flight, and dives at speeds of Mach 1.1 to Mach 1.3.
2. Repeat for configurations (2) and (3).

Measurements Required:

1. P_{static} and P_{total} at oil cooler air scoop.
2. P_{static} at station 480 between skin and shroud.
3. P_{static} at station 500 between shroud and engine.
4. P_{total}, back to back at three positions, station 621.5.
5. Record aft end temperatures where instrumented (structure, actuator, shroud and air.)

Requested by Thermo Group

Approved by
T. L. Maloy

TLM:NR:gh

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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 19

Item Description: Instrument Panel Vibration Test

Purpose: To determine compliance with Par. 3.2.8 of MIL-I-5997 (USAF) which establishes vibration requirements for the instrument panel to prevent sticking of instruments.

Procedure:

1. Paragraph 3.2.8 of MIL-I-5997 reads as follows:

> " The amplitude of vibration of the instrument board shall be between the limits of 0.003 and 0.005 inch in at least one direction, and shall not exceed 0.005 inch in any direction with the airplane in normal flight. NOTE: An auxiliary panel vibrator or tapper shall be provided if necessary to increase the vibration to 0.003 inch in at least one direction."

2. Instrumentation shall be installed to record the vibration in the longitudinal, vertical, and transverse directions. Pickups should be installed on both the center and R.H. portions of the instrument panel.

3. No special flights are required. The required data may be obtained concurrently with other flights covering a normal range of flight conditions

4. If the vibration is less than .003 inch, a panel vibrator shall be installed, and the tests repeated.

Measurements Required: Record the following:

1. Airspeed
2. Altitude
3. N_2, high rotor speed
4. Vibratory amplitude (3 axes). Two panel positions. Instr. Req. 82-27 (Ref.).
5. Vibratory frequency (3 axes). Two panel positions. This item desirable but not mandatory. Instr. Req. 82-27 (Ref.).

NOTES:

1. Schedule: Soon as possible
2. Requested by AF Ltr. dated 1 June 54. Copy of results requested by Model 8 Proj. Off. AVO dated 20 June 54, from C. B. King and W. W. Fox (cc:J.C.Hoover)

Approved by
H. K. Cheney
Chief Flight Test Engineer
Experimental Flight

HKC:JNS:gh

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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-102
Addendum No. 20

Item Description: Airspeed Calibration

Purpose:

To obtain data for airspeed calibration by three separate procedures. This new calibration is necessitated by drag reduction modifications of the aircraft.

Airplane Configuration:

Airplane modified by sharp nose, yellow canary, and items listed in minutes - Post Flight Meeting, 22 July, 1954.

Procedure A:

Pacer calibration of total and static pressure indication at subsonic speeds using USAF F-86E calibrated pacer aircraft.

1. The following tests should be performed at 15,000 feet altitude.

Configuration	Speed Range (IAS)	Speed Increments
Clean	150-500 MPH	20 MPH
Gear Extended	150-275 MPH	10 MPH

2. For comparison of data, it is suggested that some of the above tests be performed using the same pacer as was used for the previous calibration (F86, Ser. No. 582)

Procedure B:

Static pressure calibration for transonic speeds using three static orifices as reference during performance of the specified test flights.

1. Install 3 fuselage static pressure sources on the fuselage in accordance with drawing 8-00881. Connect each static port to a separate altimeter on the photo panel. Connect the middle static port also to a separate altimeter on the pilot panel.

2. Perform the following flight plan for obtaining the required calibration data for transonic and supersonic calibration:

   a. Climb to about 43,000 feet altitude and dive with a sufficiently large dive angle to reach a maximu Mach No. of about 1.2 during the dive.

   b. Pull out at about 34,000 feet altitude into level flight.

Procedure B: - Continued

2. Continued

c. Maintain constant altitude using the altitude indicated by the middle static port on the fuselage nose.

d. During this level flight decelerate slowly to an indicated Mach No. of M_i = 0.80 or less (approximate time to decelerate from M_i = 1.08 to M_i = 0.80 may be 30 seconds)

e. Record the photopanel data on high speed motion picture camera in the Mach number range between M_i = 1.08 and M_i = 0.80 or less.

f. Repeat the above procedure with pull out at about 27,000 feet altitude.

3. Evaluation of the data should be based on the method described in Report ZC-8-501-SR2 (ref.).

Procedure C:

Static pressure calibration at subsonic, transonic, and supersonic speeds using radar calibration during level flight decelerations and a dive to supersonic speeds.

1. The following flight plan shall be accomplished, and the flight pattern shall be in accordance with Figure 1:

a. Climb to altitude in the most convenient manner and do not require radar tracking during the climb. This permits the pilot to make an optimum climb without attempting to navigate within radar range.

b. Obtain all radar tracking data in a sequence of passes above highway No. 466 and within 15 miles of the radar installation, as illustrated by Figure 1.

Measurements Required:

Same as for previous calibration tests, except install additional static pressure sources and altimeters as described in paragraph 1 of Procedure B.

Approved by H. K. Cheney
H. K. Cheney
Chief Flight Test Engineer
Experimental Flight

HKC:JNS:gh

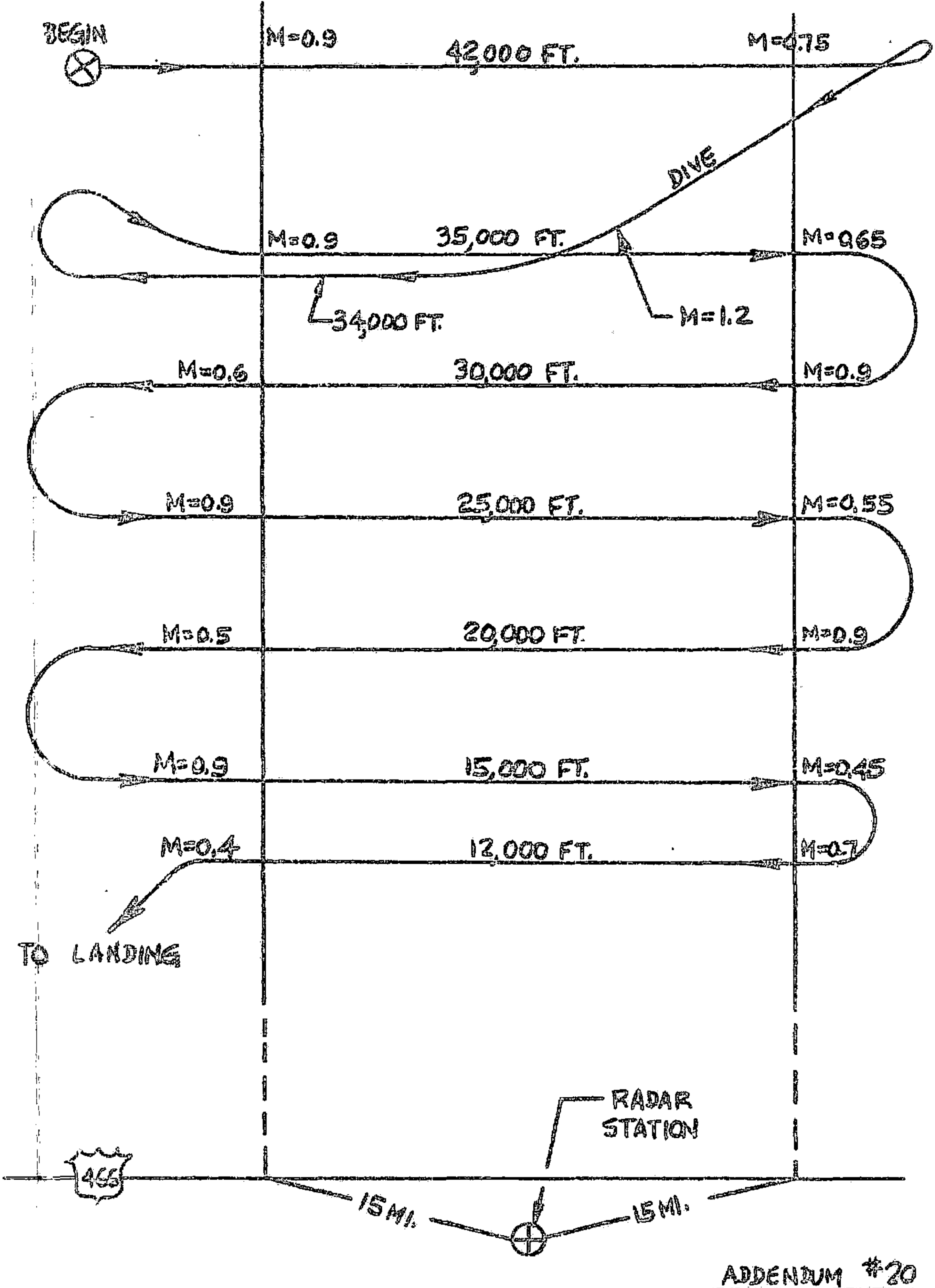
BEGIN
M=0.9
42,000 FT.
M=0.75
DIVE
M=0.9
35,000 FT.
M=0.65
34,000 FT.
M=1.2
M=0.6
30,000 FT.
M=0.9
M=0.9
25,000 FT.
M=0.55
M=0.5
20,000 FT.
M=0.9
M=0.9
15,000 FT.
M=0.45
M=0.4
12,000 FT.
M=0.7
TO LANDING
RADAR
STATION
466
15 MI.
15 MI.

FIGURE 1

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Sheet 1 of 3

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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-10~~2~~3
Addendum No. 20

Item Description: Airspeed Calibration

Purpose:

To obtain data for airspeed calibration by three separate procedures. This new calibration is necessitated by drag reduction modifications of the aircraft.

Airplane Configuration:

Airplane modified by sharp nose, yellow canary, and items listed in minutes - Post Flight Meeting, 22 July, 1954.

Procedure A:

Pacer calibration of total and static pressure indication at subsonic speeds using USAF F-86E calibrated pacer aircraft.

1. The following tests should be performed at 15,000 feet altitude.

Configuration	Speed Range (IAS)	Speed Increments
Clean	150-500 MPH	20 MPH
Gear Extended	150-275 MPH	10 MPH

2. For comparison of data, it is suggested that some of the above tests be performed using the same pacer as was used for the previous calibration (F86, Ser. No. 582)

Procedure B:

Static pressure calibration for transonic speeds using three static orifices as reference during performance of the specified test flights.

1. Install 3 fuselage static pressure sources on the fuselage in accordance with drawing 8-00881. Connect each static port to a separate altimeter on the photo panel. Connect the middle static port also to a separate altimeter on the pilot panel.

2. Perform the following flight plan for obtaining the required calibration data for transonic and supersonic calibration:

   a. Climb to about 43,000 feet altitude and dive with a sufficiently large dive angle to reach a maximu Mach No. of about 1.2 during the dive.

   b. Pull out at about 34,000 feet altitude into level flight.

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Procedure B: - Continued

2. Continued

c. Maintain constant altitude using the altitude indicated by the middle static port on the fuselage nose.

d. During this level flight decelerate slowly to an indicated Mach No. of M_i = 0.80 or less (approximate time to decelerate from M_i = 1.08 to M_i = 0.80 may be 30 seconds)

e. Record the photopanel data on high speed motion picture camera in the Mach number range between M_i = 1.08 and M_i = 0.80 or less.

f. Repeat the above procedure with pull out at about 27,000 feet altitude.

3. Evaluation of the data should be based on the method described in Report ZC-8-501-SR2 (ref.).

Procedure C:

Static pressure calibration at subsonic, transonic, and supersonic speeds using radar calibration during level flight decelerations and a dive to supersonic speeds.

1. The following flight plan shall be accomplished, and the flight pattern shall be in accordance with Figure 1:

a. Climb to altitude in the most convenient manner and do not require radar tracking during the climb. This permits the pilot to make an optimum climb without attempting to navigate within radar range.

b. Obtain all radar tracking data in a sequence of passes above highway No. 466 and within 15 miles of the radar installation, as illustrated by Figure 1.

Measurements Required:

Same as for previous calibration tests, except install additional static pressure sources and altimeters as described in paragraph 1 of Procedure B.

Approved by H. A. Cheney
H. A. Cheney
Chief Flight Test Engineer
Experimental Flight

HAC:JNS:gh

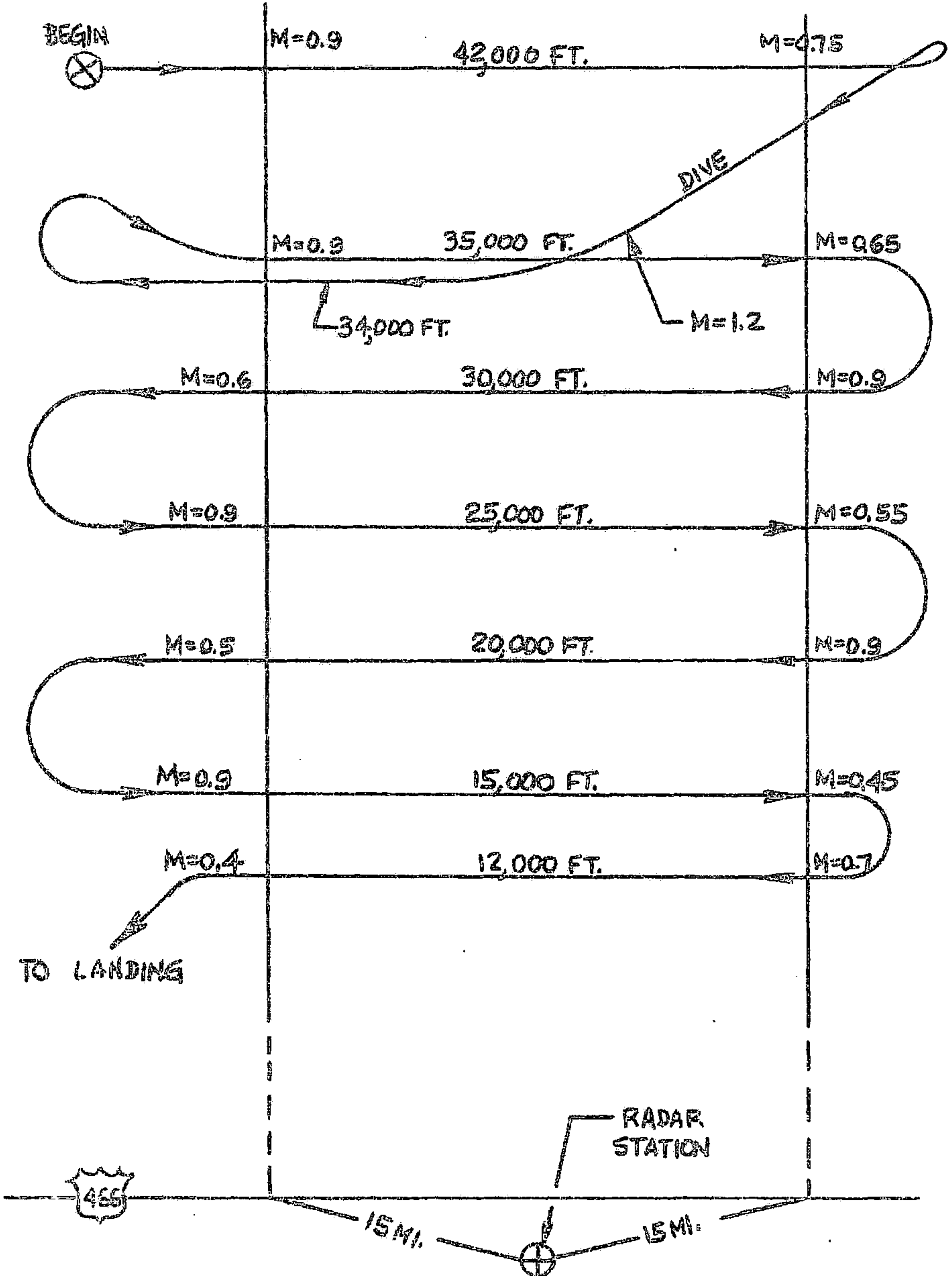
BEGIN
M=0.9
42,000 FT.
M=0.75
DIVE
M=0.9
35,000 FT.
M=0.65
34,000 FT.
M=1.2
M=0.6
30,000 FT.
M=0.9
M=0.9
25,000 FT.
M=0.55
M=0.5
20,000 FT.
M=0.9
M=0.9
15,000 FT.
M=0.45
M=0.4
12,000 FT.
M=0.7
TO LANDING
RADAR STATION
466
15 MI.
15 MI.

FIGURE 1

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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

3
Report No. ZC-8-102
Addendum No. 21

Item Description: Drag Characteristics

Purpose:

To establish a procedure and desirable sequence of operations for the obtainment of pertinent drag and calibration data.

Airplane Configuration:

Airplane modified by sharp nose, yellow canary, and items listed in minutes of Post Flight Meeting, 22 July 1954.

Procedure:

1. First priority is to obtain supersonic drag characteristics. The data required will be obtained during a series of dives. The procedure of these dives is: a) Climb to approximately 43,000 feet altitude and dive at a sufficiently large dive angle to reach a maximum Mach No. of about 1.2 during the dive, b) Pull out at specified altitude to a level flight condition, c) Maintain constant altitude using the altitude indicated by the middle static port on the fuselage nose, d) During this level flight region, decelerate to subsonic speeds approaching $M_i = 0.80$.

   Three dives are requested. The pullout altitudes are 34,000 feet and 27,000 feet and approximately 20,000 feet. The first two dives are as requested under procedure B of Addendum 20.

2. Second priority is to determine the subsonic drag level. This is best accomplished by low altitude flight (15,000 feet) in the region of $M = 0.60$ to $M = 0.85$ (W/δ = 46,000Lb. or less) in increments of $\Delta M = 0.05$.

   The evaluation of these data will be based on pacer type airspeed calibration which must be obtained concurrently or prior to this schedule.

3. Third priority is evaluation of the drag rise region. This requires stabilized data from maximum level flight Mach numbers to approximately $M = 0.85$ in increments of $\Delta M = 0.025$. Such data should be obtained from flight at three altitudes, preferably 15,000, 25,000 and 35,000 feet. (Corresponding W/δ values are 46,000; 78,000 and 110,600 lbs.)

Procedure: - Continued

4. Fourth priority is to fly progressive stabilized speed - power data at approximately 35,000 feet from Vmax to M = 0.55 in increments of $\Delta M = 0.05$ ($W/\delta \simeq 110,600$). In addition two manuevering flight conditions are requested at 1.6 g's for: a) M = 0.70 and b) M = 0.80. The altitude for these turns is 25,000 feet at a gross weight of 26,000 lbs. ($n\,W/\delta \simeq 112,000$).

Measurements Required:

Refer to Experimental Flight Memo EF-8-257, subject "Measurements Required for First Flights After Modification of S/N 7995".

Symbols:

n = Manuevering load factor, g's
W = Gross weight, lbs.
δ = Ambient pressure ratio

Reference:

1. Addendum 9-A, "Yellow Canary Evaluation".

2. Addendum 11, "Drag Data Test".

3. Addendum 20, "Airspeed Calibration".

Approved by T. L. Maloy
T. L. Maloy
Group Engineer
Experimental Flight

TLM:JEB:gh

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YF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 14A

Item Description: Longitudinal Sensitivity

REVISED & REWRITTEN
19 October 1954

PURPOSE

Tests outlined herein are for the solution of control problems regarding the sensitivity, friction, stick lightening and stick force slope of the longitudinal control system.

AIRPLANE CONFIGURATION

Revised configuration, including elongated nose and yellow canary. Test with standard cam installed in Shelley Unit.

PROCEDURE

A. Elevator to Trim Test

1. From $M = 0.8$ and 45,000 feet, increase speed to $M \cong 1.0$ in straight and level 1.0g flight. This is to check any change in trim resulting from the new configuration changes.

B. Hysteresis Ground Test

1. Hysteresis data for elevator, aileron and rudder system should be obtained, and the lag in the trim system determined. The recommended procedure is outlined in DC-8-196, which was distributed to EAFB.

C. Shelley Unit Evaluation (Use Standard Cam)

1. For the Shelley Unit evaluation, data is required covering the variation of Shelley Unit "Q" pressure with Mach No. The response characteristics of the Unit should also be evaluated. Data shall be obtained over a large Mach No. range in conjunction with other flights. Special flights are not required.

2. To complete the data for the design of the Shelley Unit cam, flights shall be conducted as follows, for determination of "Elevator and Stick Force per G".

Approx. Vertical Load Factor Hold "Steady State" Approx. 5 Seconds	Approx. Altitude (Minimum Variation During "Steady State")	Mach. No. Hold Constant During "Steady State"
2g, 3g	45,000	.80, .90, .95, 1.00
2g, 3g	35,000	.65, .80, .90, .95, 1.00, 1.20
2g, 3g	25,000	.90, .95, 1.00

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C. Continued

NOTES:

a. Establish 1.0g trim for test Mach No. and do not operate trim during maneuver.

b. Pitch damper should be disengaged.

D. Surface-Stick Position, and Operating Characteristics During 1 to 2 Minutes Operation at Constant Altitude and Airspeed for Flight Conditions of Critical Longitudinal Sensitivity.

1. Pilot should search for and hold the critical condition, and shall establish the time where sensitivity commences. The records will be inspected to determine, if possible, the source of the previously reported sensitivity problem.

MEASUREMENTS REQUIRED

Basic flight control measurements, including "Q" pressures.

Requested by Dynamics

Approved by
for E. D. Shannon
Chief of Experimental Flight

EDS:JNS:sgh

ZC-8- 103 FLT. TEST PROGRAM
S/N 7995

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11120

FLIGHT TEST PROGRAM ADDENDUM FORM

UNCLASSIFIED

Report No.: [illegible]
Addendum No.: [illegible]

Airplane No.: 52-7995

Revised and rewritten [illegible]
Reason: To include [illegible]

Item Description:

Afterbody Fillet (Yellow Canary), afterburner nozzle actuator and Ram air inlet tests.

Purpose:

To obtain pressure and/or temperature data for the above items.

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by: Dept. 430 Date 4-4-67

Configuration: Noted

Procedure:

A. Yellow Canary Evaluation

1. Install pressure pickups [illegible] in accordance with Thermodynamic Memo TA-178 dated 12 July 1954.

2. Two pressure pickups and a single temperature pickup are located on the inner surface of the yellow canary, aft of the engine EXIT. These should be recorded during one ground engine run up (afterburner included), and flight to maximum altitude with afterburner, in addition to the flights of paragraph A3.

3. All pressure and temperature pickups should be recorded simultaneously under the following flight conditions:

   a. Level flight at 15,000 feet, covering the full Mach Number range.

   b. Level flight at 30,000 feet, covering the full Mach Number range.

   c. Flight at supersonic velocity for at least 30 seconds. It is requested that Mach No. 1.1 or better be maintained. Record during preliminary dive, and during the subsequent transition to subsonic velocity.

UNCLASSIFIED

[redacted]

[illegible] (Continued)

d. Record angle of attack and elevon surface position for the above flights. These are expected to affect the pressures.

B. Temperature of Afterburner Nozzle Actuators:

1. Install temperature pickups on upper and lower surface centerline afterburner nozzle actuators. Obtain temperature data before and after shielding, for a ground engine run up (afterburner included) and flight to maximum altitude with afterburner.

NOTE. This is for record only. Tests have been accomplished.

C. Fin Ram Air Inlet Pressure and Temperature Measurements:

1. ECA 8-6225 calls for the installation of a ram air cooling duct on the leading edge of the fin. When this is accomplished, instrumentation should be added to obtain airflow and ambient air temperature at the lip of the inlet duct.

2. The flight should include a climb to maximum altitude as well as a maximum Mach number dive.

Measurements Required:

Item	Range	Reference
Hollow canopy pressures	0-200PSF	Thermo Memo No. T-I-178
Hollow canopy temperature	0-1100°F	Thermo Memo No. T-I-178
Temperature at Afterburner Nozzle Actuators (accomplished)	0-1100°F	Instr. Req. No. 32-15
Ram Air Inlet Stat. Pressure	0-30"HgA	Instr. Req. No. 32-16
Ram Air Inlet Diff. Press (Total-static)	0-25"HgA	Instr. Req. 32-18
Ram Air Inlet Ambient Air Temp.	0-175°F	Instr. Req. 32-26

Approved by:
[illegible]

Requested by: Thermo Group

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[redacted]

F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995 Report No. ZC-8-103

Addendum No. 12

Item Description: Preliminary ram air turbine tests.

Purpose: To test the ability of the turbine to maintain adequate hydraulic pressure during landing conditions. The landing condition is the most critical for the turbine because of reduced "Q" pressure.

Procedure:

1. Install a pressure pickup in the pressure line between the ram air turbine and the flow control valve.

2. Extend the ram air turbine just prior to normal take-off from the concrete runway, and retract the landing gear after take-off.

3. Circle the base at moderate speed and perform a normal approach and landing. In order to get maximum interference with the airstream ahead of the turbine, withhold the lowering of the gear until the latest practicable moment.

Measurements Required:

*1. Ram air turbine hydraulic pump pressure. (0-3500 psi).
2. Primary hydraulic system pressure.
3. Primary hydraulic reservoir fluid temperature.
4. Rudder and elevon surface position.
5. True air speed.

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by: Dept. Date
L. G. Avritt 130-1 4-4-67

Notes:

1. Do not uncouple primary or secondary hydraulic power for above tests. Flight test Addendum Item No. 13 contains a more detailed test procedure and will be forwarded when approved.

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[redacted]

2. Test with TURB 7 (replaces TURB 6).

3. This test is similar to, and replaces test per ZC-8-103, Index No 4.1, page 48.

*4. Additional instrumentation required.

Approved by H. Cheney
H. Cheney
Chief Flight Test Engineer
Experimental Flight

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[redacted]

FILE

F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

UNCLASSIFIED

Report No. ZC-8-103
Addendum No. 11

Item Description: Drag data test.

Purpose: To determine drag effects in the region of Mach 0.90.

Procedure: Perform flights as follows:

Test No.	Mach No.	Load Factor	Altitude	GW 26,000 Corr. Wt. (W/Delta)
1	.90	1G	47,500	200,000
2	.85	2G	33,000	200,000
3	.80	2G	33,000	200,000
4	.90	1.5G	44,700	263,000
5	.85	1.5G	44,700	263,000
6	.80	1.5G	44,700	263,000
7	.90	2G	41,500	300,000

Measurements Required: Same as for previous drag data tests per Note. 2.

NOTES:

1. This addendum is confirmation of advance wire to V. L. Allwardt, EAFB, dated 6-11-54 (copy to W. Redd).

2. The above tests are for completion of drag data tests initiated during the Phase II testing of 52-7995 at EAFB.

3. Point No. 7 may be marginal power-wise. If level flight cannot be maintained at 2G, themaximum load factor for level flight should be used.

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by: Dept. 130-1 Date 4-4-67
L. G. Avitt

Approved by T. L. Maley
T. L. Maley
Group Engineer
Experimental Flight

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TLM:lp

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[redacted] UNCLASSIFIED

CONVAIR

A division of General Dynamics Corporation

San Diego

F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No. 52-7995 or 53-1781

Report No. ZC-8-103

Addendum No. 13

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
54 KELLY AFB TEX.
Reclassified by: Dept. Date
L. G. Avritt 1631 4-4-67

ITEM DESCRIPTION: Simulated landings with ram air turbine emergency hydraulic power.

PURPOSE: To test the ability of the ram air turbine to furnish adequate hydraulic power for emergency landings. In addition, the tests described herein will test the ability of the primary system alone, or the secondary system alone to properly control the aircraft.

CONFIGURATION: Test with TURB 7 (replaces TURB 6)..Modify hydraulic systems in accordance with Figure 1 of this addendum.

CAUTION: Addendum No. 12 has been prepared to obtain preliminary information prior to modification in accordance with Figure 1 herein. The flight test results of Addendum No. 12 should be evaluated and approved before the tests of this addendum are conducted.

PROCEDURE:

1. The valves and switches of Figure 1 should be installed and operated sufficiently to insure proper functioning of these components. The hydraulic ground test stand may be used to furnish pressure for the initial testing, but a ground run test of the aircraft engine-driven Vickers pumps should also be performed before the first flight is attempted.

2. Take off and climb to 30,000 feet and perform the following tests:

   a. Shut off primary system pressure, and operate the aircraft with secondary pressure alone. Perform sufficient maneuvers to demonstrate adequate control of the aircraft (see Note 1).

   b. Restore the primary pressure, and operate the aircraft with primary alone by shutting off the secondary pressure and repeating the maneuvers of paragraph 2a above.

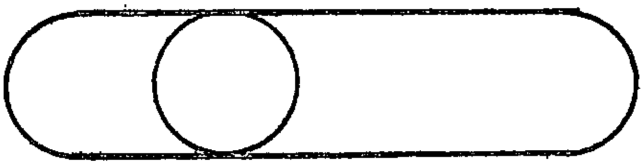
   c. Restore secondary pressure, and extend the ram air turbine at 30,000 feet and Mach. No. 0.5 in order to test the performance of the turbine extension

[redacted] UNCLASSIFIED

mechanism. Perform a normal power approach and landing.

d. After the foregoing preliminary tests are conducted, the aircraft should be placed on jacks, and an emergency gear drop performed with pneumatic pressure, to insure that the emergency system is functioning properly. The use of this system is required for the second flight.

e. For the second flight, a smoke laying aircraft shall take off along with the F-102, and shall construct a landing strip outline in smoke 6000 feet from the ground (approx. 8100 feet altitude at EAFB)..Shape of the strip should be approximately as follows:



Size of the strip must take into account the turn radius capability of the smoke laying aircraft.

f. The F-102 should climb to 30,000 feet while the smoke outline is being completed. When this altitude is reached, shutoff the primary system hydraulic pressure. Extend the ram air turbine at 30,000 feet and Mach No. 0.5. Observe the primary system pressure gage in the cockpit to see that the ram air turbine unit developes at least 2700 psi hydraulic pressure. Shut off the secondary pressure and perform mild maneuvers of the aircraft with RAM AIR TURBINE POWER ONLY at 30,000 feet and Mach No. 0.5 After the adequate control of the aircraft has been established, gradually cut the engine power to the idle position, but do not permit the engine to flame out.

g. A glide should now be performed from 30,000 feet to the smoke landing strip of paragraph d. The glide should be regulated so that an indicated airspeed of 225 knots is obtained when the altitude reduces to 14,000 feet. At this point the gear should be extended by use of EMERGENCY PNEUMATIC PRESSURE. This will test the capacity of the RAT powered control surfaces to maintain stability in the glide condition during the gear drop. The gear must now remain extended for the duration of the flight.

h. Continue the glide at 225 knots, and make a normal approach and "landing" on the smoke landing strip of paragraph d, with a "touchdown" speed of approximately 140 knots indicated. Restore engine thrust before aircraft speed drops below 125 knots indicated. NOTE: Maximum recovery speed with gear extended is 240 knots indicated. Restore the primary and secondary system hydraulic pressure, and perform a normal landing on the base.

3. A ground landing with ram air turbine power alone, or a complete engine flameout condition should not be attempted without proper authorization.

MEASUREMENTS REQUIRED:

1. RAT hydraulic pump pressure (0-3500 psi)
2. Primary system hydraulic pressure
3. Secondary system hydraulic pressure
4. Primary hydraulic reservoir fluid temperature
5. Rudder and elevon surface position
6. RAT speed, RPM (proposed)
7. Normal acceleration at C.G.
8. Roll rate
9. Angle of attack
10. Angle of sideslip (yaw)
11. Air speed
12. Altitude
13. Time

NOTES:

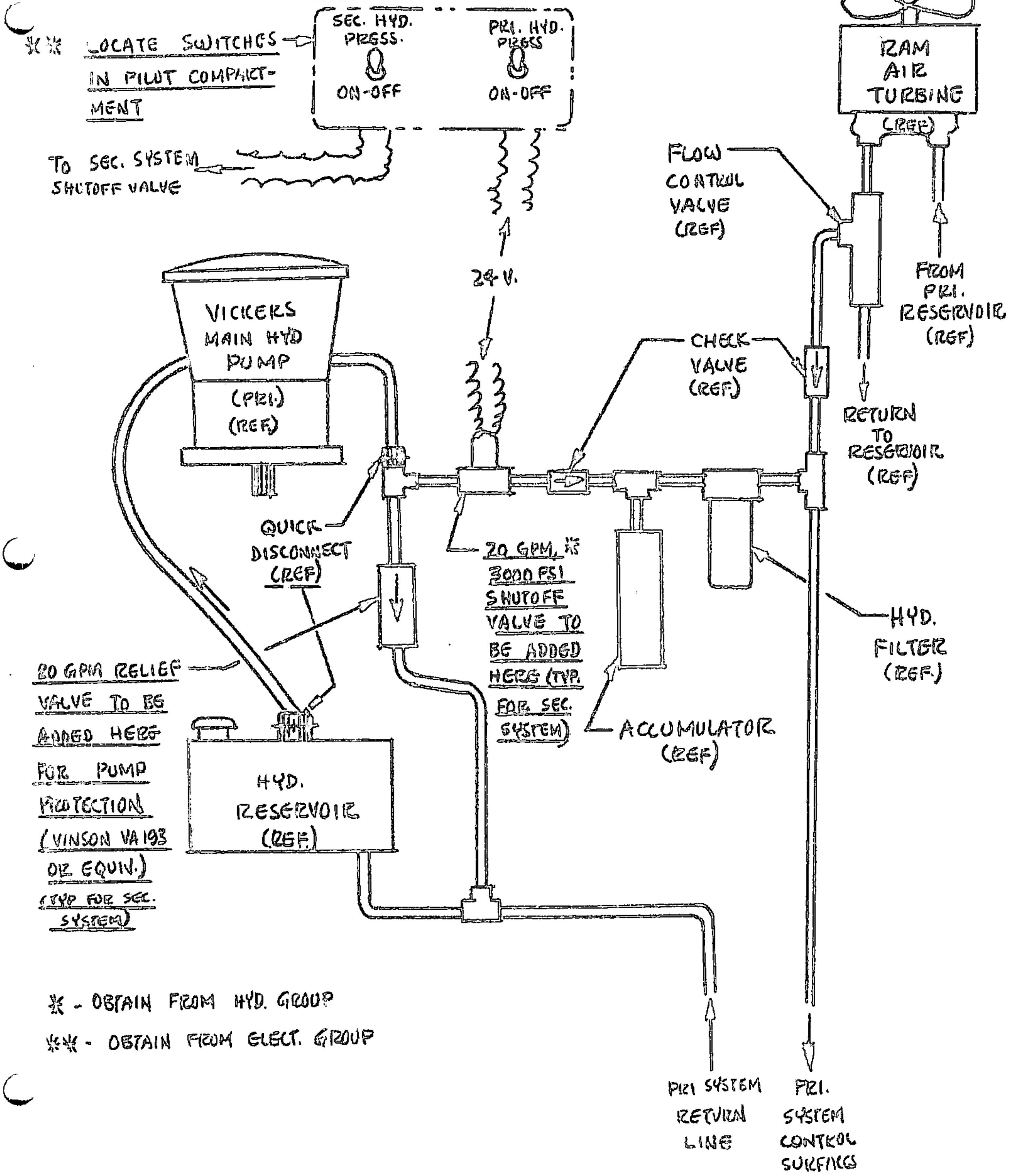
1. Full or severe maneuvers should not be attempted with primary pressure alone, secondary pressure alone, or ram air turbine pressure alone. The critical requirement should be considered as a landing under gusty air conditions. The pilot should estimate the maneuvering required to simulate this condition.

2. Complete chase plane photographs of the ram air turbine maneuvers should be obtained.

Approved by J. S. Maloy
For H. K. Cheney
Chief Flight Test Engineer
Experimental Flight

FIGURE 1
ZC-8-103
ADDENDUM NO. 13

SCHEMATIC DIAGRAM FOR RAM AIR TURBINE EMERG. LANDING TESTS


** LOCATE SWITCHES IN PILOT COMPARTMENT
SEC. HYD. PRESS. ON-OFF
PRI. HYD. PRESS ON-OFF
TO SEC. SYSTEM SHUTOFF VALVE
24 V.
RAM AIR TURBINE (REF)
FLOW CONTROL VALVE (REF)
FROM PRI. RESERVOIR (REF)
VICKERS MAIN HYD PUMP (PRI.) (REF)
CHECK VALVE (REF)
RETURN TO RESERVOIR (REF)
QUICK DISCONNECT (REF)
20 GPM, * 3000 PSI SHUTOFF VALVE TO BE ADDED HERE (TYP. FOR SEC. SYSTEM)
HYD. FILTER (REF.)
20 GPM RELIEF VALVE TO BE ADDED HERE FOR PUMP PROTECTION (VINSON VA 193 OR EQUIV.) (TYP FOR SEC. SYSTEM)
ACCUMULATOR (REF)
HYD. RESERVOIR (REF)
PRI SYSTEM RETURN LINE
PRI. SYSTEM CONTROL SURFACES

* - OBTAIN FROM HYD. GROUP

** - OBTAIN FROM ELECT. GROUP

CONVAIR
A Division of General Dynamics Corporation
San Diego

F-102 FLIGHT TEST PROGRAM ADDENDUM IT

UNCLASSIFIED

Report No. ZC-8-103

Airplane No. 52- 7995

Addendum No. 14

Item Description:

Short period longitudinal oscillation test

Purpose:

(1) To determine whether or not fluctuating type variations in elevator Q pressure are obtained due to shock effects at the pitot tube entrance, or other factors such as unstable operation of the q pressure regulating unit, or sticking of the Q pot.

(2) To record the relationship of such effects on the reported spuriousness of the eleven control forces under certain flight conditions.

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
KELLY AFB TEX.
Reclassified by: L. G. Avritt Dept. 130-1 Date 4-4-67

Spec. No.: None

Configuration: Airesearch Q pressure regulator D 254

Procedure:

(1) Install instrumentation in accordance with Sheet 2, and climb to 40,000 feet. Perform a supersonic dive and pullout with some banking, at approximately 2.0 g's.

(2) During this maneuver the pilot should search for spurious control forces, as noted from previous flights. This condition reportedly causes difficulty in holding the desired "g" force due to extraneous oscillations.

(3) Search shall be concluded if this condition cannot be obtained after several supersonic dives with pullouts ranging from 15,000 to 25,000 feet, and positive accelerations up to 3.5 g's.

(4) Where the spuriousness can be demonstrated, the flight conditions shall be repeated on later aircraft to the extent practicable, in order to demonstrate the possible effect of improved "Q" Pots and Shelley Waits.

Measurement Required:

See Sheets 2 and 3.

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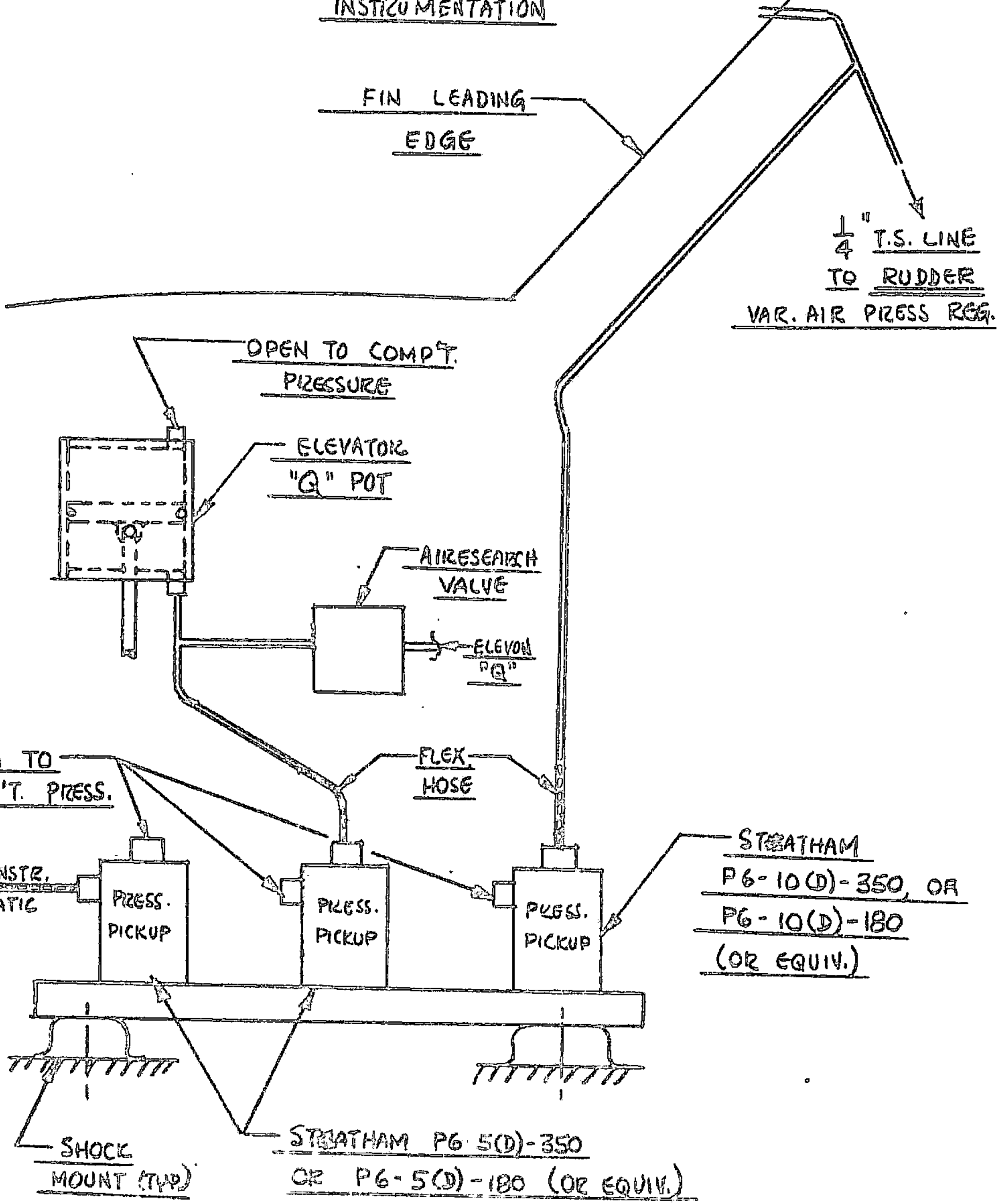
SHEET 2 OF 3

22 June 1954

SHEET 2

ADDENDUM ITEM #14-ZC-8-103

INSTRUMENTATION



Sheet 3 of 3

Report ZC-8-103

Addendum No. 14

Airplane: 52-7995

22 June 195[illegible]

Measurements (cont'd):

Record the following:

Oscillograph

1. Pitch attitude
2. Longitudinal stick position
3. Longitudinal stick force
4. Left and right elevon positions
5. Normal accel. at the pilots compartment and the C.G.

Photo-Panel

1. Airspeed
2. Altitude
3. Normal acceleration
4. Left and right elevon positions
5. Outside air temperature
6. Angle of attack
7. Time

The following are desirable, but not mandatory:

1. Pitch rate
2. Longitudinal acceleration at the C.G.
3. Lateral accel. at the pilots compartment and C.G.
4. Angle of sideslip (yaw)

NOTES:

1. Pending completion of instrumentation as shown on sheet 2, a preliminary flight should be performed to obtain rough data at the earliest practicable date. This can be accomplished by connecting the Q Pot differential pressure instrument (photo-panel) to the pressure line entering the Q Pot.

Approved by: T. L. Maloy

T. L. Maloy

Group Engineer

[redacted]

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CONSOLIDATED VULTEE

DIVISION San Diego

MODEL 8 (YF-102)

REPORT ZC-8-103

DATE 25 November 1953

TITLE

MODEL YF-102 SERIAL NO. 52-7995

FLIGHT TEST PROGRAM

SUBMITTED UNDER

Contract No. AF33(600)-5942

GR. 4
DOWNGRADED AT 3 YEAR INTERVALS; DECLASSIFIED AFTER
12 YEARS.
DOD DIR 5200.10

PREPARED BY R. W. Bonney
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GROUP EXPERIMENTAL FLIGHT

REFERENCE

APPROVED BY T. L. Maloy
T. L. Maloy
Group Engineer

NO. OF PAGES 77

NO. OF DIAGRAMS

Approved W. M. Fox
for E. D. Shannon
Chief of Experimental Flight

REVISIONS

NO	DATE	BY	CHANGE	PAGES AFFECTED
A	17 Jun 54	J. Steinmetz	Revised Pages 7, 25 and 31	

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FOREWORD

This flight test program describes the testing to be conducted by Convair on the second YF-102 airplane, Serial No. 52-7995. Tests will be conducted during the period of December 1953 through Sept. 1955. A total of 165 flight hours will be required to accomplish this flight test program.

The purpose of these tests is to demonstrate and evaluate the basic flying and operating qualities of the airplane in addition to airplane and engine performance, stability and control, and to develop the pilot assist system for the F-102 airplane.

This flight test program is authorized by Air Force Contract No. AF33(600)-5942, Supplemental Agreement No. 3. This program will, when approved, supersede all previous proposed programs for flight testing of this airplane and the first YF-102 airplane, Serial No. 52-7994.

SCOPE

The YF-102 airplane, Serial No. 52-7995, for which this flight test program has been compiled, is the second F-102 airplane to be constructed. It is powered by a Pratt & Whitney J-57-P-11 turbojet engine with afterburner.

Instrumentation for this airplane will be phased in during the flight test program, i.e., source instrumentation and circuitry for all of the measurements will be installed prior to the first flight, but only the end instrumentation necessary for measurement of data required during the early part of the flight test program will be installed. This is necessary due to the heavy load of tests and instrumentation on this airplane.

The primary assignements of this airplane are as follows:

1. Flutter and vibration
2. Airplane and engine performance
3. Dynamic response
4. Stability and control
5. Pilot assist system development
6. Phase III tests (engine and associated equipment refinement tests)

In addition, preliminary Phase II tests will be conducted by AFFTC as early in the flight test program as is practicable. A complete list of the flight tests to be conducted is presented on page 5 of this report.

The Phase III tests, item 6 above, are to be conducted utilizing whatever time is available at the end of the flight test program. No attempt has been made to break down the details of the testing necessary as these tests are contingent on prior operating characteristics and/or deficiencies.

A detailed breakdown of the tests stating purpose, requirements and procedures for each test has been made and is presented on pages [illegible] through 50 of this report. The airplane instrumentation required to measure the necessary data during the tests has been tabulated in functional groups according to the type of test involved and is presented in this manner on pages 51 through 66. A tabulation of the flight tests to be accomplished in the approximate order in which they will be conducted, including the pertinent test parameters and the measurement groups required to obtain test data is presented on pages 67 and 68. A further organization of the flight test program into operational test phases including a time schedule showing the estimated flight hours and months required to accomplish each phase is presented in Appendix I. A breakdown of the

SCOPE (Continued)

testing to be accomplished by AFFTC during the Phase II tests has not been done as this portion of the test program is at the discretion of the testing agency.

This flight test program was compiled in accordance with operating limits and restrictions per the F-102 design specification, ZD-8-001.

Temporary restrictions will be imposed on this airplane until deleted by results of tests, design studies, or agreement. These restrictions will be imposed by Convair Engineering or the USAF. The temporary restrictions will be furnished to the operating agencies by Convair Engineering.

The airplane is not to be operated in excess of current operating limits or restrictions at any time during the flight test program.

SPECIAL TEST EQUIPMENT

Flutter and Vibration Tests

A hydraulic powered vibrator is installed, in each wing to produce structural vibrations during flutter and vibration tests. Each vibrator has a frequency range of 5 - 50 cps. The force produced by the vibrators is not a function of frequency and may be constant over a wide frequency range.

The vibrators have an automatic sweep over the frequency range and are operable either in phase or 180° out of phase. A stop control is capable of stopping the vibrators within one cycle of operation. The vibrator frequency, phase relationship, and stop control are all controllable from the cockpit. The vibrators are completely submerged within the wing.

Dynamic Response Tests

An electronic scheduling device referred to as "programmer" is installed in the airplane to supply an input signal to the control valve servo amplifier which causes a pre-determined flight control system operation during dynamic response tests. The various flight parameters are set into the "programmer" on the ground to produce a scheduled series of control system movements of specified rate, amplitude, and duration during flight. The "programmer" may also be set by the pilot during flight to produce a given control system operation.

The "programmer" will also be used during the dynamic stability demonstration tests to produce specified elevon and rudder deflections.

CONFIGURATION AND SYMBOLS

The following configurations and symbols are used in this report and are identical to those used in Air Force Requirements R1815-B.

Configurations:

CR	Cruise	Thrust for level flight at approximately speed for maximum range, speed brakes and gear up.
D	Dive	25% NRT, speed brakes extended.
G	Glide	Idling thrust, unless otherwise specified, speed brakes and gear up.
L	Landing	Idling thrust, gear down.
P	Power-on Clean	NRT, speed brakes and gear up.
PA	Power Approach	Thrust for level flight at 1.20 V_{S_L}, gear down.
WO	Wave-Off	T.O. Thrust, gear down.

Symbols:

V_S	True indicated stalling speed for $C_L = 0.75$ at the gross weight corresponding to the specified condition. Subscripts (PA, L, etc.) refer to the airplane configurations as defined above.
$V_{max.}$	Level flight speed with maximum available thrust (full after-burner operation).
$V_{max.}$ NRT	Level flight speed with normal rated thrust.

TESTS TO BE ACCOMPLISHED

1. Static Thrust Calibration
2. Brake Operation Evaluation
3. Taxi Tests
4. Engine Control System Operation
5. Shakedown Flight
6. Equipment Operation - Speed Brakes

Fuel Tank Pressurisation System Tests

8. Engine Equipment Operation
9. Flutter and Vibration
10. Airspeed Calibration
11. Preliminary Flight Characteristics
12. Angle of Attack and Angle of Yaw Pickup Calibration
13. Preliminary Airplane Performance
14. Phase II Tests (Conducted by AFFTC)
15. Effect of Missile Bay Door Operation on Flight Characteristics
16. Airplane Performance (Optimum Wing L.E. and Tip Configuration)
17. Engine and Structural Cooling
18. Dynamic Response
19. Stability and Control
20. Pilot Assist System Development
21. Engine Operation - Air Starts
22. Drogue Chute Tests
23. Emergency Fuel Control
24. Equipment Operation - Ram Air Turbine
25. Flight Control System Operation
26. Engine and Associated Equipment Operation (Phase III Tests)

Test: Static Thrust Calibration CVAC Index No. 2.7

Purpose: (1) To calibrate the in-flight engine thrust instrumentation.
(2) To determine installed power plant operating characteristics.

Requirements: Determine in-flight thrust instrumentation calibrations.

Procedure: 1. Stabilize engine RPM at settings listed below and record pertinent data.

a. Idle, 70, 80, 90, 92, 94, 96, 98, 100%, Mil RPM without afterburner.

b. Min. A/B thrust, 2 intermediate settings with A/B, and full A/B thrust.

Note: 1. The above runs are to be conducted without the ground air ejector.
2. The exit nozzle pressure survey swinging probe will be utilized during these runs.

[illegible] Brake Operation Evaluation CVAC Index No. 4.2

Purpose: To determine the relationship between brake cylinder pneumatic pressure and brake effectiveness.

Requirements: Determine deceleration and brake cylinder pressure during taxi braking and landing stops.

Procedure: (1) Brake operation and effectiveness will be evaluated by recorded brake cylinder pneumatic pressures, qualitative pilot's estimates of brake pedal forces, and deceleration records of the airplane during taxi tests and landing stops.

Test: Taxi Tests

Purpose:

(1) To determine ground handling, engine and structural cooling characteristics, and ground operation of all airplane equipment.

(2) To check operation of the engine, control system, brakes, radio, test instrumentation and all other equipment while taxiing the airplane at speeds up to take-off speed.

Requirements:

A. Engine Operating Characteristics

(1) Starting
(2) Power Control
(3) Cooling - static ground operation and all taxi speeds
(4) Fuel consumption
(5) Engine thrust required for breakout from standstill
(6) Engine thrust required to taxi airplane.
(7) Maximum engine thrust withheld by brakes
(8) Engine pressure, temperatures, and speed, to remain within the range of P & W specifications during all operations.

B. Airplane handling and control characteristics

(1) Flight control system operation
(2) Nose wheel steering
(3) Maximum engine thrust for nose wheel scrubbing
(4) Minimum turn radius
(5) Directional control satisfactory in cross-winds up to 30 MPH
(6) Minimum practical speed for nose wheel lift-off
(7) With the C.G. at the most forward position attainable with maximum gross weight it shall be possible to make a nose wheel lift-off at not more than ~~[illegible]~~ $V_{S_L} = 0.8$
(8) Effective lateral control at high taxi speeds
(9) Wheel brake operation, fading and effectiveness
(10) Drogue chute deployment and jettison

C. Airplane Equipment Operation

(1) Satisfactory operation of all hydraulic, pneumatic, electrical and radio equipment operable on the ground while taxiing the airplane at speeds up to take-off speed.

Procedure:

1. Conduct taxi tests at or near take-off gross weight. Turn all test instrumentation on during taxi tests.

2. Pilot will make pre-take-off cockpit check and taxi airplane at speeds up to 100 MPH. Check applicable items in requirements above.

3. Make simulated take off runs during which take-off speed is attained. Operate engine in normal and afterburner regimes. Check all applicable items in requirements above.

4. Procedure for these tests will be at the pilots discretion. Pilot should make oral comments to be recorded on the voice wire recorder during all runs.

Tests: Engine Control System Operation CVAC Index No. 3.10

Purpose: To demonstrate that the engine power control lever satisfactorily controls the engine power output.

Requirements: The power control lever must operate smoothly and easily over the entire range of power settings and must have no tendency to creep from a set position.

Take-off lock shall operate easily with positive lock position.

Procedure:

1. Measure power lever angle vs. engine RPM and turbine discharge total pressure for stabilized conditions during the static thrust calibration runs.
2. Measure power lever angle during in-flight speed-power runs.
3. General evaluation of power-control lever operation will be made from pilot's comments. Qualitatively note forces required to move power control lever and any creeping tendency.

Test: Shakedown Flights CVAC Index No. 1.12

Purpose: To determine general flight handling characteristics, engine operation and cooling, structural cooling and equipment operation.

Requirements:
Airplane response to flight controls displacement.
Engine and structural cooling.
Engine controls operation.
Low and high pressure pneumatic systems operation.
Hydraulic system operation.
Electrical system operation.
Preliminary air speed system check.
Cockpit cooling
Missile bay door operation

Notes: The primary purpose of the shakedown flight will be to conduct the initial flutter and vibration tests as outlined on page 13 of this report.
The pilot will qualitatively check the requirements listed above and record data as required.
The preliminary airspeed system check will consist of verbal airspeed checks between the pilot of the chase airplane and the pilot of the YF-102

Test: Equipment Operation - Speed Brakes CVAC Index No. 4.14

Purpose: To determine speed brake operation characteristics.

Requirements:

1. The speed brakes shall have acceptable smoothness and sensitivity.
2. The speed brakes shall not cause more than .5g deceleration at high speeds.
3. Speed brake retraction during gear extension shall not cause unfavorable trim characteristics.
4. Satisfactory static and dynamic directional and longitudinal stability must be maintained with speed brakes extended.

Procedure:

1. The pilot will modulate the speed brakes between full closed and open to determine smoothness and sensitivity of operation.

2. The pilot will obtain maximum practical speed in a dive. The speed brake switch will be held to open as the airplane speed decreases until full speed brake position is obtained.

3. The speed brakes will be fully extended and airspeed stabilized at maximum gear extension speed. The landing gear will then be extended and the speed brakes will automatically close.

4. With speed brakes partially extended at 300 mph V_i, the pilot will execute pull-ups, pushovers, and will yaw the airplane in wings level flight up to 5° of yaw.

Note: The above test will be conducted during or immediately after the shakedown flights period.

Test: Fuel Tank Pressurization System Tests CVAC Index No. 4.16

Requirements: Fuel tank pressure limits: 4.5 to 8.00 psig relative to ambient air pressure.

Procedure: Record fuel tank pressure and fuel temperature (in tank) during flight.

Periodic readings should be obtained throughout the flight to determine pressure regulation with varying fuel quantity.

The desired data will be obtained concurrently with other tests. Fuel tank pressures and bleed air temps will be automatically recorded throughout flights until adequate data are obtained.

Test: Engine Equipment Operation CVAC Index No. 3.0

Purpose: To demonstrate and evaluate the various engine accessories listed below.

Requirements: The following engine equipment will function according to manufacturer's specifications:

1. Engine Oil Pump
2. Engine Oil Coolers
    a. Air-Oil Cooler
    b. Fuel-Oil cooler
3. Engine Fuel Pump
4. Ignition System
5. Low-Speed Compressore Bleed Valve

Procedure: The above items will be qualitatively evaluated during all engine operation. Any unsatisfactory operations noted will be subjected to further investigation and quantitative data will be obtained if required.

Test: Engine Equipment Operation
Engine Fuel Control
CVAC Index No. 31.0

Purpose: To demonstrate and evaluate the engine fuel control unit.

Requirements: The engine fuel control unit will meter fuel to the engine in the correct manner in accordance with the demand from the power control lever.

Procedure:

1. Determine engine power output vs. fuel flow and power lever angle during static thrust calibration and speed-power runs as an indication of correct fuel control operation.

2. Determine fuel flow vs. RPM during jam accelerations and deceleration for ground, 15,000' and 35,000' operation.

Test: Engine Equipment Operation
Afterburner Operation
CVAC Index No. 3.10

Purpose: To demonstrate and evaluate the engine afterburner and controls.

Requirements: The engine afterburner will operate according to manufacturer's specifications.

Procedure:

1. Determine afterburner thrust and operating characteristics during static thrust calibration.
2. Light afterburner at various RPM's and note power lever angle.
3. Make accelerations and decelerations in AB regime.
4. Determine power lever angle vs. engine thrust in AB regime.
5. Light up afterburner at 15,000' and 35,000' PA. Determine AB thrust, acceleration and deceleration characteristics and power lever angle at the above altitudes.
6. Determine fuel consumption during all afterburner tests.
7. Determine AB eyelid operation characteristics in the tests listed above.

Test: Flutter and Vibration — CVAC Index No. 1.8

Purpose: To demonstrate that the airplane has positive damping of externally induced vibrations throughout the speed range of the airplane.

Procedure:

1. Stabilize the airplane at M = .83 at 35,000 ft. P.A. in level flight. Turn on oscillograph. Turn on vibrators for anti-symmetric excitation. Take decay records at 12 cps, 18 cps, 22 cps, 29 cps, 36 cps, and 46 cps. Repeat with symmetric excitation.

2. Stabilize airplane at M = .95 at 35,000' PA & repeat (1) above.

3. Stabilize airplane at maximum level flight speed at 35,000 ft. P.A. Turn on oscillograph. Turn on vibrators for anti-symmetric excitation. Take decay records at 12 cps, 18 cps, 22 cps, 29 cps, 36 cps and 46 cps. Repeat with symmetric excitation.

(4) Stabilize airplane in dives from 35,000 ft. P.A. in increments of M = .05 above maximum level flight speed. Obtain decay records for symmetric and anti-symmetric excitation at one or more of the frequencies in ascending order as are given in (3) above.

Note: Maximum speed to be attained during any flight will be subject to approval based on the results of flutter and vibration data taken during previous flights.

Test: Airspeed Calibration - Low Speed CVAC Index No. 2.1

Purpose: To determine test nose boom airspeed position error.

Requirements: The airspeed boom will be a test article only. No specification requirements have to be met. The position error shall be reasonable and repeatable calibration data must be obtainable.

Procedure: Calibrate the airspeed system by the pacer method in the following speed ranges:

a.	Clean	150-350 mph V_{io}
b.	Speed brakes extended	150-350 mph V_{io}
c.	Landing gear extended	150-Placard speed for gear extension

The altitudes for these tests are at the pilot's discretion.

Test: Airspeed Calibration - High Speed CVAC Index No. 2.1

Purpose: To determine the test nose boom airspeed position error.

Requirements: The airspeed boom will be a test article only. No specification requirements have to be met. The position error shall be reasonable and repeatable calibration data must be obtainable.

Procedure: The system is to be calibrated by radar plotting during an atmosphere-surveying climb ($M < .85$) and a subsequent high speed dive.

a. Climb at NRT from 15,000' to 37,000' (Do not exceed $M = .85$).
b. Go into a shallow dive from 37,000' to 35,000' with military plus afterburner power. Attain V_{max}.
c. Make descent from 35,000' to 15,000' at maximum allowable Mach number.

Note: The above test must be repeated on at least (2) additional days.

Airspeed calibration data may be obtained from any flight providing the aircraft is radar-tracked during the flight and an atmosphere-surveying climb is made to the maximum altitude attained during the flight.

Test: Preliminary Flight Characteristics Test CVAC Index No. 1.12 (Preliminary Stability and Control)

Purpose: Determine general flight characteristics of the airplane.

Requirements: The airplane shall exhibit safe and reasonable stability and control characteristics. Compliance with all parts of Specification R 1815-B will not be required during these tests since the airplane control system will not be in final configuration at this stage. Compliance with Specification R 1815-B with authorized deviations will be demonstrated during the stability and control tests on pages 24 through 44 .

Procedure: The flight characteristics items listed below will be preliminarily evaluated during the early stages of the flight test program. Evaluation will be principally obtained from qualitiative comments and implemented by recorded data.

The following items will be evaluated during the early flights concurrent with other operations and require no special test operations.

1. Effectiveness of controls.
2. Control forces.
3. Trim control operation (rate and sensitivity).

The following flight characteristic items will require that tests be conducted to determine their values. The desired altitude for the tests is 30,000 ft. with the airplane in the clean configuration.

1. Static Longitudinal Stability

   Determine stick force and stick position stability at two trim speeds in the level flight speed range.

   Trim in level flight at 1.4 V_{S_L}. Vary speed in increments of 20 mph above and below trim speed to maximum practical by use of elevator control only. Maintain altitude within ±2000 ft. Repeat the above at max. allowable level flight speed.

2. Dynamic Longitudinal Stability

   Determine at the trim speeds called out in Item 1 above. Deflect elevator in small increments and record damping.

Test: Preliminary Flight Characteristics Test (Continued) CVAC Index No. 1.12

Procedure: 3. Static Directional Stability

Determine in straight steady sideslips at approximately $1.4V_{S_L}$ and maximum allowable level flight speed, follow procedure outlined in items 3 and 4 under Static Directional Stability, page 34.

4. Dynamic Directional Stability

Determine at the trim speeds called out in Item 1 above. Follow procedure outlined in Items 3, 4 and 5 under Dynamic Directional Stability page 35.

5. Longitudinal Stick Force

Determine stick force per "G" during diving turns and pull-outs.

6. Dynamic Lateral Stability

Determine at the trim speeds called out in Item 1 above, follow procedures outlined in Items 3 and 4 under Dynamic Lateral Stability, page 39.

Note: These tests will be conducted in the "Direct Manual Mode" of flight.

Test: Angle of Attack and Angle of Yaw Pickup Calibration — CVAC Index No. 5.0

Purpose:

(a) Calibration of error of fire control system angle of attack pickup.

(b) Determination of optimum location of pickups.

Requirements: The angle of attack pickups will be located in the position for minimum position error consistent with other design criteria.

It is desired that the calibration error be less than 0.25 degree.

Note: Design specifications for the production instruments are to be furnished at a later date.

Procedure: Special flight operations are not required for these calibrations. The errors will be determined by statistical analysis of photo panel data of pickup readings under various conditions of flight including maximum speed and load factor conditions.

Test: Preliminary Airplane Performance, Climb — CVAC Index No. 2.3

Purpose: To determine service ceiling, best climb speed, and rate of climb.

Requirements: None.

Procedure:

(1) Configurations to be tested:
  (a) Airplane with straight leading edges
  (b) Airplane with cambered leading edges and reflexed wing tips.
(2) Make climbs to service ceiling with military and military plus afterburner thrust settings at the speeds tabulated in schedules A, B, C and D given below:

Fuel consumption data will also be obtained during the above tests.

MILITARY THRUST
Climb-Speed Schedules, Mach No.'s.

Altitude	A	B	C
5000	.62	.65	.68
10000	.66	.69	.72
15000	.71	.73	.75
20000	.76	.78	.80
25000	.81	.83	.85
30000	.83	.85	.87
35000	.85	.87	.89
40000	.85	.87	.89

MILITARY + AFTERBURNER THRUST
Schedule D

M = .86)
.88) Hold Mach. No.
.90) Constant During
.92) Climb

Test: Preliminary Airplane Performance Speed-Power, Fuel Consumption — CVAC Index No. 2.2

Purpose: To determine the speed-power characteristics and fuel consumption of the airplane-engine combination.

Requirements: None.

Procedure:

1. Configurations to be tested:
   a. Airplane with straight leading edges only
2. Conduct at 15,000' and 40,000' PA
3. Stabilize speed in trimmed steady level flight at the speed increments given below:

15,000'	40,000'
M = .3	M = .7
M = .4	M = .8
M = .5	M = .9
M = .6	M = .95
M = .7	M = Max.
M = .8	
M = .90	
M = .95	
M = Max.	

Note: These tests are planned primarily to determine the optimum wing leading edge and tip configuration. Sufficient instrumentation will be installed in the airplane during all earlier flights to obtain airplane performance data while other tests are being conducted.

Test: Cambered Wing L.E. and Reflex Wing Tips — CVAC Index. No. 1.12

Purpose: To determine wing tip air loads during performance tests.

Requirements: Air loads on wing tips should not exceed structural limitations.

Procedure: Wing tip loads will be measured concurrent with performance tests.

Test: Phase II

Purpose: AFFTC Evaluation of the airplane.

Requirements: None

Procedure: The procedure for these tests are at the discretion of the testing agency, therefore no detailed procedure is stipulated in this report.

Tests: Effects of Missile Bay Door Operation on Flight Characteristics   CVAC Index No. 1.7

Purpose: To determine the effects of open missile bay doors on flight characteristics and performance.

Requirements: (1) Sequenced operation or prolonged extension of the missile bay doors should not adversely affect stability of the airplane.

(2) The pressures in the missile bay shall be within acceptable structural limits.

(3) The bay doors shall operate satisfactorily at all speeds.

Procedures: 1. Make stabilized speed runs at approximately 25,000 ft. (P.A.); gross weight = 26,000# (approximately); thrust and drag shall be measured. Shallow dive is permissible if required to obtain speed at M = .5, .6, .7 with missile bay doors open.

2. At the above stabilized speeds and with missile bay doors open, check static longitudinal stability through a small range of elevator deflections.

3. Check static directional stability, wings, level, under conditions stated in (1) above.

After analysis of the data from the above tests, repeat items (1) through (3) at M = .8 to max. allowable, then:

A. Operate missile bay doors in a simulated rocket firing cycle during level flight at 25,000', and gross weight of 26,000 pounds. Conduct at four different speeds between M = .8 and maximum allowable.

B. Open missile bay doors at the speeds given in (A) above at a load factor of approximately .8g and pull up to maximum allowable load factor.

C. Introduce elevon step inputs through the programmer during level flight at M = .8 and maximum allowable or M = .9.

D. Same as (C) except with rudder inputs.

E. Land with missile bay doors open.

Test: Airplane Performance, Determination of Optimum Leading Edges and Wing Tips — CVAC Index No. 2.2

Purpose: To obtain drag polar data.

Requirements: The comparison of performance between the two types of wing leading edges and tips. The following data will be obtained.

1. Drag polars in level flight
2. Drag polars in accelerated flight.

Procedure:

1. Conduct in clean configuration at approximately 40,000' (PA) and 15,000' (PA) for each of the following:
   - (a) Airplane with straight wing leading edges
   - (b) Airplane with cambered wing leading edges and reflexed wing tips
2. Stabilize at the approximate speeds and altitudes tabulated below. Record data.
3. Starting with the test altitude and speed of (2) above, perform a diving turn tightening the turn until maximum allowable positive g has been attained. Record data throughout the maneuver (approximately 10 seconds).
4. Repeat (3) at all speeds and altitudes tabulated below:

Alt. - 15,000'	Alt. - 40,000'
M = .6	M = .7
= .7	= .8
= .8	= .9
= .85	= .95
= .90	= Max.
= .95	
= Max.	

Test: Airplane Performance, Take-Off Distance CVAC Index No. 2.4

Purpose: To determine minimum distance required to become airborne and to clear a 50 ft. obstacle.

Requirements: Estimated take-off distance has been computed but no contractual requirements exist.

Procedure:

1. Determine horizontal distance from start of take-off roll to point where airplane is airborne and to point where airplane is 50 ft. off ground.
2. Make series of high performance take-offs with military plus afterburner power and determine minimum distances defined above.
3. Pilot shall use his own judgment in developing a technique for maximum performance take-offs.
4. Use Air Force photo-theodolite to determine distances.

Note: The above tests will be conducted when the optimum leading edge and wing tip configuration has been established. Preliminary take-off data will be obtained during the early flight testing.

Test: Airplane Performance, Landing Distance CVAC Index No. 2.5

Purpose: To determine the minimum distance required to land the airplane over a 50 ft. obstacle.

Requirements: Estimated landing distance has been computed but no contractual requirements exist.

Procedure:

1. Make a series of landings with and without deploying drogue chute to determine minimum landing distance.
2. Pilot shall use his own judgment in developing a technique for minimum distance landings.
3. Use Air Force photo-theodolite to determine distances.

Note: The above tests will be conducted when the optimum leading edge and wing tip configuration has been established. Preliminary landing distance data will be obtained during the early flight testing.

Test: Engine and Structural Cooling — CVAC Index No. 3.1

Purpose: To determine airplane structure temperature in the vicinity of the power plant.

Requirements: Design criteria require that structural temperatures forward of the firewall do not exceed 250°F. Structural temperatures aft of the firewall must not exceed 350°F.

Procedure: Temperature data will be obtained during performance testing and taxi operations.

Tensile coupons have been installed in the airplane to determine the effect of structural temperatures on structural strength. Three of the coupons have been installed at each of the locations listed below. One coupon should be removed from each station after 10 hours, 50 hours, and 100 hours of flight test.

Location of Coupons:

(1) Frame at Fuselage Station 528.5, BL 101 R.H., WL +28 (24ST80)
(2) Frame at Fuselage Station 499, BL 25 R.H., WL +21.25 (24ST80)
(3) Skin at Fuselage Station 499, BL 10 R.H., WL +29 (24ST81)
(4) Bulkhead at Fuselage Station 552, BL 0, WL +30.5 (Titanium RC-130A)

Test: Dynamic Response — CVAC Index No. 1.7

Purpose: To determine dynamic stability and response of the airplane and control system in manual and automatic modes of flight to programmed control system input signals.

Requirements:

1. The airplane shall exhibit satisfactory response to input signals.
2. Measurements shall be obtained with sufficient accuracy to provide the fire control system manufacturer with design data for automatic flight control.
3. The various parameters for the pitch damper, yaw damper, and turn coordinator will be determined from these tests.

Procedure: The airplane will be trimmed in level flight. A specified signal will be introduced to the control system. The response of the airplane and control system will be recorded during hands-off flight. These signals will be introduced singly or in a programmed series by selection of automatic or manual operation of the programmer. Details of the configuration of test will be specified at a later date.

Test: Static Longitudinal Stability CVAC Index No. 1.1A

Applicable Spec: R1815B Par 4.2.1

Purpose: To demonstrate that the airplane has positive static stick-fixed longitudinal stability for the specified conditions with exception of the noted deviation.

Requirements: Demonstrate that the airplane has positive static stick-position and stick force longitudinal stability at all normal service loadings when flying with constant power settings under conditions given in Table I, page 25 . Except in configuration D, the existence of such stability shall be proved by determination of the stick-fixed neutral points. The neutral point determined for the condition of most aft C.G. location must be at least 5% of the MAC aft of the most aft C.G. location. In configuration D, the stability shall be indicated by stable variations of elevator control force and angle with indicated airspeed.

Deviation: At Mach numbers above 0.85 with the airplane trimmed at M = 0.85, the variation of elevator force and deflection with Mach number may be negative if the force changes involved are not greater than 20 lbs. and occur over a reasonably wide Mach number range.

Procedure:

1. Conduct at altitudes specified in Table I, page 25.
2. Trim airplane at specified speed in configuration to be tested.
3. Stabilize speed in specified increments through speed range by use of elevator control only.
4. Determine δ_e and F_e vs. airspeed or Mach number.

Remarks: Since C.G. position varies during flight, the data should be taken for those loadings which determine the most forward and most aft C.G. position possible during flight, i.e., to obtain C.G. variation necessary to determine the longitudinal neutral point.

TABLE I

[illegible] CONDITIONS FOR STABILITY

Configuration	Symbol	Flaps	Speed Brakes	Power	Trim Speed	Speed Range	Altitude (feet)
Cruise	CR	Up	In	M = .85 in L.F.	M = .85	$1.1V_{S_G}$ ~~[illegible]~~-V_{max} practical (50 mph increments)	12,000 and 30,000
Power On Clean	P	Up	In	NRT	V_{max} NRT	$.75V_H$* ~~[illegible]~~ to V_{max} practical (50 mph increments)	12,000 and 30,000
				As necessary	$V_{R/C}$ to be specified later	~~[illegible]~~ to V_{max} practical (50 mph increments)	
Dive	D	Up	Out	25% NRT	All speeds in the specified speed range	All permissible speeds speeds above V_{S_G}	From 30,000
Glide	G	Up	In	Idle	1.4 V_{S_G}	All permissible speeds above V_{S_G} (50 mph increments)	12,000 and 30,000
						All permissible speeds above V_{S_G} (25 mph increments)	
Power Approach	PA	Down	In	L.F. at 1.2 V_{S_L}	L.F. at 1.2 V_{S_L}	All permissible above V_{S_G} (25 mph increments)	12,000
Land	L	Down	In	Idle	1.4 V_{S_L}	All permissible speeds above V_{S_G} (25 mph increments)	12,000

NOTE: Maintain specified altitudes to ± 2000'.
A plot of stalling speed vs gross weight is presented on page 94.

* - V_H is the maximum level flight speed with military power

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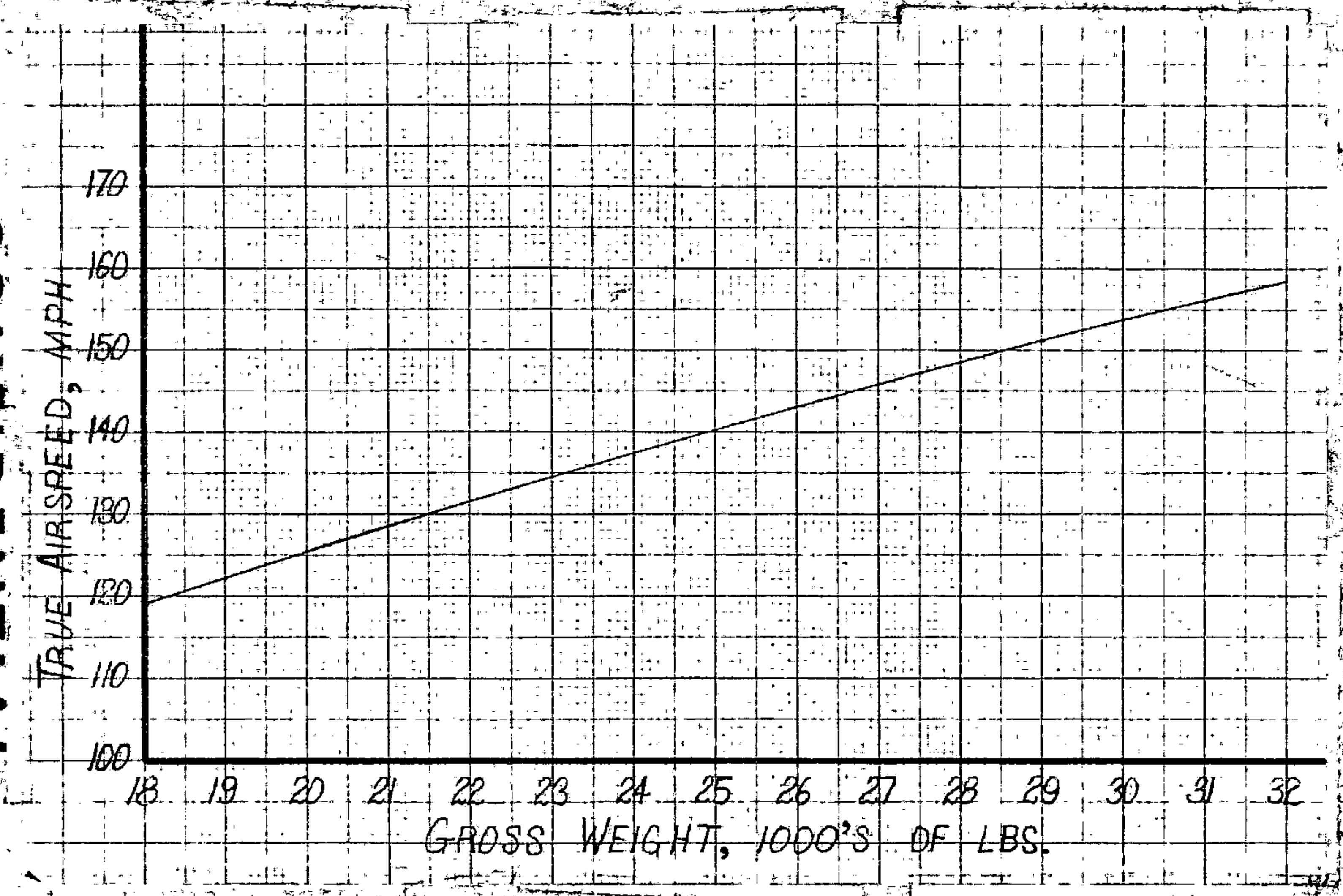

STALLING SPEED VS. GROSS WEIGHT, SEA LEVEL

BASED ON C_L = 0.75

Note: Maximum allowable angle of attack is +20°.
This angle will be exceeded for a C_L = 0.75 with C.G. forward of 30% MAC

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Test: Static Longitudinal Stability CVAC Index No. 1.1A

Purpose: To determine the effect of C.G. shift with airplane attitude on static longitudinal stability.

Requirements: None

Remarks: Center of gravity shift due to attitude is maximum when No. 2 fuel tank is approximately 1/2 full (No. 3 full) Fuel quantity of 3114 lbs.

Procedure:

1. Conduct at 12,000 ft.
2. Trim airplane in steady level flight with approximately 60% NRT
3. Vary airplane attitude to 5° nose up and 5° nose down in stabilized 1° increments. Use increasing or decreasing power to maintain trim speed.
4. Determine δ_e vs. C.G. Position.

Test: Dynamic Longitudinal Stability CVAC Index No. 1.1C
Applicable Spec. R1815-B Par. 4.1

Purpose: To demonstrate that elevator induced longitudinal oscillations are damped out.

Requirements: Short period dynamic oscillations produced by sudden elevator deflection shall be damped out. Damping oscillations must be stabilized but number of cycles required to dampen to be approved by WADC (Spec. Deviation).

Procedure:

1. Conduct at altitudes specified in Table I, page 25.
2. Trim airplane in steady level flight in configurations and trim speeds specified in Table I, page 25.
3. Deflect elevator abruptly and release. Record data until short period oscillations cease. Both pull-ups and push-overs should be accomplished. The Programmer will be used to supply specified elevon deflections when practical.
4. Repeat at high and low speeds of each configuration.
5. The above tests are to be conducted with manual and direct manual mode control system configurations.

Test: Longitudinal Control Force CVAC Index No. 1.1D
Applicable Spec. R1815-B Par. 4.4.1, 4.4.2, 4.4.4

Purpose: To determine elevator control force gradient (stick force per "g").

Requirements: Show that elevator control force gradient is within the specified limits: 12.9 lbs/g max.; 4.85 lbs/g min.

In steady turning flight and in pull-outs, increase in pull force shall be required to produce increases in normal acceleration, throughout the range of permissible accelerations.

Elevator control force gradient for maximum normal acceleration in sudden pull-ups from trimmed straight flight shall never be less than the gradient in steady turning flight under the same conditions of gross weight, configuration, altitude, C.G. and airspeed.

Procedure:

1. Conduct at altitudes specified in Table I, page 25.
2. Trim airplane in configurations P and D listed in Table I, page 25.
3. Climb to 2000 ft. above test altitude.
4. Make steady diving turns in which the steepness of the turn is used to control the acceleration and the airspeed is held constant at trim value by losing altitude.
5. Develop maximum positive load factor in turns.
6. Make sudden pull-ups from straight trimmed flight during which maximum positive load factor is developed for configuration P only. Conduct at most rearward C.G. and at trim speed of M = 0.7, speed for NRT, and at maximum practical.

Test: Longitudinal Control Force CVAC Index No. 1.1D
Applicable Spec. R 1815-B Par. 4.5.4

Purpose: To determine elevator control force during high speed dive.

Requirements: With airplane trimmed in steady level flight with NRT, configuration P, it shall be possible to hold the airplane in dives up to the maximum practical speed with an elevator push force not to exceed 56 pounds. If a pull force is required to maintain a steady dive, this pull force shall not exceed 35 lbs. and the additional force required to execute a 3g pull-out shall not exceed 75 lbs.

Test: Longitudinal Control Force (Continued) CVAC Index No. 1.1D
Applicable Spec. R 1815-B Para. 4.5.4

Procedure:

1. Trim airplane in steady level flight at service ceiling.
2. Dive airplane and stabilize speed at approximately 50 mph increments during dive until maximum practical speed is reached. The maximum practical speed is expected to be less than maximum permissible speed designed for the J-67 engine installation.
3. Execute a 3g pull-out with recovery at not less than 15,000 ft.

Test: Longitudinal Control Effectiveness CVAC Index No. 1.1E
Applicable Spec. R 1815-B Para. 4.3.1

Purpose: To prove that sufficient elevator control exists to maintain steady flight for the given conditions.

Requirements: With the airplane trimmed at the speeds specified in Table I, page 25, it shall be possible in each configuration to maintain steady flight over the specified speed range.

Procedure: This requirement will be demonstrated in tests of static longitudinal stability.

Test: Longitudinal Control Effectiveness CVAC Index No. 1.1E
Applicable Spec R-1815-B Para. 4.3.2.

Purpose: To prove that sufficient elevator control exists to develop maximum positive or negative load factor at trim speed.

Requirements: Develop maximum positive or negative limit load factor at trim speed by means of elevator control only for each configuration listed in Table I, page 25.

Above requirement is subject to limitations imposed by Dynamics Group in accordance with data obtained from previous tests.

[illegible] Applicable, except as limited by angle of attack corresponding to a $C_L = 0.75$ at stall speed, maximum available elevator deflection or maximum limit load factor.

Angle of Attack at $C_L = 0.75$, $\alpha = 20^\circ$, (see page 26)
Max elevator deflection = -25°
$+10^\circ$

Test: Longitudinal Control Effectiveness (Con't) CVAC Index No. 1.1E
Applicable Spec. R-1815-B Para. 4.3.2

Procedure:

1. Data to comply with above positive load factor requirements will be obtained from tests for elevator control force gradient.
2. Develop maximum negative limit load factor at the specified trim speeds by making push-overs from straight, steady trimmed flight.

Note: Tests on this airplane will be of a preliminary nature only as demonstration of compliance with this specification will be obtained during flight loads testing on a later airplane.

Test: Longitudinal Control Effectiveness CVAC Index No. 1.1E
Applicable Spec. R-1815-B Para. 4.3.3

Purpose: To demonstrate that sufficient elevator control exists to hold airplane in close proximity to the ground at the given speeds.

Requirements:

1. With C.G. in most forward position and airplane trimmed at 1.4 V_{SL} or at maximum allowable ground angle, whichever attitude corresponds to the higher speed.
2. Longitudinal stick force shall not exceed 35 lbs. during (1).

Note: Maximum allowable ground angle = 20°

[illegible] As outlined in Requirements above.

Test: Longitudinal Control Effectiveness — CVAC Index No. 1.1E
Applicable Spec. R 1815-B Para. 4.3.5
Longitudinal Control Forces — CVAC Index No. 1.1F
Applicable Spec. R1815-B Para. 4.4.7 & 4.4.8

Purpose: To prove that sufficient elevator control exists to attain take-off attitude at 90% V_{S_L} and to show that elevator stick force during take-off does not exceed the given limits.

Requirements: Airplane shall be able to assume take-off attitude at 90% V_{S_L} for all normal service loadings. This includes the G.W. and C.G. condition which produces the largest static nose down moment about the main wheels. With airplane trimmed for 1.25 V_{S_L}, the nose wheel lift off at 90% V_{S_L} shall not require a stick pull force greater than 35 lbs.

With airplane trimmed for 1.25 V_{S_L} with C.G. in most aft position, the transition from take-off speed to 1.25 V_{S_L} shall not require a stick-push force greater than 10 lbs.

Procedure: See taxi tests.

1. Set elevator trim control for trimmed flight at 1.25 V_{S_L} and record data during take-off with maximum G.W. for most aft C.G. Position.

Note: Trim position for 1.25 V_{S_L} can be set with pilot's test cockpit instrumentation.

Test: Longitudinal Trim Changes — CVAC Index No. 1.1F
Applicable Spec. R 1815-B Para. 4.6.1, 4.6.2 and 4.6.4.

Purpose: To determine longitudinal trim change due to effects of power and configuration.

Requirements: With the airplane trimmed for zero elevator control force at any given speed and using any combination of engine thrust, gear position or speed brake setting, it shall be possible to maintain the given speed without exerting push or pull forces on the stick greater than 35 lbs. when the power, gear or speed brake settings are varied in any manner.

Test: Longitudinal Trim Changes (Continued) CVAC Index No. 1.1F
Applicable Spec. R 1815-B Para. 4.6.1, 4.6.2 & 4.6.4

Requirements: With airplane trimmed in steady level flight with maximum power, it shall be possible to maintain the trim speed for one second following full extension of speed brakes with an elevator push force not to exceed 20 lbs. or a pull force not to exceed 16 lbs.

Procedure:

1. Conduct at 12,000 ft.
2. Trim airplane in steady flight at trim speeds given below and perform operation listed under "variable."

	Trim Speed	Gear	Speed Brakes	Power	Variable
1.	1.4 V_{SG}	Up	In	Idle power	Gear Down
2.	1.4 V_{SG}	Up	In	50% NRT	Gear Down
3.	1.4 V_{SL}	Down	In	50% NRT	Idle Power
4.	1.2 V_{SL}	Down	In	Max. Power	Gear Up
5.	V_{max} NRT	Up	In	Level Flt.	Max. Power
6.	1.2 V_{SL}	Down	In	Level Flt.	Max. Power
7.	V_{max} NRT	Up	In	Level Flt.	Idle Power
8.	1.4 V_{SG}	Up	Out	Level Flt.	Gear Down
9.	V_{max} power	Up	In	Level Flt.	Speed Brakes Out

Note: V_S based on stalling speed at $C_L = 0.75$. It is desirable that the trim change in item 3 be such that a pull force is required to maintain the given trim speed.

Test: Longitudinal Trim Change CVAC Index No. 1.1F
Applicable Spec. R 1815-B Para. 4.6.3

Purpose: Determine longitudinal trim change in sideslips.

Requirements: The stick force in pounds required to maintain trim speed in sideslips obtainable with 50 lbs. rudder pedal force or up to maximum attainable sideslip, shall not exceed numerically the value of the stick force gradient in accelerated maneuvers or 10 lbs., whichever is lower. The preferred variation of stick force in sideslip is for increased pull force on stick with increased sideslip.

Procedure: No special test needed. Measure and record stick force during tests for static directional stability.

Test: High Speed Requirements
Applicable Spec. R 1815-B Para. 4.5.1 CVAC Index No. 1.7

Purpose: To demonstrate that no dangerous effects occur when the airplane is flown through the transonic speed range.

Requirements: The following adverse or dangerous effects due to compressibility shall not occur:

1. Severe buffeting or shaking.
2. Excessive longitudinal trim changes or uncontrollable recovery moments.
3. Erratic or violent behavior of airplane or controls.

Procedure: Demonstration of compliance for high speed requirements will be accomplished concurrently with flutter tests.

Test: Static Directional Stability
Applicable Spec. R 1815-B Para. 5.2.1 CVAC Index No. 1.2A

Purpose: To prove that the airplane has positive rudder-fixed static directional stability.

Requirements: The rudder-fixed static stability shall be such that rudder deflection from its trim position at zero bank produces sideslip in the correct direction when performing a steady sideslip; that is, positive (left) rudder deflection from trim shall produce positive (right) sideslip and vice versa. This requirement applies to all configurations and speed ranges listed in Table I, page 25, and in configuration PA with maximum power at all permissible speeds above $V_{S_{PA}}$. For angles of sideslip between plus and minus 15 degrees, the angle of steady sideslip shall be approximately proportional to the rudder deflection from trim. For angles of sideslip greater than 15 degrees, increases in rudder deflection shall produce increases in the angle of steady sideslip up to full rudder deflection.

Procedure:

1. Conduct at altitudes specified in Table I, page 25.
2. Trim airplane for wingslevel flight in all of the configurations given in Table I, page 25, at the given trim speeds.

Test: Static Directional Stability (Continued) CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 5.2.1

Procedure:

3. Perform a straight steady sideslip, banking the airplane as necessary to maintain straight flight, in increments of three degrees of rudder deflection until limit side load factor is reached.
4. Stabilize at each rudder setting and record data.

Remarks: Direct manual mode of control.

Test: Static Directional Stability CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 5.2.2

Purpose: To determine adverse yaw produced during roll-outs from banked steady turns.

Requirements: Rudder-fixed static directional stability shall be sufficient to restrict the angle of sideslip due to sudden application of aileron deflection to not more than one degree of sideslip per 5% of full aileron deflection, when rolling out of a trimmed steady 45° banked turn at 1.4 V_{SCR} in configuration G with power for straight and level flight at 1.4 V_{SCR} and in configuration PA at 1.4 V_{SCR}.

Procedure:

1. Conduct in configuration G and in configuration PA at 12,000 ft.
2. Trim airplane in configuration to be tested at 1.4 V_{SCR} in a steady 45° banked turn.
3. Apply aileron deflections abruptly to roll airplane into opposite bank, holding rudder fixed in trim position, continue roll to 45° bank on opposite side.
4. Resume a steady turn in trimmed flight, then reverse aileron displacement back to original turn again.

Note: The limiting factor in conducting these tests is the side load.

Test: Dynamic Directional Stability CVAC Index No. 1.2B
Applicable Spec. R 1815-B Para. 5.1.1

Purpose: To determine the damping of the yaw induced lateral-directional oscillation.

Requirements: Damping shall be positive and shall be such that the time to damp to one-half amplitude and the period shall fall within the satisfactory area of the chart given in Figure 1, page 36. There shall be no tendency for undamped small amplitude oscillations to persist.

Deviation: Artificial damping devices may be used to meet the above requirements.

Procedure:

1. Conduct tests at altitudes specified in Table I, page 25.
2. Airplane to be tested in all configurations and trim speeds given in Table 1, page 25.
3. Develop wings level yaw with rudder, applying aileron as necessary to keep wings level.
4. Release rudder abruptly, return aileron to neutral and hold fixed.
5. Record data until lateral-directional oscillations cease.
6. Use manual mode of control.

Test: Spiral Stability CVAC Index No. 1.2C
Applicable Spec. R 1815-B Para 5.1.2

Purpose: To demonstrate the airplane has satisfactory spiral stability characteristics.

Requirements: Spiral stability is not required but the allowable rate of divergence of the spiral mode with controls free shall not be so great that the spiral motion doubles amplitude in less than 4 seconds. This requirement applys to the conditions of Table I, page 25.

Procedure:

1. Conduct tests in configuration and at the altitudes and trim speeds specified in Table I, page 25.
2. Trim airplane carefully in straight steady level flight for the conditions specified above. Cause a small lateral displacement by slowly applying a small aileron displacement (approximately 2-3 degrees) and record data during subsequent airplane behavior. Perform maneuver with all controls free and also with all controls returned to neutral and held fixed following initial displacement.

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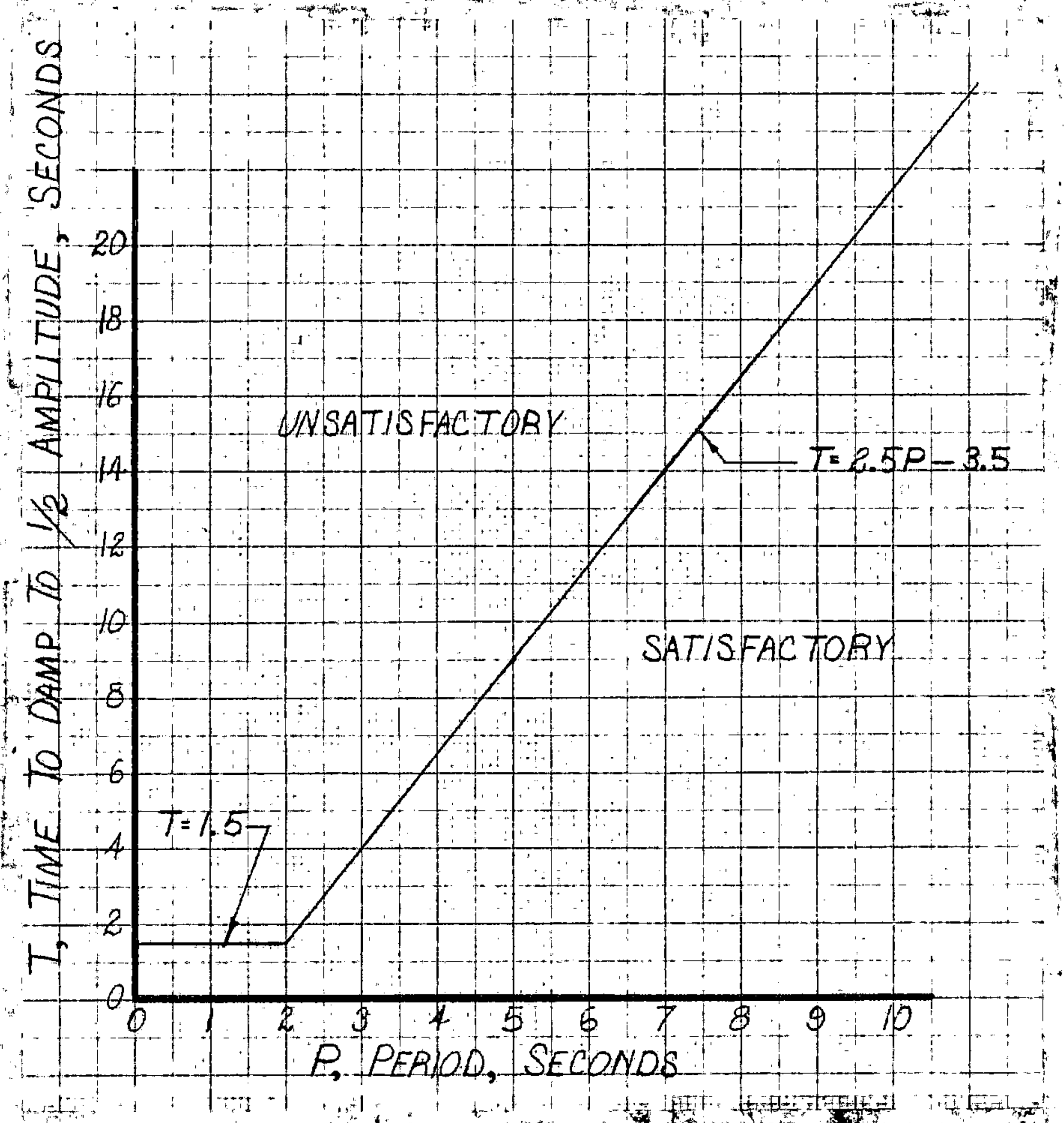

FIGURE 1

REQUIRED DAMPING OF THE LATERAL-DIRECTIONAL OSCILLATION

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Test: Rudder Control Effectiveness CVAC Index No. 1.2B
Applicable Spec. R 1815-B Para. 5.3.1

Purpose: To prove that rudder effectiveness is sufficient to maintain straight level flight in all configurations.

Requirements: Maintain straight level flight for all configurations and trim speeds given in Table I, page and also in configuration WO trimmed at 1.2 V_{S_L}. Maximum rudder pedal forces not to exceed 180 lbs. except for configuration WO where they shall not exceed 100 lbs.

Procedure: No special test required. This will be demonstrated in other tests. Refer to static longitudinal stability.

Note: Configuration WO
Maximum Available Thrust-Gear Down-Trim Speed 1.2 V_{S_L}.

Test: Rudder Control Effectiveness CVAC Index No. 1.2H
Applicable Spec. R 1815-B Para. 5.3.6

Purpose: To show the rudder control is sufficient to overcome adverse yaw due to aileron action.

Requirements: Rudder control shall be sufficient to overcome adverse yaw when airplane is rolled abruptly out of steady 45° banked turns in configuration G with power for straight and level flight at 1.4 $V_{S_{CR}}$ and in configuration at PA at 1.4 $V_{S_{CR}}$.

Procedure:

1. Conduct at 12,000 ft.
2. Trim airplane in configuration to be tested in a steady 45° banked turn at 1.4 $V_{S_{CR}}$.
3. Apply aileron deflection abruptly to roll airplane into opposite bank using necessary rudder to prevent skidding. Continue roll to 45° bank on opposite side or until maximum allowable side load factor is reached.
4. Resume a steady turn in trimmed flight, then reverse aileron displacement back to original turn.

Test: Directional Trim Control Effectiveness — CVAC Index No. 1.2F
Applicable Spec. R 1815-B Para. 5.6.3

Purpose: To demonstrate that directional trim is sufficient to reduce rudder pedal force to zero for the given conditions.

Requirements: The directional trimming device shall be capable of reducing the rudder pedal forces to zero and/or trim for straight, zero yaw, wings level flight in the following configuration and speed ranges:

Configuration	Speed Range
G	$1.2V_{S_G}$ to V_{max} practical
P	$1.2V_{S_C}$ to V_{max} practical
PA	$1.2V_{S_L}$ to 310 MPH

Procedure: No special test is needed. The above requirements will be demonstrated during tests for static longitudinal stability.

Test: Static Lateral Stability — CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 6.2.1

Purpose: To demonstrate that the airplane has positive dihedral effects for the conditions specified.

Requirements: Right stick force and deflection is required to depress right wing when airplane is in right steady sideslip with the airplane trimmed for steady unyawed flight and vice versa.

This is required for all configurations and trim speeds given in Table I, page 25, and for configuration WO.

Procedure: No special tests are required. Data obtained from tests for static directional stability will demonstrate compliance with the requirements listed above.

Test: Dynamic Lateral Stability — CVAC Index No. 12.B
Applicable Spec. R 1815-B Para. 6.1.3

Purpose: To determine the damping of the roll induced lateral-directional oscillations.

Requirements: Same as for dynamic directional stability.

Procedure:

1. Conduct at altitudes specified in Table I, page 25.
2. Airplane to be tested in all configurations and at all trim speeds given in Table I, page 25.
3. Abruptly deflect ailerons by sharp application of approximately one-eighth to one-quarter full lateral displacement of stick and immediately release control.
4. Record data until lateral-directional oscillations cease.
5. Use manual mode of control.

Test: Dihedral Effect — CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para 6.2.3

Purpose: To determine the aileron deflection required to overcome the dihedral effect in steady sideslip.

Requirements: No more than 90% of full aileron deflection shall be required to counteract the rolling moment produced by full available rudder pedal deflection in steady sideslips.

Procedure: No special testing is needed. Data will be obtained during tests for static directional stability.

Test: Dihedral Effect — CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 6.2.5

Purpose: To show that side force due to sideslip is in the right direction.

Requirements: Right skidding turns shall accompany right rudder deflection and vice versa when the wings are held level and the rudder is deflected from the position required for straight flight.

Procedure: No special tests are needed. During steady sideslips, show that an increase in right bank accompanies an increase in right sideslip.

Test: Aileron Control Effectiveness — CVAC Index No. 1.2D
Applicable Spec. R 1815-B Para. 6.3.4

Purpose: To demonstrate that the ailerons are sufficiently powerful to produce peak roll rates in a rudder-fixed roll.

Requirements: The ailerons shall be sufficiently powerful to produce the following peak roll rates with the rudder held fixed in the position required for steady flight prior to the roll.

Config.	Gear	Power	Speed Range	Min. Acceptable Peak Roll Rate
P	Up	L.F.	$1.1V_{S_G}$ to V_{max} NRT	$\frac{Pb}{2V} = .090$
L	Down	Idle	$1.1V_{S_L}$ to $1.5V_{S_L}$	$\frac{Pb}{2V} = .07$
L	Down	Idle	$1.1V_{S_L}$	$Pb = 10$ ft/sec

Procedure:

1. Conduct at 12,000' PA.

2. Trim airplane at the required speed for the configuration to be tested in 45° banked turn. Speed increments – configuration P, 50 MPH increments, configuration L, 10 MPH increments.

3. After turn is stabilized, hold rudder fixed and apply full opposite aileron abruptly and hold until airplane is rolled into an opposite 45° bank.

4. Repeat with roll in opposite direction.

Deviation: Peak roll rates at V_{max} NRT to be approved by WADC.

Test: Aileron Control Effectiveness CVAC Index No. 1.2D
Applicable Spec. R 1815-B Para. 6.3.1 and 6.3.3.

Purpose: To show that rolling velocity produced by abrupt aileron action varies smoothly with aileron deflection and is in the correct direction. (Rudder fixed)

Requirements: The variation of angle of bank with time immediately following an abrupt control deflection shall always be in the correct direction.

The maximum rolling velocity obtained by abrupt aileron deflection, with rudder held fixed in the trim position, shall vary smoothly with the aileron deflection from the trim position and shall be approximately proportional to the aileron deflection from the trim position.

Procedure: No special test needed, data obtained from tests for aileron control effectiveness will demonstrate above requirements.

Test: Aileron Control Forces CVAC Index No. 1.2E
Applicable Spec. R 1815-B Para. 6.4.1.

Purpose: To determine aileron control forces during accelerated maneuvers.

Requirements: Aileron control force shall vary smoothly with stick deflection during rolling maneuvers and steady sideslips. Aileron control shall be free from shake and snatching tendency at any speed for any permissible acceleration.

Procedure: No special test needed, aileron control force data obtained from tests for static directional stability and aileron control effectiveness will demonstrate above requirements.

Test: Aileron Control Effectiveness — CVAC Index No. 1.2D
Applicable Spec. R 1815-B Para. 6.3.2

Purpose: To determine the magnitude and time lag of the peak rolling acceleration.

Requirements: The peak value of rolling acceleration shall not occur later than 3.75 $\bar{c}/V_t$ seconds or 0.1 seconds whichever is greater after full aileron deflection is reached in a rudder-locked roll in configuration G with level flight power at 1.2 V_{SG}. ($\bar{c}$ is the wing mean aerodynamic chord in feet and V_t is the true airspeed in feet per second.)

Procedure: No special test needed; data obtained during tests for aileron control effectiveness will demonstrate above requirements.

Test: Lateral Trim Control Effectiveness — CVAC Index No. 1.2F
Applicable Spec. R 1815-B Para. 6.6.2 and 6.6.3

Purpose: To determine lateral trimming effectiveness.

Requirements: The aileron trim control shall maintain a given setting indefinitely unless changed intentionally.

The lateral trim control shall be capable of reducing the aileron control force to zero for the same conditions given for directional trim control effectiveness, page 38.

Procedure: No special test needed. Trimming effectiveness for all speeds with wings level will be determined during tests for static longitudinal stability.

Test: Stalling Characteristics (Continued) CVAC Index No. 1.4
Applicable Spec. R 1815-B Para. 7.1.1, 7.1.2, 7.1.3, 7.3.1

Procedure:

2. Gradually reduce airspeed from steady flight at a speed approximately 20 MPH above the expected stall speed by pulling back on stick until maximum elevator deflection is reached. The rate of decreasing speed should not exceed 1 MPH per second.
3. Trim airplane as for (1) above and perform coordinated turns in which the maximum positive load factor ($n = 5.33$) is reached.
4. Perform in direct manual flight mode.

Note: V_S used for trim will be based on a stall speed at $C_L = 0.75$ and not on the stall speeds as defined in the requirements above.

Test: Power Control System Operation CVAC Index No. 4.6
Applicable Spec. R 1815-B, Para. 8.4.1

Purpose: To demonstrate satisfactory operation of the control system when either the primary or secondary hydraulic system fails.

Requirements: In trimmed level flight in configuration P at 10,000 ft. at all speeds between 1.4 V_{SG} and V_{max}, abrupt failure of either the primary or secondary hydraulic systems shall not result in out of trim conditions exceeding the following limitations:

Elevator: ±1/2g acceleration - controls free
Rudder: maintain zero sideslip with no more than 100 pounds pedal force.
Ailerons: rate of roll not to exceed 5°/sec. with controls free.

Procedure: Compliance with the above requirements will be demonstrated during ground test and subsequent flight test.

Test: Stalling Characteristics CVAC Index No. 1.4
Applicable Spec. R 1815-B Para. 7.1.1, 7.1.2, 7.1.3 and 7.3.1.

Purpose: To demonstrate that the airplane exhibits no abrupt changes in flight characteristics in the region of maximum lift coefficient.

Requirements:

1. For purposes of stalling characteristics demonstration only, the stall shall be considered to be identified by full up elevator deflection.
2. If it can be shown that the aircraft exhibits no abrupt changes in flight characteristics in the region of maximum lift coefficient, the complete stall shall be coincident with the minimum speed in straight flight or the maximum acceleration in turning flight.
3. The airplane shall exhibit no excessive downward pitch, excessive roll, excessive loss of altitude, or excessive delay in recovery from the stalled flight condition as defined above.
4. The stalling requirements shall govern for both straight and turning flight for all service loading conditions in the following configurations:

Cruise	CR
Glide	G
Land	L
Power Approach	PA

Procedure:

1. Trim airplane in straight, steady, level flight in the following configurations:

Configuration	Trim Speed	Power Setting	Altitude
CR	M = .85	Power for level flight at M = .85	12,000' & 30,000' PA
G	$1.4 V_{S_G}$	Idle	20,000' PA
L	$1.4 V_{S_L}$	Idle	Minimum Safe Altitude
PA	$1.4 V_{S_L}$	Power for level flight at $1.4 V_{S_L}$	Minimum Safe Altitude
[illegible]	$1.4 V_{S_{CR}}$	Power for level flight at $1.4 V_{S_L}$	Speed Brakes Extended

Test: Pilot Assist Development CVAC Index No. 4.6

Purpose: Development of the Pilot Assist System

Requirements: The pilot assist equipment shall operate satisfactorily over the Mach number range from M = .7 to M = .95 and at altitudes ranging from S.L. to 50,000 ft. The pilot assist system shall be capable of maintaining constant attitudes of pitch and bank. The pilot assist shall have smooth trimming and adjustment characteristics. The system shall have acceptable transients in change from pilot assist to manual mode of flight and vice versa.

Procedure: Utilization of a programmer for introducing input signals to the controls will be made. Detailed test procedures have not been developed at this time.

Test: Engine Operation (Air Starts) CVAC Index No. 3.8

Purpose: To determine altitude and speed ranges in whichair starts can be accomplished.

Requirements: It shall be possible to accomplish air starts at all altitudes up to 35,000 ft. P.A.

Procedure: Shut down engine and make an air start at approximately 35,000 ft. P.A. Record engine data from time of stabilized flight through shut down and up to power after air start. This start shall be accomplished at approximately the minimum engine windmilling speed specified for air starts by the engine manufacturer ($N_2 = 12\%$). Maximum engine speed for air starts, $N_2 = 30\%$. For this test, follow the diagram on page 46 for altitude and speed range recommended by P. & W.

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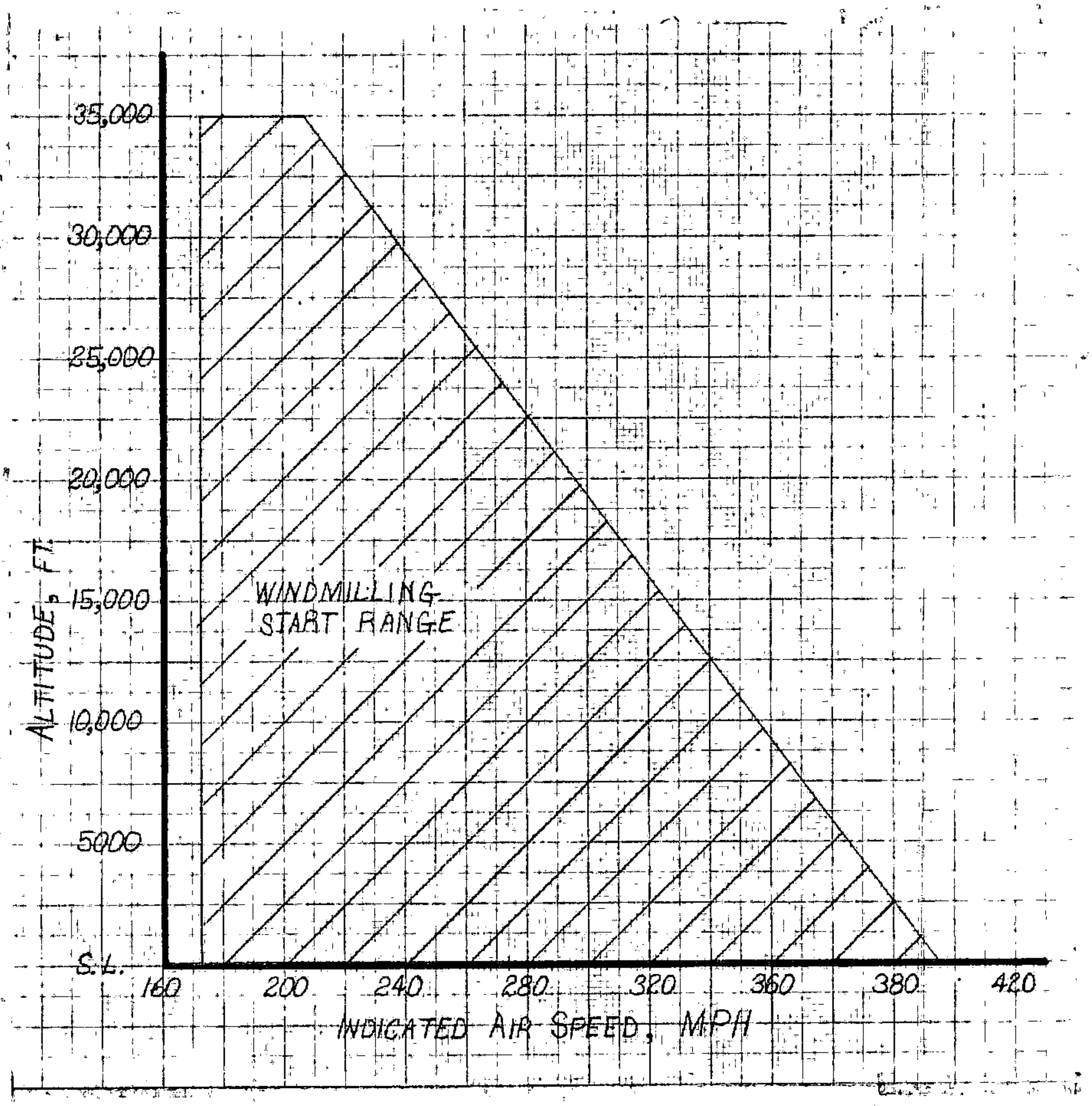

FLIGHT RELIGHT CHART
J57-P-11 Engine
STANDARD ATMOSPHERIC CONDITIONS

Taken from Pratt & Whitney PWA OI 130 Specific Operating Instructions, J-57-P-7 and -11 Engines.

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Test: Drogue Chute Tests CVAC Index No. 4.2

Purpose: Determine drogue chute effectiveness and operational technique.

Requirements:

1. The drogue chute shall deploy smoothly and immediately without fouling.
2. The drogue chute shall release immediately upon actuation of the release.
3. The drogue chute shall be useable in cross winds up to 30 mph.

Procedure:

1. Deploy the drogue chute while taxiing in level attitude and take-off speed. Record data on the photo panel and with a theodolite.
2. Deploy the drogue chute with the nose wheel off the ground with angles of attack of 10°, 15°, and /or maximum practical. Conduct tests at 0.9 V_{S_L} up to approximately 160 mph.
3. Repeat (1) and (2) with a cross wind component not greater than 30 mph.
4. Jettison the drogue chute at approximately 30, 90, 120 and 160 mph V_i.

Note: The tests shown above will be the final tests. Deployment speeds, release speeds and cross winds for initial pilot familiarization will be at the pilot's discretion.

Deployment and jettisoning of the drogue chute will be done with idling thrust.

Test: Equipment Operation (Emergency Fuel Control System)

CVAC Index No. 3.3

Purpose: To determine that the emergency fuel control system operates satisfactorily.

Requirements: The emergency fuel control system shall provide smooth transition from normal to emergency system operation without excessive surging or tailpipe temperatures and will also provide smooth regulation at all speeds and altitudes.

Procedure: Change from normal to emergency fuel control system according to the engine manufacturer's recommended operating procedures and vary power between idle and afterburner regimes. Accomplish at both 5000 and 35,000 feet approximate pressure altitudes. Tests will be conducted along with performance testing.

Test: Equipment Operation - Emergency Ram Air Hydraulic Power Unit CVAC Index No. 4.1

Purpose: Determine effectiveness of the emergency ram air hydraulic power unit.

Requirements: The emergency hydraulic pump must supply 1000 psi at 6.3 gals./min. at 125 knots indicated airspeed to the primary hydraulic system.

1. Extend the emergency hydraulic pump at approximately 160 mph during a high speed taxi run.
2. Extend the emergency hydraulic pump in level flight at maximum level flight speed at 35,000 ft. P.A.
3. Make a normal descent and land directly after extending pump.

Note: Pump flow and pressure data can be obtained by measurement of pump outlet pressure.

Test:	Flight Control System Operation	CVAC Index No. 4.6

Purpose: To evaluate the components of the control system.

Requirements: The components of the control system as listed below shall operate in an acceptable manner under all flight conditions.

Procedure: Data for the following will be qualitative and quantitative and will be obtained during most flight operations. During the shakedown period the controls will be operated in the direct manual mode, therefore additional data will have to be obtained at a later date when operating the controls in the manual mode.

Rudder Pedals (Directional)

1. Travel
2. Comfort
3. Relation to surface travel.

Rudder Pedal Adjustment (Directional)

1. Comfort

Rudder Pedal Force Sensing Elements (Directional)

1. Force vs. Surface Deflection
2. Summing and Differentiation
3. Environment Effects
4. High Speed Limitations of Rudder Deflection

Stretch Spring (Directional)

1. Stretch vs. Pedal Force vs. Rudder Position
2. Environment

Trim Motors (All)

1. Rate
2. Sensitivity

Manual Mode Trim

1. Rate
2. Sensitivity
3. Stability

Test: Flight Control System Operation (Con't.) CVAC Index No. 4.6-1

"Q" Pot (Directional, Longitudinal)

1. Pressure vs. airspeed and/or regulated press.
2. Effect of Attitude, Yaw, Load Factor
3. Control Forces vs. pressures and position from trim.

"Q" Pot Pressure Regulation (Directional, Longitudinal)

1. Pressure sensed vs. airspeed
2. Regulated Press vs. Sensed Pressure
3. Effect of Attitude, Yaw, Load Factor

Valve Follow up Lock out (All)

1. Rate of operation
2. Breakout forces

Servo Valve Actuator (All)

1. Rate of operation
2. Lags
3. Stability
4. Lockout operation

Servo Valve Operation (All)

1. Valve deflection vs. surface travel rate
2. Stability

Surfaces (All)

1. Lag behind servo valve operation

Control Stick (Longitudinal, Lateral)

1. Comfort and room
2. Position relative to surface deflections
3. Play

Aileron Feel Spring (Lateral)

1. Force vs. deflection vs. airspeed.

Friction (All)

1. Effect of control force sensitivity
2. Effects of Environment

MEASUREMENT GROUPS

1. General
2. Power Plant Operation
3. Airplane Performance
4. Flutter Analysis
5. Longitudinal Dynamic Response
6. Lateral Dynamic Response
7. Directional Dynamic Response
8. Control System Operation
9. Damping System
10. Pilot Assist System
11. Engine Cooling
12. Structural Cooling
13. Cooling - Hydraulic & Control Systems
14. Cooling - Engine Accessories
15. Cooling - High Pressure Pneumatic System
16. ATM Components

YF-102 No. 52-7995

Flight Test Program

** MEASUREMENT GROUPS

Group No. 1
Type Test: General

	Measurement	Recording	Range
1.	Outside Air Temperature	Photo	-70°C +170°C
2.	Pressure Altitude	Photo	0 - 60,000 ft.
3.	Airspeed	Photo	50 - 700 MPH
4.	Pitch Attitude	Photo	±40°
5.	Roll Attitude	Photo	±60°
6.	Directional Heading	Osc.	±180°
7.	Angle of Attack, Forward Nose Boom	Photo	-5° +25°
8.	Angle of Yaw, Nose Boom	Photo	±15°
9.	Time of Day	Photo	
10.	Normal Acceleration - Photo Panel	Photo	-5g +12g
11.	Turbine Discharge Temp.	Photo	0 - 1830°F
12.	Turbine Discharge Total Press.	Photo	0 - 200" Hg
13.	Engine RPM, N_2	Photo	0 - 11,000 RPM
14.	Nozzle Position Indicator	Photo	Open - Closed
*15.	Fuel Total, Left and Right Wing	Photo	0 - 600 gal.
16.	Data Synchronizer	Photo	0 - 99999

* Not installed and operable for initial flight testing.

** Measurement Groups include only measurements recorded on the photo panel oscillographs, or telemetering system. The pilot's panel instrumentation is not included in these groups.

Group No. 2
Type Test: Power Plant Operation

Measurement	Recording	Range
1. Air Inlet to Engine, Temp.	Photo	-96°F +300°F
2. Comp. Discharge, Total Temp.	Photo	0 - 900°F
3. Inter Comp. Temp.	Photo	0 - 400°F
4. Turbine Discharge Temp.	Photo	0 - 1830°F
* 5. Air Inlet to Engine, Total Press.	Photo	0 - 200" Hg
6. Inter Comp. Total Press.	Photo	0 - 200" Hg
7. Comp. Discharge, Total Press.	Photo	0 - 400" Hg
* 8. Turbine Inlet Total Press.	Photo	0 - 400" Hg
9. Turbine Discharge Total Press.	Photo	0 - 200" Hg
10. Afterburner Exit Total Press.	Photo	0 - 200" Hg
11. Engine Oil Press.	Photo	0 - 70 PSI
12. Power Lever Angle	Photo	0 - 62.5°
13. Fuel Flow, L/Wing, R/Wing	Photo	0 - 52 gal/min
14. Fuel Outlet Temp., L/Wing, R/Wing	Telem.	0 - 100°F
15. Fuel Flow, Engine	Photo	
16. Fuel Temp., Out of Hydraulic Oil Cooler	Telem.	
17. Fuel Temp., Into Engine Oil Cooler	Telem.	
* 18. Fuel Temp., Out of Engine Oil Cooler	Telem.	
19. Fuel Pump Inlet Press.	Telem.	0 - 50 PSIA
20. Engine RPM, N_1	Photo	0 - 7000 RPM
21. Engine RPM, N_2	Photo	0 - 11,000 RPM
22. Engine Oil Temp., Into Engine	Telem.	0 - 250°F
* 23. Oil In, Hydraulic Oil Cooler, Primary	Telem.	0 - 300°F
* 24. Oil In, Hydraulic Oil Cooler, Secondary	Telem.	0 - 300°F
* 25. Oil Out, Hydraulic Oil Cooler, Primary	Telem.	0 - 160°F
* 26. Oil Out, Hydraulic Oil Cooler, Secondary	Telem.	0 - 160°F
27 Regulator Press. to Fuel Tank. R/H & L/H (above sensing Press.)	Telem.	0 - 10 PSIG
28 Fuel Tank Sensing Press., R/H & L/H Wings	Telem.	0 - 20 PSIA
* 29 Total Air Press., Upstream of Air-Oil Cooler	Telem.	0 - 37 PSIA
* 30 Engine breather - Out Total Air Pressure		
* 31 Oil Cooler Ramp Static Air Press.		
* 32. Turbine Inlet Total Temp.		
33 Afterburner Fuel Flow		
34 Fuel Pump Interstage Pressure		
35. Fuel Pump Outlet Pressure		
36. Primary Pressure at Pressurization Valve		
37 Secondary Pressure at Pressurization Valve		
38. Pressure at Ignitor Signal Line Tap in A/B System.		

Group No. 3
Type Test: Airplane Performance

	Measurement	Recording	Range
1.	Group 1		
2.	Air Inlet to Engine, Temp.	Photo	-96°F + 300°F
3.	Air Inlet to Engine, Total Press.	Photo	0 - 200" Hg
4.	Comp. Discharge Temp.	Photo	
5.	Comp. Discharge, Total Press.	Photo	0 - 400" Hg
6.	Inter Comp. Total Press.	Photo	0 - 200" Hg
7.	Afterburner Exit Total Press.	Photo	0 - 200" Hg
8.	Engine RPM, N_1	Photo	0 - 7000 RPM
9.	Fuel Flow, Engine	Photo	0 - 26 gal/min
10.	Fuel Flow, Left Wing, Right Wing	Photo	0 - 52 gal/min
11.	Rudder Surface Position	Osc.	±25°
12.	Elevon Surface Position	Osc.	-19° +34°
13.	Speed Brake Actuator Position, Top & Bottom	Photo	0 - 60°
14.	Normal Acceleration, at Pilot	Osc.	-2g +6g
15.	Longitudinal Acceleration, at C.G.	Osc.	±2g
16.	Lateral Acceleration, at C.G.	Osc.	±2g
* 17.	Angular Acceleration, Pitch	Osc.	+3 rad/sec^2
18.	Normal Acceleration, at C.G.	Osc.	-2g +6g
19.	Pitch Attitude	Osc.	±90°
20.	Fuel Outlet Temp., R. Wing & L. Wing	Telem.	0 to 100°F
21.	Flow Limiter, Static Press.	Telem.	0 - 195 PSI
22.	Flow Limiter, Differential Press.	Telem.	
23.	Flow Limiter, Total Temperature	Telem.	
* 24.	Missile Bay Pressures	Osc.	
25.	Tail Section Vent Ducts, Total Pressure	Telem.	0 - 25 PSIA
* 26.	Right Wing Tip Loads, Shear	Osc.	
* 27.	Right Wing Tip Loads, Shear	Osc.	
* 28.	RightWing Tip Loads Torsion	Osc.	
* 29.	Rolling Acceleration	Osc.	± 11 rad/sec^2
30.	Speed Brake Actuator Press.	Photo	0 - 3000 PSI
31.	Angle of Attack, Nose Boom, Aft	Photo	-5° +25 "
32.	Wheel Brake Pneumatic Pressure, L & R	Photo	0 - 3000 PSI

Group No. 4
Type Test: Flutter Analysis

	Measurement	Recording	Range
1.	Group (1)		
2.	Elevon Surface Position	Photo-Osc.	+34° -19°
3.	Rudder Surface Position	Photo-Osc.	±25°
4.	Wing Vibrator Oscillator Signal	Osc.	
5.	Vertical Accelerations, Wings & Elevons	Osc.	±5g
6.	Lateral Accelerations Fin	Osc.	±5g
7.	Wing Vibrator Mass Displacement L & R		
8.	Wing Vibrator Mass Acceleration L & R		

Group No. 5
Type Test: Longitudinal Dynamic Response

1.	Group (1)		
2.	Longitudinal Trim Position	Photo	+12.5° -2°
3.	Elevon Surface Position	Photo-Osc.	+34° -19°
4.	Speed Brake Positions	Photo	0 - 60°
5.	Static Pressure Elevon "Q" Pot	Photo	0 - 60,000 ft.
6.	Differential Pressure Elevon "Q" Pot	Photo	50 - 700 mph
7.	Programmer Longitudinal Signal Output	Osc.	----
8.	Elevon Valve Spool Positions	Osc.	±.125"
9.	Longitudinal Stick Position	Osc.	+20° -9°
10.	Pitch Rate	Osc.	±60°/sec.
* 11.	Pitch Acceleration	Osc.	±3 RAD/sec.^{2}
12.	Pitch Attitude	Osc.	±90°
13.	Angle of Attack	Osc.	+25° -5°
14	Pitch Damper Signal	Osc.	----

Group No. 6
Type Test: Lateral Dynamic Response

		Measurement	Recording	Range
	1.	Group (1)		
	2.	Lateral Trim Position	Photo	±3°
	3.	Rudder Surface Position	Osc.	±25°
	4.	Pitch Attitude	Osc.	±90°
	5.	Roll Attitude	Osc.	±175°
	6.	Angle of Attack	Osc.	+25° -5°
	7.	Angle of Yaw	Osc.	±15°
	8.	Normal Acceleration at C.G.	Osc.	±10g
	9.	Yaw Damper Signal	Osc.	----
*	10.	Yaw Acceleration	Osc.	±1.5 RAD/sec.^{2}
	11.	Yaw Rate	Osc.	±25°/sec.
	12.	Elevon Surface Positions	Photo-Osc.	+34° -19+
	13.	Elevon Valve Spool Positions	Osc.	±.125"
	14.	Lateral Stick Position	Osc.	±20°
	15.	Lateral Stick Force	Osc.	±100 lbs.
*	16.	Roll Rate	Osc.	±300°/sec.
*	17.	Roll Acceleration	Osc.	±3 RAD/sec.^{2}
	18.	Elevon Actuators Positions	Photo	+35° -20°
	19.	Programmer Lateral Signal Output	Osc.	----

Group No. 7
Type Test: Directional Dynamic Response

	Measurement	Recording	Range
1.	Group (1)		
2.	Rudder "Q" Pot Differential Pressure	Photo	0 - 5000 PSI
3.	Speed Brake Positions	Photo	0 - 60°
4.	Rudder Trim Position	Photo	±5°
5.	Rudder Surface Position	Osc.	±25°
6.	Angle of Attack Nose Boom	Osc.	+25° -5°
7.	Programmer Output (Yaw)	Osc.	
8.	Lateral Acceleration	Osc.	±5g
9.	Yaw Damper Signal		
10.	Rudder Pedal Position	Osc.	±3.75 in.
11.	Direction Control Force	Osc.	0 - 300 lbs.
12.	Rudder Control Valve Spool Position	Osc.	±.125 in.
13.	Rudder Control Valve Servo Actuator Pos.	Osc.	±.120 in.
* 14.	Yaw Acceleration	Osc.	±1.5 RAD/sec.^{2}
15.	Yaw Rate	Osc.	±25°/sec.
16.	Yaw Angle	Osc.	±15°
17.	Elevon Surface Positions	Osc.	+34° -19°
18.	Pitch Rate	Osc.	±60°/sec.
* 19.	Roll rate	Osc.	±300°/sec.
20.	Roll Attitude	Osc.	±175°
21.	Directional Heading	Osc.	±180°

Group No. 8
Type Test: Control System Operation

	Measurement	Recording	Range
1.	Group (1)		
2.	Elevon Surface Positions	Photo	+34° - 19°
3.	Elevon Surface Positions	Osc.	+34° -19°
4.	Rudder Surface Position	Photo	±25°
5.	Rudder Surface Position	Osc.	±25°
6.	Longitudinal Trim Position	Photo	+12.5° -2°
7.	Lateral Trim Position	Photo	±3°
8.	Directional Trim Position	Photo	±5°
9.	Rudder "Q" Pot Total Sensing Pressure	Photo	50 - 700 mph
10.	Rudder "Q" Pot Differential Pressure	Photo	0 - 5000 psi
11.	Elevon "Q" Pot Differential Sensing Press.	Photo	50 - 700 mph
12.	Engine RPM N_1	Photo	0 - 7000 rpm
13.	Rudder Pedal Position	Osc.	±3.25 in.
14.	Longitudinal Stick Position	Osc.	+20° -9°
15.	Lateral Stick Position	Osc.	±20°
16.	Longitudinal Stick Force	Osc.	±150 lbs.
17.	Lateral Stick Force	Osc.	±100 lbs.
18.	Rudder Pedal Force	Osc.	0 - 300 lbs.
19.	Rudder Control Valve Spool Position	Osc.	±.125 in.
20.	Elevon Control Valves Spools Positions	Osc.	±.125 in.
* 21.	Rudder Control Valve Servo Act. Pos.	Osc.	±.120 in.
* 22.	Elevon Control Valve Servo Act. Pos.	Osc.	±.120 in.
23.	Angle of Yaw	Osc-Photo	±15°
24.	Angle of Attack	Osc.-Photo	+25° -5°
25.	Rudder Pedal Force Signal Output	Osc.	——
26.	Rudder Cont. Overload Spring Position	Osc.	±18°55'
27.	Lateral Acceleration	Osc.	±2g
28.	Rate of Yaw - C.G.	Osc.	±25°/sec.
* 29.	Rate of Roll	Osc.	±300°/sec.
30.	Elevon "Q" Pot, Static Pressure	Photo	0 - 60000 ft.
31.	Hydraulic Oil Pressure Primary		
32.	Hydraulic Oil Pressure Secondary		
33.	Air Turbine, Outlet Oil Press		0 - 3000 psi

Group No. 9
Type Test: Damping System

	Measurement	Recording	Range
1.	Rudder Surface Position	Osc.	±25°
2.	Rudder Control Valve Spool Position	Osc.	±.125"
3.	Rudder Control Valve Servo Position	Osc.	±.125"
* 4.	Angular Accel., Yaw	Osc.	±.5 rad/sec^2
5.	Lateral Accel.,	Osc.	±2g
6.	Rate of Yaw	Osc.	±25°/sec.
* 7.	Rate of Roll	Osc.	±300°/sec.
8.	Rate of Pitch	Osc.	±60°/sec.
9.	Roll Attitude	Osc.	±180°
10	Pitch Rate Sensing Gyro Output	Osc.	±32V DC
* 11	Rudder Summing Amplifier Output	Osc.	±26V DC
* 12	Rudder C.V. Act. Followup Pot. Output	Osc.	±3V DC
* 13	Rudder Pedal Force Sensing Element Output	Osc.	±60V DC
* 14	Rt. Elevon C.V. Act. Followup Pot Output	Osc.	±8V DC
* 15	Lt. Elevon C.V. Act. Followup Pot Output	Osc.	±8V DC
* 16	Rudder Position Pot. Output	Osc.	±25V DC

Group No. 10
Type Test: Pilot Assist System

Measurement	Recording	Range
1. Group 1		
2. Rudder Feel Cyl. Regulator Press.	Photo	0 - 5000 PSI
3. Elevon "Q" Pot Diff. Press	Photo	50 - 700 MPH
4. Rudder Surface Position	Osc.	±25°
*5. Elevator Pos. at Mixer	Osc.	
6. Aileron Pos. at Mixer	Osc.	
7. Normal Acceleration at Pilot	Osc.	-2g +6g
8. Normal Acceleration at C.G.	Osc.	-2g +6g
*9. Roll Rate Sensing Gyro Output	Osc.	±90V DC
*10. Pilot Assist Vert. Gyro Output, Roll Att.	Osc.	±80V DC
*11. Pilot Assist, Roll Summing Amp. Output	Osc.	±30V DC
*12. Pilot Assist Synch. Pot. Output, Roll	Osc.	±50V DC
*13 Pilot Assist Vert. Gyro Output, Pitch Att.	Osc.	±60V DC
*14 Pilot Assist Pitch Summing Amp. Output	Osc.	±30V DC
*15. Pilot Assist Synch. Pot. Output, Pitch	Osc.	±50V DC
*16. Yaw Rate Sensing Gyro Output	Osc.	±40V DC
*17 Rt. Elevon F.C. Summing Amp. Output	Osc.	±40V DC
*18. Rt. Elevon F.C. Synch. Pot Output	Osc.	±50V DC
*19 Rt. Elevon Position Pot. Output	Osc.	±108V DC
*20. Lt. Elevon F.C. Summing Amp. Output	Osc.	±40V DC
*21. Lt. Elevon F.C. Synch. Pot Output	Osc.	±50V DC
*22. Programmer Elevon Voltage Output	Osc.	±4V DC
*23. Lt. Elevon Position Pot. Output	Osc.	±108V DC
*24. Pilot Assist Up Elevator Signal	Osc.	±30V DC
*25. Compass Coupler Output	Osc.	±25V DC
*26. Compass Coupler Engage	Osc.	28V DC Input
*27. Programmer Rudder Voltage Output	Osc.	±10V DC
*28. Aileron Position Pot. Output	Osc.	
Pitch Attitude Trim Signal (Beep)	Osc.	On or Off
[illegible] Attitude Trim Signal (Beep)	Osc.	Up or Down
Pilot Assist Engage Voltage	Osc.	Up or Down

Group No. 11
Type Test: Engine Cooling

		Measurement	Recording	Range
	1.	Engine Oil, Into Engine	Telem.	0 - 250°F
*	2.	Engine Oil, Out of Engine	Telem.	0 - 350°F
*	3.	Engine Oil, Out of Air-Oil Cooler	Telem.	0 - 285°F
	4.	Engine Oil, Out of Fuel-Oil Cooler	Telem.	
*	5.	Engine Mounts	Telem.	0 - 1100°F
*	6.	Engine Mounts	Telem.	0 - 1100°F
*	7.	Engine Mounts	Telem.	0 - 1100°F
*	8.	Engine Mounts	Telem.	0 - 350°F
*	9.	Engine Surface, Fwd. of Firewall	Telem.	0 - 800°F
*	10	Engine Surface, Fwd. of Firewall	Telem.	0 - 800°F
*	11	Engine Surface, Fwd. of Firewall	Telem.	0 - 800°F
*	12	Engine Surface, Fwd. of Firewall	Telem.	0 - 800°F
*	13	Air In, Air-Oil Cooler	Telem.	0 -180°F
*	14	Air Out, Air-Oil Cooler	Telem.	0 - 300°F
	15	Fuel Out, Hyd. Oil Cooler	Telem.	
*	16	Fuel In, Engine Oil Cooler	Telem.	
	17	Fuel Out, Engine Oil Cooler	Telem.	

Group No. 12
Type Test: Structural Cooling

	Measurement	Recording	Range
* 1.	Frame Sta. 544	Telem.	0 - 350°F
* 2.	Flange, Engine Side, Sta. 544	Telem.	0 - 350°F
3.	Flange, Engine Side, Sta. 513	Telem.	0 - 350°F
4.	Flange, Engine Side, Sta. 513	Telem.	0 - 350°F
* 5.	Wing Root Rib	Telem.	0 - 350°F
*6.	Wing Root Rib	Telem.	0 - 350°F
*7.	Wing Root Rib	Telem.	0 - 350°F
*8.	Wing Root Rib	Telem.	0 - 350°F
*9.	Wing Root Rib	Telem.	0 - 350°F
10.	Rear Spar Carry Thru, Sta. 513	Telem.	0 - 350°F
11.	Rear Spar Carry Thru, Sta. 513	Telem.	0 - 350°F
*12	Rear Spar Carry Thru, Sta. 544	Telem.	0 - 350°F
*13	Rear Spar Carry Thru, Sta. 544	Telem.	0 - 350°F
14	Wing Spar, Sta. 513	Telem.	0 - 250°F
15	Wing Spar, Sta. 513	Telem.	0 - 250°F
16	Wing Spar, Sta. 513	Telem.	0 - 250°F
*17.	Wing Spar, Sta. 544	Telem.	0 - 350°F
*18.	Wing Spar, Sta. 544	Telem.	0 - 350°F
*19.	Wing Spar, Sta. 544	Telem.	0 - 350°F
20.	Fin Spar, Sta. 513	Telem.	0 - 350°F
21.	Fin Spar, Sta. 513	Telem.	0 - 350°F
22	Fin Spar, Sta. 513	Telem.	0 - 350°F
*23	Fin Spar, Sta. 544	Telem.	0 - 350°F
*24	Frame (Minor) Sta. 528.5	Telem.	0 - 350°F
25	Frame (Minor) Sta. 558.5	Telem.	0 - 350°F
*26	Frame (Minor) Sta. 576.0	Telem.	0 - 350°F
*27	Engine Radiation Shield	Telem.	0 - 750°F
*28	Engine Radiation Shield	Telem.	0 - 750°F
29	Engine Radiation Shield	Telem.	0 - 750°F
30	Engine Radiation Shield	Telem.	0 - 750°F

Group No. 12 (Continued)
Type Test: Structural Cooling

	Measurement	Recording	Range
31.	Drogue Chute Pan	Telem.	0 - 350°F
* 32.	Skin Back of Firewall	Telem.	0 - 270°F
* 33.	Skin Back of Firewall	Telem.	0 - 270°F
* 34.	Skin Back of Firewall	Telem.	0 - 270°F
* 35.	Skin Back of Firewall	Telem.	0 - 270°F
* 36.	Air Behind Firewall	Telem.	0 - 285°F
* 37.	Air Behind Firewall	Telem.	0 - 285°F
* 38.	Air Behind Firewall	Telem.	0 - 285°F
* 39.	Air Behind Firewall	Telem.	0 - 285°F
* 40.	Tail Pipe Temp.	Telem.	0 - 1200°F
* 41.	Tail Pipe Temp.	Telem.	0 - 1200°F
* 42.	Tail Pipe Temp.	Telem.	0 - 1200°F
* 43.	Tail Pipe Temp.	Telem.	0 - 1200°F
* 44.	Engine Shroud, Inside Total Air Press.	Telem.	
* 45.	Engine Shroud, Inside Total Air Press.	Telem.	
* 46.	Engine Shroud, Outside Total Air Press.	Telem.	
* 47.	Engine Shroud, Outside Total Air Press	Telem.	
48.	Total Air Press., Upstream Sta. 479	Telem.	
49.	Total Air Press., Downstream Sta. 397	Telem.	
50.	Total Air Press., Upstream Sta. 397	Telem.	

Group No. 13
Type Test: Cooling - Hydraulic & Control Systems

Measurements	Recording	Range
*1. Rudder "Q" Cylinder	Telem.	0 - 280°F
*2. Rudder Trim Actuator	Telem.	0 - 280°F
*3. Rudder Damper Actuator	Telem.	0 - 280°F
*4. Aileron Trim Actuator	Telem.	0 - 280°F
*5. Elevon Cylinder (Inbd. Rt.)	Telem.	0 - 180°F
*6. Elevon Cylinder (Inbd. Lt.)	Telem.	0 - 180°F
*7. Elevon Cylinder Valve	Telem.	0 - 180°F
*8. Elevon Cylinder Valve	Telem.	0 - 180°F
*9. Elevon Cylinder Trim Actuator	Telem.	0 - 280°F
*10. Elevon Cylinder Damper Actuator	Telem.	0 - 280°F
*11. Elevon Damper Actuator	Telem.	0 - 280°F
*12. Elevon "Q" Feel Intelligence	Telem.	0 - 280°F
*13. Top Dive Brake Cylinder	Telem.	0 - 180°F
*14. Dive Brake Valve	Telem.	0 - 180°F
*15. Elevon "Q" Feel Cylinder	Telem.	0 - 350°F
*16. Oil In, Hyd. Cooler, Primary	Telem.	0 - 300°F
*17. Oil In, Hyd. Cooler, Secondary	Telem.	0 - 300°F
*18. Oil Out, Hydraulic Cooler, Primary	Telem.	0 - 160°F
*19. Oil Out, Hydraulic Cooler, Secondary	Telem.	0 - 160°F
20. Oil Hydraulic Res., Primary	Telem.	0 - 160°F
21. Oil Hydraulic Res., Secondary	Telem.	0 - 160°F

Group No. 14
Type Test: Cooling - Engine Accessories

	Measurement	Recording	Range
1.	Fuel Pump, Ambient Air Temp.	Telem.	0 - 250°F
2. *	Fuel Control, Ambient Air Temp.	Telem.	0 - 250°F
3. *	Tach. Generator, Ambient Air Temp.	Telem.	0 - 250°F
4.	GLA Ignition, Ambient Air Temp.	Telem.	0 - 200°F
5. *	Bleed Governor, Ambient Air Temp.	Telem.	0 - 250°F
6. *	Engine Flow Meter, Ambient Air Temp.	Telem.	0 - 250°F
7. *	Air-Oil Cooler Actuator Motor, Amb. Air	Telem.	0 - 180°F
8. *	Fuel Regulator, Ambient Air	Telem.	0 - 250°F
9. *	CVAC "J" Box	Telem.	0 - 270°F
10. *	Air In, Accessory Compt.	Telem.	0 - 180°F
11 *	Air In, Accessory Compt.	Telem.	0 - 180°F
12. *	Air In, Accessory Compt.	Telem.	0 - 180°F
13 *	Air Out, Accessory Compt.	Telem.	0 - 220°F
14 *	Air Out, Accessory Compt.	Telem.	0 - 220°F
15 *	Air Out, Accessory Compt.	Telem.	0 - 220°F

Group No. 15
Type Test: Cooling - High Pressure Pneumatic System

	Measurement	Recording	Range
1. *	High Press. Comp. Air In	Telem.	0 - 270°F
2. *	High Press. Comp. Air Out	Telem.	0 - 800°F
3. *	High Press. Air Bottles	Telem.	0 - 230°F
4. *	High Press. Air Bottles	Telem.	0 - 230°F
5. *	High Press. Air Bottles	Telem.	0 - 230°F
6. *	High Press. Air Bottles	Telem.	0 - 230°F
7. *	Drogue Chute Cylinder	Telem.	0 - 275°F

Group No. 16
Type Test: Cooling - AMT Components

	Measurement	Recording	Range
1.	ATM, Oil In Temp.	Telem.	0 - 300°F
2.	D. C. Generator, Air Out	Telem.	0 - 300°F
3.	A. C. Generator, Air Out	Telem.	0 - 300°F
4. *	High Press. Air Comp. Housing	Telem.	0 - 500°F
5. *	A. C. Alternator Frame Temp.		
6. *	D. C. Generator Frame Temp.		
7. *	ATM Frequency + Voltage		
8. *	ATM Oil Press.		
9. *	ATM Discharge Press.		
10. *	ATM Discharge Temp.		

YF-102 No. 52-7995 FLIGHT TEST PROGRA

CVAC Index No.	Test	Configuration	Trim Speed	Speed Range	Pressure Altitude	G.W.	
2.7	Static Thrust Calibration	--	--	--	Ground	Optional	Opt
4.2	Brake Operation Evaluation	--	--	--	Ground	Optional	Opt
1.5	Taxi Tests	L	T.O.	0 to T.O.	Ground	T.O.	Res
3.10	Engine Control System Operation						
1.12	Shakedown Flight	WO,P,CR,PA,L	Optional	To M = .95	To 35,000	Optimum	Res
4.14	Speed Brake Tests	P,D,PA	[illegible]	To V_{max}	To 35,000	Optional	Opt
4.16	Fuel Tank Pressurization System Tests	All		To Vmax	To 35,000	NSL	26
3.0	Engine Equipment Operation						
1.8	Flutter and Vibration	P	Each speed point	M = .95 to max.	35,000	60% Fuel	Res
2.1	Airspeed Calibration						
	(a) Low Speed	P & D	"	150 - 350 mph	Optional	Optimum	Res
		PA	"	150 - 310 mph	Optional	Optimum	Res
	(b) High Speed	P	"	M ≦ .85	See Remarks	Optimum	Res
		D	"	M = .80 to max.	"	Optimum	Res
1.12	Preliminary Flight Characteristics	All	[illegible]	To V_{max}	To serv. ceiling	NSL	26
5.0	Angle of Attack and Angle of Yaw Pickup Calibration	All		To V_{max}	Optional	Optional	Opti
2.3	Preliminary Airplane Performance [illegible] Phase II Tests (conducted by AFFTC)	P					

YF-102 No. 52-7995 FLIGHT TEST PROGRAM

CVAC Index No.	Test	Configuration	Trim Speed	Speed Range	Pressure Altitude	G.W.	C %
2.7	Static Thrust Calibration	--	--	--	Ground	Optional	Opt
4.2	Brake Operation Evaluation	--	--	--	Ground	Optional	Opt
1.5	Taxi Tests	L	T.O.	0 to T.O.	Ground	T.O.	Res
3.10	Engine Control System Operation						
1.12	Shakedown Flight	WO,P,CR,PA,L	Optional	To M = .95	To 35,000	Optimum	Res
4.14	Speed Brake Tests	P,D,PA		To V_{max}	To 35,000	Optional	Opt
4.16	Fuel Tank Pressurization System Tests	All		To Vmax	To 35,000	NSL	26
3.0	Engine Equipment Operation						
1.8	Flutter and Vibration	P	Each speed point	M =.95 to max.	35,000	60% Fuel	Res
2.1	Airspeed Calibration (a) Low Speed	P & D PA	" "	150 - 350 mph 150 - 310 mph	Optional Optional	Optimum Optimum	Res Res
	(b) High Speed	P	"	M $\leq$ .85	See Remarks	Optimum	Res
		D	"	M = .80 to max.	"	Optimum	Res
1.12	Preliminary Flight Characteristics	All		To V_{max}	To serv. ceiling	NSL	26 -
5.0	Angle of Attack and Angle of Yaw Pickup Calibration	All		To V_{max}	Optional	Optional	Opti
2.3	Preliminary Airplane Performance Phase II Tests (conducted by AFFTC)	P					

ST PROGRAM

.W.	C.G. % MAC	Meas. Req'd. (Instr. Groups)
ional	Optional	2+11+12 +14+16
ional	Optional	3
	Resulting	All
		1+2+3
mum	Resulting	All
onal	Optional	1+3+5+ 6+7
	26 - 30	1+2
		1+2+3+ 11+14
uel	Resulting	1+4
um	Resulting	1+2+3
um	Resulting	1+2+3
um	Resulting	
um	Resulting	
	26 - 30	1+3+8
al	Optional	1+3+8
		1+2+3+ 11+14

YF-102 No. 52-7995 FLIGHT TEST PROGRAM

Index No.	Test	Configuration	Trim Speed	Speed Range	Pressure Altitude	G.W.	C %
1.12	Effect of Missile Bay Door Operation on Flight Characteristics	P		To V_{max}	25,000	26,000	Re
	Airplane Performance, Optimum Wing-Leading-Edge and Tip Config.						
2.3	(a) Speed-Power	P	Each Speed Point	M = .3 to V_{max}	15,000 & 40,000	NSL	26
2.5	(b) Drag Polars	P	"	M = .6 to V_{max}	15,000 & 40,000	NSL	26
	(c) Take-off Distance	WO	1.4 V_{S_L}	—	Ground	NSL	
	(d) Landing Distance	L	1.4 V_{S_L}	—	Ground	NSL	26
3.1	Engine and Structural Cooling	All	—	To V_{max}	To Service Ceiling	NSL	26
1.7	Dynamic Response	All	—	To V_{max}	"	NSL	26
1.1, 1.2, 1.4	Stability and Control Stall Tests	All	As Specified	To V_{max}	12,000 & 30,000	NSL	26
4.6	Pilot Assist System Development	All		M = .7 to .95	To 50,000	NSL	26
	Engine Operation - Air Starts	D	—	—	To 40,000	Optional	Op
	Drogue Chute Tests	L	1.4 V_{S_L}	30 -160 mph	Ground	Optional	Op
	Emergency Fuel Control System Tests	P	—	—	5,000 & 35,000	Optional	Op
	Emergency "Ram-Air" Hydraulic Power Unit Tests	D	—	To V_{max}	35,000	Optional	Op
	Flight Control System	All	All	To V_{max}	To Service Ceiling	NSL	26
3.0	Engine & Associated Equipment Operation, Phase III Tests	As Required	All	—	—	—	

RAM

C.W.	C.G. % MAC	Meas. Req'd. (Instr. Groups)
6,000	Resulting	1+3+8
SL	26 - 30	1+2+3+ 11+14
SL	26 - 30	
SL		
L	26 - 30	
L	26 - 30	1+2+11 +12+14
L	26 - 30	1+3+5+ 6+7+8
	26 - 30	1+5+8+9
	26 - 30	1+5+6+7 +8+9+10
ional	Optional	1*2*16
ional	Optional	1+12
ional	Optional	1+2
ional	Optional	1+8
	26 - 30	1+3+8+9
	--	As Req'd.

APPENDIX I

OPERATIONAL BREAKDOWN
OF
FLIGHT TEST PROGRAM

TESTS TO BE ACCOMPLISHED:

The flight test program describes the testing to be conducted by CVAC on the second YF-102 airplane, USAF S/N 52-7995. For ease of operation in the field, this program has been divided into thirteen (13) phases and a detailed flight plan will be constructed for each phase. The test phases are as follows:

A. Ground Tests
B. Aircraft Systems Shakedown and In-Flight Flutter Investigation
C. Preliminary Flight Characteristics
D. Preliminary Performance Evaluation with Straight Wing Leading Edges
E. Preliminary Performance Evaluation with Cambered Wing Leading Edges and Reflexed Tips
F. Optimum Performance Evaluation
G. Dynamic Response
H. Stability and Control
I. Drogue Chute Evaluation
K. Missile Bay Door Evaluation
L. Development of the Pilot Assist System
M. Airstart Program
N. Emergency Equipment Operational Check

Although some overlap of applicable test data will be encountered from one phase to another, each phase will be considered to be a test program in itself and all efforts made to bring it to completion before undertaking another portion of the overall flight test program.

The tests are listed in the approximate order in which they will be conducted, however, this may be revised at any time during the program in order to obtain the data for which the requirement is the greatest at that particular time.

The USAF Phase II Flight Tests will be conducted by the AFFTC, at the present time upon completion of Phase F. At this time, sufficient knowledge should have been gained to present them with the airplane at its optimum drag configuration [illegible] with the required operating data to satisfactorily conduct their flight

BREAKDOWN OF PROGRAM PHASES

A. Ground Tests

1. Ground Static Thrust Calibration:

A ground static thrust calibration will be obtained on the AFFTC Universal Thrust Stand for each engine and/or afterburner installed during the course of the tests. These calibrations will serve a twofold purpose in that: (1) A check will be made on the thrust of the installed power plant and a determination made of its operating characteristics, (2) A nozzle calibration will be obtained for determination of in-flight thrust.

2. Brake Pedal Force Calibration and Operation Evaluation:

This test will be conducted to determine the relationship between brake pedal force, brake hydraulic pressure, and brake cylinder pneumatic pressure.

3. Taxi Tests:

Prior to flight, taxi tests will be conducted at speeds up to airplane lift-off for the purpose of determining ground handling, engine and structural cooling characteristics, and ground operation of all airplane equipment.

4. Other items to be checked in conjunction with the above tests include:

a. Engine control system operation
b. Engine equipment operation
c. Engine and structural cooling

B. Aircraft Systems Shakedown and In-Flight Flutter Investigation

Systems Shakedown:

A series of shakedown flights will be required to determine the general flight handling characteristics, engine operation and cooling, structural cooling and operation of all systems and airborne equipment. The systems shakedown will include evaluation of the following:

a. Flight controls
b. Engine controls
c. Engine and structural cooling
d. Low and high pressure pneumatic systems
e. Hydraulic system
f. Electrical system

B. Aircraft Systems Shakedown and In-Flight Flutter Investigation (Continued)

1. Systems Shakedown:

   g. Heating and ventilating system
   h. Missile bay door operation
   i. Speed brake operation
   j. Fuel tank pressurization
   k. Engine equipment operation
   l. Afterburner operation
   m. Fuel System operation

2. Flutter Investigation:

   The purpose of the flutter investigation is to demonstrate that the airplane has positive damping of externally induced vibrations throughout the entire speed range. This will be demonstrated by the use of mechanical shakers mounted internally in the aircraft's wings. The shakers will be vibrated at specific critical frequencies, turned off and the decay measured. This will be performed throughout the series of flights at increasing Mach numbers until the maximum speed of the aircraft is reached.

3. Other data that will be obtained concurrently with the above tests include the following:

   a. Preliminary airspeed calibration
   b. Preliminary drag and speed-power evaluation
   c. Qualitative analysis of the airplane handling characteristics

C. Preliminary Flight Characteristics

1. General Handling Characteristics:

   The general flight handling characteristics will be evaluated qualitatively and implemented wherever possible by quantitative data. Compliance with all phases of Specification 1815-B will not be required during this phase of the program since the airplane control system will not be in its final configuration. For this reason, special attention will be given to the following:

   Control system effectiveness
   Control forces
   Trim control operation

2. Quantitative Stability Investigation:

   The following flight characteristic items will require that tests be conducted to determine their values. The tests will be conducted at an altitude of approximately 30,000 feet with the airplane in the clean configuration. Test data will be obtained for two appropriate trim speeds in each case:

a. Static longitudinal stability
b. Dynamic longitudinal stability
c. Static directional stability
d. Dynamic directional stability
e. Maneuvering flight characteristics
f. Dynamic lateral stability
g. Aileron characteristics

3. Other test data that will be obtained concurrently with the above tests include the following:

a. Preliminary airspeed calibration
b. Engine and structural cooling

D. Preliminary Performance Evaluation with Straight Wing Leading Edges

1. Airspeed Calibration:

The airspeed calibration will be finalized during this phase of the program. Methods of calibration to be utilized will include: pacer aircraft, tower and aircraft flybys, and radar tracking of climbs and high speed dives.

2. Climb Performance:

Check climbs will be made to approximate service ceiling at predetermined airspeed schedules to determine the following: best climbing speed, optimum rate of climb, and service and combat ceilings. These items will be determined for both maximum and military power conditions.

3. Level Flight Performance:

Complete speed versus power required and thrust calibrations will be conducted at indicated pressure altitudes of 15,000 and 40,000 feet. Data will be obtained in stabilized level flight for predetermined increments of airspeed.

Take-off and Landing Performance:

Take-off and landing performance will be determined by use of the AFFTC photo-theodolite. No additional flights will be flown for this data, but all take-offs and landings conducted from the concrete runway, during this phase, will be photographed for this purpose.

D. Preliminary Performance Evaluation with Straight Wing Leading Edges (Continued)

5. Data pertaining to engine and structural cooling will also be obtained during this phase of the program.

E. Preliminary Performance Evaluation with Cambered Wing Leading Edges and Reflexed Tips

1. These tests, together with the previous phase (D) are planned primarily to determine the optimum wing leading edge and tip configuration. All performance testing listed under phase D will be repeated in this phase except for airspeed calibration where checks will be performed to detect any shift in the position error.

F. Optimum Performance Evaluation

With the airplane at its optimum drag configuration, based on the results of the preliminary performance evaluations, the following final performance tests will be conducted:

1. Drag analysis for both level and accelerated flight
2. Final checks on climb performance and service ceiling
3. Optimum take-off and landing performance

G. Dynamic Response Tests

The purpose of these tests will be to determine the dynamic stability and response of the airplane and control system to predetermined control system input signals. The tests will be performed by utilizing an automatic programmer to deliver the input signals to the control system. Detailed requirements for this phase will be submitted by the CVAC dynamics group at a later date.

H. Stability and Control

Complete stability and control tests will be conducted during this phase of the airplane for compliance with USAF Specification 1815-B. All paragraphs and requirements of this specification will be investigated and complied with where practical. It is estimated that this phase of the program will require approximately 45-50 hours of flight time.

[illegible] of Drogue Chute

Drogue chute installation shall be evaluated as to effectiveness and [illegible] technique required. This shall be done mainly during taxi runs and at termination of other scheduled flights and should require no [illegible] for this phase of the program.

K. Evaluation of Missile Bay Doors

A short program will be necessary to determine the effect of open missile bay doors on the aircraft flight characteristics and performance. This will be investigated through the speed ranges of level flight and with the aircraft in a shallow dive. A check will also be made of the aircraft's landing characteristics with the missile bay doors in the open position.

L. Development of the Pilot Assist System

For this test, the utilization of a programmer will again be made for introducing input signals to the control system. The detailed test procedures have not been developed at this date but will be supplied by the CVAC dynamics group.

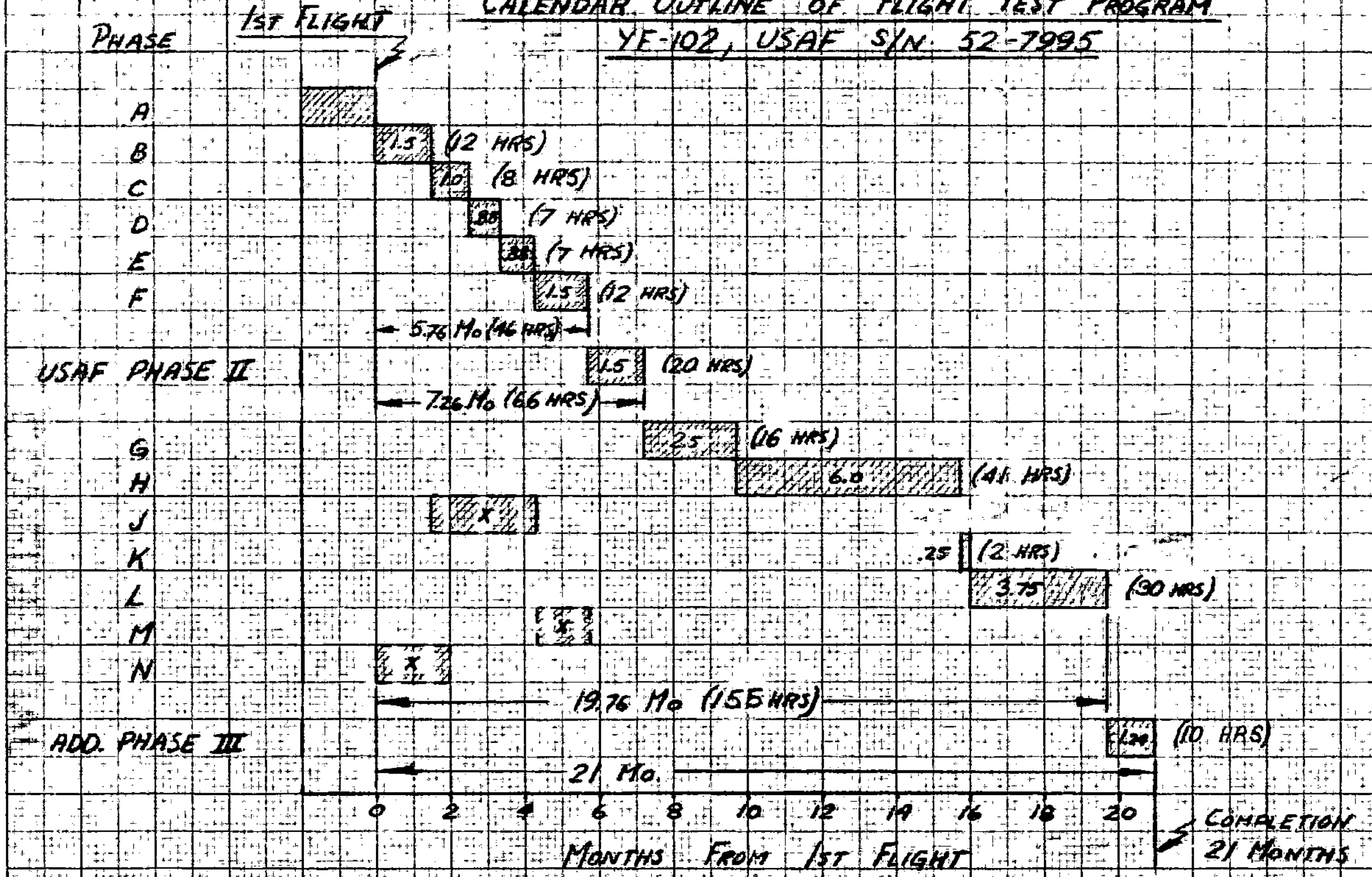
M. Airstart Program

An airstart program will be conducted to determine the altitude and speed ranges in which satisfactory air starts can be accomplished. This phase will not require any special flights but can be accomplished concurrently with other phases of the program.

N. Emergency Equipment Operation

The purpose of this phase is mainly to determine that the emergency fuel control system and ram air hydraulic power unit operate satisfactorily No special flights will be required and all test data can be obtained concurrently with the performance phase of the program.

CALENDAR OUTLINE OF FLIGHT TEST PROGRAM
YF-102, USAF S/N 52-7995



X — — — NO SPECIAL PERIOD ALLOTTED, DATA OBTAINED CONCURRENTLY WITH APPROX. PHASE SHOWN

FLIGHT TESTS BASED ON AVERAGE FLYING RATE OF 8.0 HRS/1.0 MONTH

12-1-53 Fel

FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No. 52-7995 Addendum No. 1

Item Description: * Landing Gear Hydraulic Actuating Pressures.

Purpose: To determine main landing gear and nose landing gear actuating pressures vs gear position and time. To prove design and measure airloads.

Configuration: V_{io} = 230 and 264 mph, minimum practical altitude, optional G.W. and C.G., speed brakes in, level flight where possible.

Procedure: 1. Retract and extend landing gear during accelerated flight with the following load factors: +1g, +1-1/2g, and approximately 0g, -1g.
2. Repeat test with restrictor in "up" line to give a minimum retraction time of 15 sec.

Measurements Required: V_{io}, pressure altitude, normal acceleration, hydraulic actuator pressure for the nose landing gear cylinder and one main landing gear cylinder, (pressures at cylinder inlet and outlet) and landing gear actuator extension.

* Additional Instr. Required.

F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Report No. ZC-8-103
Addendum No. 2

Airplane No. XF-102 S/N 52-7995

Item Description: * Inlet Duct Performance

Purpose: (a) To determine inlet duct loss

(b) To evaluate Std. Pt2 instrumentation

Spec. No. No

Configuration

Procedure: Obtain inlet duct pressures in the following flight conditions:

Power	Mach. No.	Condition	Press. Alt.
Military	0.80 to 0.95+	Level - Acceleration	25,000 Ft.
85% Military	0.90 to 0.95	Approximately Level	35,000 Ft.
70% Military	0.90 to 0.95	Approximately Level	45,000 Ft.
Military	0.7 to 0.9	Climb	10,000 - 35,000
85% Military	0.7 to 0.9	Climb	20,000 - 30,000
70% Military	0.7 to 0.9	Climb	20,000 - 30,000

Measurements Required: Pt2, special Pt2 pickups, Ps manifolds at duct inlet and aft end of duct per measurement requirement No. 82-7. Also Std. power and performance measurements.

Remarks: Determine concurrently with other tests.

* Additional Instr. Required.

Requested by: T P Maloy Group
H. K. Cheney

F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No. 52-7995

Report No. 7C-t-10

Addendum No. 3

Item Description: * Dummy fuel tank vent pressure, L/H reflexed wing tip.

Purpose: To survey static pressure at the proposed wing tip fuel tank vent location.

Spec. No. None

Configuration: Cambered Wing Leading Edges and Reflexed Tips

Procedure: No special procedure required. Record data throughout flight during which a climb to altitude is made and over the maximum permissible range of indicated airspeeds.

Measurements Required: Absolute static pressure inside fairing at approximately dummy fuel tank vent location in L/H reflexed wing tip.

* Additional Instr. Required.

Requested by Power Plant Grp.

Approved by:
T. L. Maloy

F-102

FLIGHT TEST PROGRAM ADDENDUM ITEMS

Report No. ZC-8-303

Airplane No. XF-102 S/N 52-7995

Addendum 4.

Item Description: Right hand engine inlet duct pressures and strains.

Purpose: To secure high frequency pressure and strain data for:

(a) conditions considered critical for duct strength, and

(b) comparison with engine test stand data.

Configuration: Reworked inlet duct for Flight No. 10

Procedure: Obtain data for the following conditions:

- A. Ground
  - 1. Stabilized power settings
    - a. idle
    - b. 60% Mil. thrust
    - c. Mil thrust with A/B
  - 2. Slow engine acceleration (idle to Mil with √' and deceleration.
- B. Take-Off and Climb
  - 1. Continuous records during T.O. until 15 seconds after A/B turned off.
  - 2. 10 second record run at each 5000 ft. altitude until leveling out for subsequent runs.
- C. Level Flight (Stabilized)
  - 1. At 15,000 ft. altitude
    - a. 60% Mil. thrust
    - b. Mil. thrust
    - c. Mil. thrust with A/B
  - 2. At 25,000 feet, repeat above
  - 3. At 35,000 feet, repeat above
- D. Mach. No. greater than 1.0

Measurements Required:

A. Location of pressure and strain measurement points, including range and accuracy, are listed in Flight Test Instrument Requirements 82-8 and 82-[illegible]. The principal requirement is determination of highfrequency (0-2000 cps) pressure and strain fluctuations. Measurements may be commutated on an oscillograph at about 4 points per second, except for 4 pressure points (see 82-8) to be recorded continuously.

B. The following required measurements are already instrumented in the airplane. See Groups 1 and 2, pages 52 and 53.

1. CIT (T_{t2})
2. Pressure altitude
3. Airspeed
4. Pitch attitude
5. Turbine discharge temperature
6. Turbine discharge total pressure
7. Engine RPM, N_1
8. Engine RPM, N_2
9. Nozzle position
10. Surge bleed valve position
11. Air inlet to engine, temperature
12. Air inlet to engine, total pressure

C. The following measurements (Group 2, Page 53) are desirable if the airplane is so instrumented at the time.

13. Intercompressor, temperature
14. Intercompressor total pressure
15. Compressor discharge, total temperature
16. Compressor discharge, total pressure
17. Fuel flow, engine
18. Fuel flow, afterburner
19. Afterburner exit, total pressure
20. Power lever angle
21. Inlet pressure, fuel pump
22. Total air pressure upstream of air-oil cooler
23. Static air pressure, oil cooler ramp.

Requested by Structures & Power Plant

Approved by T. L. Maloy
T. L. Maloy

C OLIDATED VULTEE AIRCRAFT CORPO ION
San Diego Division - San Diego, California

F-102 FLIGHT TEST PROGRAM ADDENDUM ITEMS

Report No. ZC-8-103

Airplane No. 52-7995

Addendum No. 5

Item Description:

Dynamic Directional Stability.

Purpose:

To establish the inherent stability characteristics of the F-102 airplane.

Spec. No.

R-1815-B

Configuration:

Yaw damper inoperative (soonest possible flight).

Procedure:

Tests to be performed at 15,000' and at 30,000' at lowest middle and highest practical speeds at each altitude. Pilot shall excite the airplane laterally by application of a kick to the rudder, and allow the resulting oscillations to damp out. Rudder kick should be at least 2° amplitude for approximately one second. This test should be performed at least once with pilot's feet in normal position on the pedals, and once with pilot's feet removed from pedals during damping cycle. (Control stick should not be moved during damping cycle.)

Measurement Required:

The following measurements at the indicated sensitivities are required.

Yaw angular acceleration	.82 rad./sec/sec/inch oscillograph trace
Yaw rate gyro	2°/sec/inch
Lateral Acceleration (all pickups)	.25g/inch
Pedal force	10 lbs/inch
Pedal travel	.5 inch/inch
Rudder valve	.03 inch/inch
° Position of pivot point at pilot's input on rudder walking beam	.03"/inch

Possible output of damper-system-yaw rate gyro and roll-rate-gyro.

° Additional Instr. Required.

Requested by: DYNAMICS Group

Approved by:
T. L. Maloy
Group Engineer
Experimental Flight

TLM:lp

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
San Diego Division - San Diego, California

F-102 FLIGHT TEST PROGRAM ADDENDUM ITEM

Report No. ZC-8-103

Airplane No. 52-7995

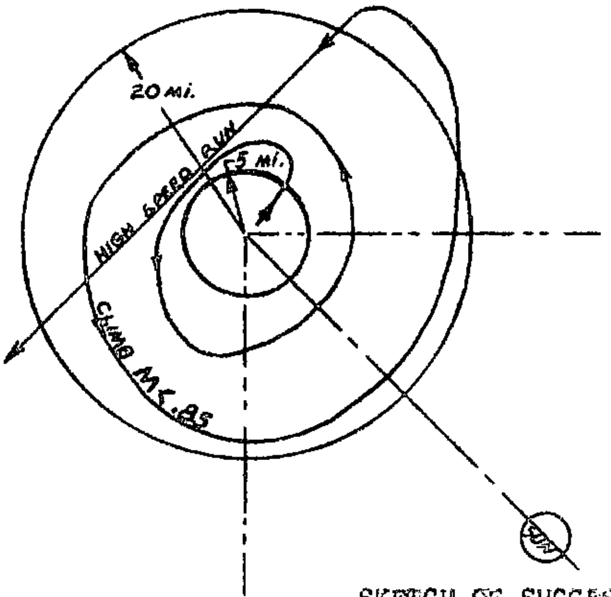
Addendum No. ______

Item Description: Airspeed calibration - Transonic speed.

Purpose: To determine the test nose boom airspeed position error. (This addendum is intended as a clarifying addition to CVAC Index No. 2.1, page No. 14, Report No. ZC-8-103.

Procedure:

1. Request special Radiosonde data at approximately the time of flight.
2. Track the entire flight of the airplane, or as much of it as possible with the radar available at EAFB using 2.0 second intervals between data points.
3. Obtain photo panel data at a minimum rate, of one frame every two seconds during the entire flight.
4. Climb at NRT to 37,000' P.A. Maintain indicated speed over the range of M = .6 to M = .8, but do not exceed M = 0.85. Maintain flight pattern during climb as shown in following sketch. Stay within 20 miles distance of the radar station during climb and fly a pattern which keeps the airplane opposite the sun as much as possible.
5. Make high speed runs starting at about 20 miles out at the highest level flight speed to be attained for that particular days flight. Make these high speed runs approaching but not crossing directly over the radar tracking station, as indicated on the following sketch.
6. Descent may be made at any speed, but airplane should be kept within a 20 mile radius and out of the sun's quadrant as much as possible.



SKETCH OF SUGGESTED FLIGHT PATTERN

Requested by: AERODYNAMICS
Group

Approved by:
H. K. Cheney
Chief Flight Test Engr.
Experimental Flight

San Diego

F-102

[illegible]
Sheet 1 of 2

FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Addendum No. 7-A

Item Description:

"A" Revision. 26 Oct. 1954
Added Sheet 2 (Pump Temperature Tests).

Primary hydraulic system oil temperatures.

Purpose:

To determine the necessity of an oil cooler for the primary hydraulic system by measuring the designated oil temperatures with (1) the oil cooler operative, and (2) the oil cooler by-passed.

Configuration:

Optional.

Procedure:

Make measurements at convenient intervals (not to exceed 10 seconds) during a typical performance flight.

Note: Temperature pick-ups, if insulated from ambient air, may be installed on the outside of the hydraulic oil lines.

Measurements Required:

Hydraulic Oil Temperatures		References	
		ZC-8-103	ZC-8-052
Reservoir	C - 160°F	Page 64 #20	Page II - 8 [illegible]
Pump Outlet	C - 180°F		
Oil into Hydraulic Oil Cooler	C - 300°F	Page 64 #16	
Oil out of Elevon Servo Valve	C - 280°F	Page 64 #10	
Speed Brak[illegible] Line	C - 280°F	Page 64 #13	
Out of [illegible]	C - 280°F	Page 64 # 3	

[illegible]l Instr. R[illegible]

Requested by Hydraulics Group

Approved by T. L. Maloy
T. L. Maloy
Group Engineer
Experimental Flight

TLM:lp

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Addendum No. 7-A

"A" REVISION:

PURPOSE:

The purpose of this revision is to require additional testing to determine the cause of the high temperature conditions which were experienced at the hydraulic pumps after the deletion of the hydraulic fluid coolers. Data analysis shall include determination of the following:

A. Amount of heat added to the pump by engine radiation.

B. Amount of heat carried across the flanges by conduction.

C. Amount of heat added by internal leakage within the pump.

PROCEDURE:

Measurements shall be recorded during several typical flights, and maintained for at least two minutes after shutdown.

MEASUREMENTS:

In addition to Items 1, 2 and 4 of Sheet 1, the following temperatures shall be recorded, for the Primary System:

A. Lower side of pump flange.

B. Lower side of mating engine flange.

C. Top aft end of pump case.

D. Pump internal by-pass (Case drain line).

NOTE: Instrument Requirements No. 82-31.

Requested by General Design

Approved by D. H. Bennett
for E. D. Shannon
Chief of Experimental Flight

EDS:JHS:gh

F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Report No. ZC-8-103
Airplane No. XF-102, S/N 52-7995 Addendum No. 8

Item Description: Pressure Data on the elongated nose.

Purpose: Data is needed for production airplane design and to lift restrictions on present design.

Spec. No.: None

Configuration: 8-80 #2 with elongated nose modification.

Procedure: Obtain continuous pressure measurements over the available Mach No. range at 8000 feet, 15,000 feet, and 30,000 feet. Load factor to be held to unity.

Measurements Required: Refer to EF-8-203 memo soon to be released. This memo references proper tap locations and instrumentation required. Parts are presently being manufactured in San Diego. Maximum pressures expected are +6 PSI -3 PSI. High frequency pressures are not required.

Requested by: DYNAMICS Group

Approved by: T. L. Maloy
T. L. Maloy
Group Engineer
Experimental Flight

* ADDITIONAL INSTR. REQUIRED.

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F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

"B" Revision, 25 Oct. 1954
Deleted fin ram air inlet test
Reason: Test assigned to
AF53-1783.

Report No.: ZC-8-103
Addendum No.: 9-B

"A" Revision:
Revised and rewritten 23 July 54
Reason: To include data for drag
evaluation

Airplane No.: 52-7995

Item Description:

Aftbody Fillet (Yellow Canary), afterburner nozzle actuator, ~~and~~ [illegible]

Purpose:

To obtain pressure and/or temperature data for the above items.

Instrumentation: Noted

Procedure:

A. Yellow Canary Evaluation/

1. Install pressure pickups (and one temperature pickup) in accordance with Thermodynamic Memo T-I-178 dated 12 July 1954.

2. The pressure pickups and a single temperature pickup are located on the inner surface of the yellow canary, aft of the engine EXIT. These should be recorded during one ground engine run up (afterburner included), and flight to maximum altitude with afterburner, in addition to the flights of paragraph A3.

3. All pressure and temperature pickups should be recorded simultaneously under the following flight conditions:

   a. Level flight at 15,000 feet, covering the full Mach Number range.

   b. Level flight at 30,000 feet, covering the full Mach Number range.

   c. Flight at supersonic velocity for at least 30 seconds. It is requested that Mach No. 1.1 or better be maintained. Record during preliminary dive, and during the subsequent transition to subsonic velocity.

Procedure: (Continued)

d Record angle of attack and elevon surface position for the above flights. These are expected to effect the pressures.

b Temperature of Afterburner Nozzle Actuators:

1. Install temperature pickups on upper and lower surface centerline afterburner nozzle actuators. Obtain temperature data before and after shielding, for a ground engine run up (afterburner included) and flight to maximum altitude with afterburner.

NOTE: This is for record only. Tests have been accomplished.

~~[illegible]~~

~~[illegible]~~

~~[illegible]~~

Measurements Required:

Item	Range	Reference
Yellow canary pressure	0-200PSF	Thermo Memo No. T-R 178
Yellow canary temperature	0-1100°F	Thermo Memo No. T-R 178
Temperature at Afterburner Nozzle Actuators (accomplished)	0-1100°F	[illegible] Rep. No. 82-25
~~[illegible]~~	~~[illegible]~~	~~[illegible]~~
~~[illegible]~~	~~[illegible]~~	~~[illegible]~~
~~[illegible]~~	~~[illegible]~~	~~[illegible]~~

Approved by:

Requested by: ... Group

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F 762

FLIGHT TEST PROGRAM ADDENDUM ITEM

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Airplane No. 52-7995

Report No. ZC-8-103

Addendum No. 10 - A

Item Description:

Engine performance tests at two conditions of bleed air consumption.

Note: This addendum is confirmation of advance wire to V. L. Allwardt, Edwards Air Force Base, dated 24 May 1954.

Purpose:

To determine the effect of bleed air on engine performance.

Spec. No. None

Configuration: Normal

Procedure:

At approximately 40,000 feet at maximum power fly 10 consecutive runs as follows:

A. 10 - 20 seconds with ATM bleed only.
B. 10 - 20 seconds with ATM bleed plus exhaust jet pump and generator jet pump bleed.

Note: Altitude and speed should be kept as constant as possible. A parallel circuit from the nose wheel switch will permit the pilot to operate the jet pumps when desired.

Measurements Required:

Use existing instrumentation to record the following:

A. Jet pumps on or off
B. Flow Limiter differential pressure
C. Flow Limiter static pressure
D. Flow Limiter air temperature
E. PT7 and TT7
F. N_1 and N_2
G. PT2
H. OAT
I. T_{t4}

"A" Change:

Added T_{t4} and warning note.

Warning: Pilot should watch rise in T_{t7} when jet pumps are operating. Pilot comments are requested regarding any noticeable thrust dropoff when jet pumps are operated.

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Approved by T. L. Maloy
T. L. Maloy
Group Engineer
Experimental Flight

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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103

Addendum No. 11

Item Description: Drag data test.

Purpose: To determine drag effects in the region of Mach 0.90.

Procedure: Perform flights as follows:

Test No.	Mach No.	Load Factor	Altitude	GW 26,000 Corr. Wt. (W/Delta)
1	.90	1G	47,500	200,000
2	.85	2G	33,000	200,000
3	.80	2G	33,000	200,000
4	.90	1.5G	44,700	263,000
5	.85	1.5G	44,700	263,000
6	.80	1.5G	44,700	263,000
7	.90	2G	41,500	300,000

Measurements Required: Same as for previous drag data tests per Note. 2.

NOTES:

1. This addendum is confirmation of advance wire to V. L. Allwardt, EAFB, dated 6-11-54 (copy to W. Reda).

2. The above tests are for completion of drag data tests initiated during the Phase II testing of 52-7995 at EAFB.

3. Point No. 7 may be marginal power-wise. If level flight cannot be maintained at 2G, themaximum load factor for level flight should be used.

Approved by T. L. Maloy
T. L. Maloy
Group Engineer
Experimental Flight

TLM:lp

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"A" Revision: This addendum for reference only. Test was performed on S/N 53-1781.

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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No [redacted]

Report No. ZC-8-103

Addendum No. 12 - A

Item Description: Preliminary ram air turbine tests.

Purpose: To test the ability of the turbine to maintain adequate hydraulic pressure during landing conditions. The landing condition is the most critical for the turbine because of reduced "Q" pressure.

Procedure:

1. Install a pressure pickup in the pressure line between the ram air turbine and the flow control valve.
2. Extend the ram air turbine just prior to normal take-off from the concrete runway, and retract the landing gear after take-off.
3. Circle the base at moderate speed and perform a normal approach and landing. In order to get maximum interference with the airstream ahead of the turbine withhold the lowering of the gear until the latest practicable moment.

Measurements Required

1. Ram air turb[illegible] pump pressure, (0-3500 psi)
2. [illegible] sure.
3. [illegible] d temperature
4. [illegible] on
5. [illegible]

[illegible] secondary hydraulic power for above tests [illegible] Item No. 13 contains a more detailed test procedure and will be [illegible] when approved.

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[illegible]

[illegible]

Approved by

[illegible] Cheney
Chief Flight Test Engineer
Commercial Flight

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A division of General Dynamics Corporation

San Diego

"A" Revision:
This addendum for reference only. Test was performed on S/N 53-1783.

YF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.

Report No. ZC-8-103

Addendum No. 12 - A

ITEM DESCRIPTION: Simulated landings with ram air turbine emergency hydraulic power.

PURPOSE: To test the ability of the ram air turbine to furnish adequate hydraulic power for emergency landings. In addition, the tests described herein will test the ability of the primary system alone, or the secondary system alone to properly control the aircraft.

CONFIGURATION: Test with TURB 7 (replaces TURB 6). Modify hydraulic systems in accordance with Figure 1 of this addendum.

CAUTION: Addendum No. 12 has been prepared to obtain preliminary information prior to modification in accordance with Figure 1 herein. The flight test results of Addendum No. 12 should be evaluated and approved before the tests of this addendum are conducted.

PROCEDURE:

1. The valves and switches of Figure 1 should be installed and operated sufficiently to insure proper functioning of these components. The hydraulic ground test stand may be used to furnish pressure for the initial testing, but a ground run test of the aircraft engine-driven Vickers pumps should also be performed before the first flight is attempted.

2. Take off and climb to 30,000 feet and perform the following tests:

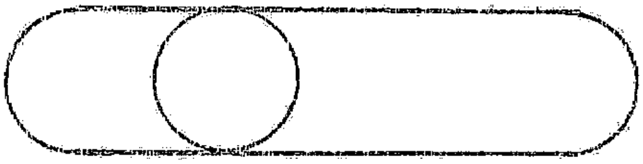
   a. Shut off primary system pressure, and operate the aircraft with secondary pressure alone. Perform sufficient maneuvers to demonstrate adequate control of the aircraft (see Note 1).

   b. Restore the primary pressure, and operate the aircraft with primary alone by shutting off the secondary pressure and repeating the maneuvers of paragraph 2a above.

   c. Restore secondary pressure, and extend the ram air turbine at 30,000 feet and Mach. No. 0.5 in order to test the performance of the turbine extension mechanism. Perform a normal power approach and landing.

d. After the foregoing preliminary tests are conducted, the aircraft should be placed on jacks, and an emergency gear drop performed with pneumatic pressure, to insure that the emergency system is functioning properly. The use of this system is required for the second flight.

e. For the second flight, a smoke laying aircraft shall take off along with the F-102, and shall construct a landing strip outline in smoke 6000 feet from the ground (approx. 8100 feet altitude at EAFB). Shape of the strip should be approximately as follows:



Size of the strip must take into account the turn radius capability of the smoke laying aircraft.

f. The F-102 should climb to 30,000 feet while the smoke outline is being completed. When this altitude is reached, shutoff the primary system hydraulic pressure. Extend the ram air turbine at 30,000 feet and Mach No. 0.5. Observe the primary system pressure gage in the cockpit to see that the ram air turbine unit developes at least 2700 psi hydraulic pressure. Shut off the secondary pressure and perform mild maneuvers of the aircraft with RAM AIR TURBINE POWER ONLY at 30,000 feet and Mach No. 0.5 After the adequate control of the aircraft has been determined, gradually cut the engine power to the idle position, but do not permit the engine to flame out.

[illegible] should now be performed from 30,000 feet to the smoke landing strip of paragraph e. The glide should be regulated so that an indicated airspeed of [illegible] knots is obtained when the altitude reduces to [illegible] feet. At this point the gear should be extended by use of EMERGENCY PNEUMATIC PRESSURE. This will test the capacity of the RAT powered control surfaces to maintain stability in the glide condition during the gear drop. The gear must now remain extended for the duration of the flight.

h Continue the glide at 225 knots, and make a normal approach and "landing" on the smoke landing strip of paragraph e, with a "touchdown" speed of approximately 140 knots indicated. Restore engine thrust before aircraft speed drops below 125 knots indicated. NOTE: Maximum recovery speed with gear extended is 240 knots indicated. Restore the primary and secondary system hydraulic pressure, and perform a normal landing on the base.

A ground landing with ram air turbine power alone, or a complete engine flameout condition should not be attempted without proper authorization.

DATA REQUIRED:

1 RAT hydraulic pump pressure (0-3500 psi)

2 Primary system hydraulic pressure

Secondary system hydraulic pressure

Primary hydraulic reservoir fluid temperature

Rudder and elevon surface position

RAT speed, RPM (proposed)

Normal acceleration at C.G.

[illegible]

Angle of attack

Angle of sideslip (yaw)

[illegible]

[illegible]

Full or severe maneuvers should not be attempted with primary pressure alone, secondary pressure alone, or ram air turbine pressure alone. The critical requirement should be considered as a landing under gusty air conditions. The pilot should estimate the maneuvering required to simulate this condition.

Complete chase plane photographs of the ram air turbine maneuvers should be obtained.

Approved by C. S. Maloy
For H. K. Cheney
Chief Flight Test Engineer
Experimental Flight

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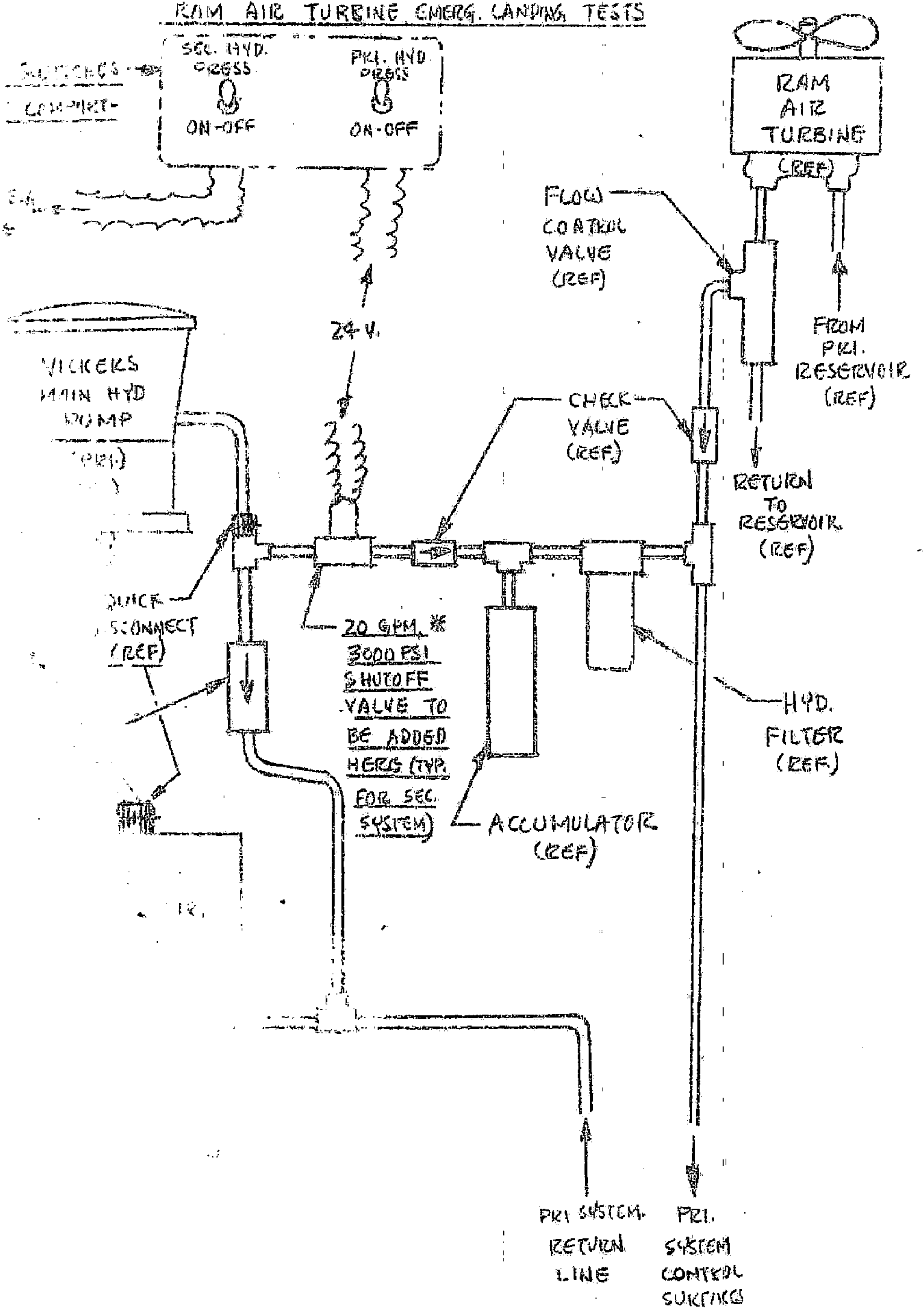
FIGURE 1
BC-8-103
ADDENDUM NO. 13-A

SCHEMATIC DIAGRAM FOR
RAM AIR TURBINE EMERG. LANDING TESTS



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YF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 14A

Item Description: Longitudinal Sensitivity

REVISED & REWRITTEN
19 October 1954

PURPOSE

Tests outlined herein are for the solution of control problems regarding the sensitivity, friction, stick lightening and stick force slope of the longitudinal control system.

AIRPLANE CONFIGURATION

Revised configuration, including elongated nose and yellow canary. Test with standard cam installed in Shelley Unit.

PROCEDURE

A. Elevator to Trim Test

1. From M = 0.8 and 45,000 feet, increase speed to $M \cong 1.0$ in straight and level 1.0g flight. This is to check any change in trim resulting from the new configuration changes.

B. Hysteresis Ground Test

1. Hysteresis data for elevator, aileron and rudder system should be obtained, and the lag in the trim system determined. The recommended procedure is outlined in DC-8-196, which was distributed to EAFB.

C. Shelley Unit Evaluation (Use Standard Cam)

1. For the Shelley Unit evaluation, data is required covering the variation of Shelley Unit "Q" pressure with Mach No. The response characteristics of the Unit should also be evaluated. Data shall be obtained over a large Mach No. range in conjunction with other flights. Special flights are not required.

2. To complete the data for the design of the Shelley Unit cam, flights shall be conducted as follows, for determination of "Elevator and Stick Force per G".

Approx. Vertical Load Factor Held "Steady State" Approx. 5 Seconds	Approx. Altitude (Minimum Variation During "Steady State")	Mach. No. Hold Constant During "Steady State"
2g, 3g	45,000	.80, .90, .95, 1.00
2g, 3g	35,000	.65, .80, .90, .95, 1.00, 1.20
2g, 3g	25,000	.90, .95, 1.00

C. Continued

NOTES:

a. Establish 1.0g trim for test Mach No. and do not operate trim during maneuver.

b. Pitch damper should be disengaged.

D. Surface-Stick Position, and Operating Characteristics During 1 to 2 Minutes Operation at Constant Altitude and Airspeed for Flight Conditions of Critical Longitudinal Sensitivity.

1. Pilot should search for and hold the critical condition, and shall establish the time where sensitivity commences. The records will be inspected to determine, if possible, the source of the previously reported sensitivity problem.

MEASUREMENTS REQUIRED

Basic flight control measurements, including "Q" pressures.

Requested by Dynamics

Approved by D. H. Bennett
for E. D. Shannon
Chief of Experimental Flight

EDS:JWS:gh

Report No. ZC-8-103
Addendum 15-A
Date - 19 July 1954

"A" Revision:
Added Two (2) Flts. with blanket and extended shroud.

CONVAIR
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(San Diego)

FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No: YF102 S/N52-7995

ITEM DESCRIPTION:

Temperature of engine outer wall and nozzle actuators.

PURPOSE:

To determine the temperature of the engine outer wall and the nozzle actuators in order to ascertain the amount of insulation necessary to relieve the heat damage to the nozzle actuators.

PROCEDURE:

Record temperatures continuously from take-off to service ceiling with airplane at maximum power plus afterburner.

Reference AVO dated 7-13-54 signed by F.A. Stephenson.
See following table for airplane use.

Airplane S/N	No. Flts. Req'd.	Configuration		
		Shields	Blanket	Blanket & Extended Shroud
52-7995	2	X		
52-7995	2		X	
52-7995	2			X

As requested by The AF.

Approved by
T. L. Maloy

"A" Revision

C O N V A I R
A Division of General Dynamics Corporation
(San Diego)

XF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Date:	21 July 1954	Report No.:	ZC-8-103
Airplane No.:	52-7995	Addendum No.:	16-A

ITEM DESCRIPTION: J57-P-11 Compressor Stall Test

PURPOSE:

To isolate the conditions resulting in compressor stall characteristics as have been experienced in recent flights at EAFB.

SCHEDULE:

Requested target completion date: 13 August 1954.

PROCEDURE:

Perform one or more flights as required to obtain the reported engine pulsation and stall conditions. Operations shall include the following:

(1) Steady state (power lever fixed) flights with climb to altitude where most pulsations occur. Obtain continuous data recordings from 20,000 feet and up. More than one pulsation should be recorded at each time, I.E., do not retard power lever immediately after first occurrance, but continue to permit engine to pulsate for at least three times.

(2) Perform what the pilot considers a normal engine acceleration (transient condition) at the following altitudes:

a. Maximum attainable altitude
b. 40,000 feet
c. 35,000 feet
d. 30,000 feet

MEASUREMENTS REQUIRED:

(a) Altitude
(b) Airspeed
(c) O.A.T.
(d) T_{T2} (Total inlet temperature)
(e) T_{T7} (Turbine discharge temperature)
(f) N_1 (RPM)
(g) N_2 (RPM)

"A" Revision

1. Perform several turn and bank maneuvers during the flights of this Addendum.
REASON: USAF request in order to complete the inlet duct pressure survey initiated by Addendum 2.

2. Added measurements as shown on Sheet 2.

3. Perform tests with both old and new type fuel control units.
Old unit part No. 190075-13
New unit part No. 190255-11

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XF-1[illegible]
FLIGHT TEST PROGRAM ADDENDUM [illegible]

MEASUREMENTS REQUIRED: (Continued)

°(h) P_{T_2} (Total inlet pressure) Use a minimum of 4 (four) rakes each containing 4 (four) pressure pickups. A minimum of one rake to have dynamic pickups with oscillograph recording in addition to photo panel recordings.
NOTE: Engine P_{T2} instrumentation probes may be used on an interim basis, if necessary, pending completion of rakes.
(i) One total pressure pickup in each intake duct upstream of splitter.
°(j) P_{T4} (Compressor discharge total pressure).
°(k) P_{S4} (A/B meter signal pressure).
°(l) $P_{S4.5}$ (Burner pressure).
°(m) P_{T7} (One position minimum on oscillograph)
(n) P_{T0} (Airspeed total pressure)
(o) Main engine fuel flow.
(p) A/B fuel flow
°(q) A/B fuel manifold pressure.
°(r) Primary fuel pressure at pressurizing valve.
°(s) Secondary fuel pressure at pressurizing valve
°(t) Eng fuel pump inlet pressure
(u) Compressor bleed valve positions.
(v) A/B nozzle position.
(w) PLA (power lever angle)
°(x) Holley bleed valve governor, case temp
°(y) Fuel temp. between fuel pump exit and fuel control.

° Denotes items which should be recorded by oscillographs for more accurate study of sequencing of events and dynamic characteristics.

REFERENCE:

Instrument Requirement No. 82-23, 82-24, 82-25, 82-28.

Requested by: Engineering Power Plant Design
Group Memo dated 13 July 1954

Approved by [illegible]
[illegible]. L. Maloy
Group Engineer
Experimental Flight

TLM/JS:gh

Add the Following to Measurement Requirements:

°1. P_{T3} (P&W Kit)
°2. Holley governor pressure bias
3. Holley governor temp. bias
4. Fuel pump inlet fuel temp.
5. Static press. in vicinity of fuel control
6. Fuel inlet press. (piezometer)

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CONVAIR
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F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Date: 4 August 1954

Report No. ZC-8-103
Addendum No. 17

Airplane No.: S/N 52-7995

Item Description:

Engine Lubricating Oil Temperatures with and without Air-Oil Cooler.

Purpose:

To determine whether or not the engine lubricating oil can be cooled adequately with only the fuel-oil cooler.

Configuration:

Completely by-pass air-oil cooler.

Procedure:

Measure the following oil and fuel temperatures throughout a typical high altitude flight.

Measurements Required:

1. Temperature of engine lubricating oil from engine.
2. Temperature of engine lubricating oil into reservoir.
3. Temperature of engine lubricating oil into engine.
4. Temperature of fuel out of cooler.
5. Temperature of fuel out of pumps.
6. Temperature of fuel in tank.
7. Engine fuel flow to nozzles.

NOTE: Fuel to eng. should not exceed 225°F as it tends to "coke" system. Advancing throttle and decreasing altitude should lower temp. Pilot indicator needed.

Requested by Thermo Group

Approved by
For T. L. Maley

TLM:NR:gh

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CONVAIR
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F-102

FLIGHT TEST PROGRAM ADDENDUM ITEM

Date: 4 August 1954

Report No. ZC-8-103
Addendum No. 18

Airplane No: 52-7995

Item Description:

Aft section cooling test.

Purpose:

To investigate the effects of the distribution duct and auxiliary air-oil cooler on after section cooling air flow.

Configurations to be Tested:

1. Manifold louvers and oil cooler "open".
2. Manifold louvers "open" and oil cooler "closed".
3. Manifold louvers "closed" and oil cooler "open". (Oil cooler shutter controllable from pilot's compartment, louvers to be closed by clamps lined with asbestos for ease of removal and duct to be plugged with stopper that does not damage duct.)

Procedure:

1. Record pressure, temperature, altitude and Mach number for an entire typical flight -- Maximum power climb to ceiling, one minute at maximum power level flight, and dives at speeds of Mach 1.1 to Mach 1.3.
2. Repeat for configurations (2) and (3).

Measurements Required:

1. P_{static} and P_{total} at oil cooler air scoop.
2. P_{static} at station 480 between skin and shroud.
3. P_{static} at station 500 between shroud and engine.
4. P_{total}, back to back at three positions, station 621.5.
5. Record aft end temperatures where instrumented (structure, actuator, shroud and air.)

Requested by Thermo Group

Approved by
T. L. Maloy

TLM:DR:gh

CONVAIR
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(San Diego)

F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 19

Item Description: Instrument Panel Vibration Test

Purpose: To determine compliance with Par. 3.2.8 of MIL-I-5997 (USAF) which establishes vibration requirements for the instrument panel to prevent sticking of instruments.

Procedure:

1. Paragraph 3.2.8 of MIL-I-5997 reads as follows:

   " The amplitude of vibration of the instrument board shall be between the limits of 0.003 and 0.005 inch in at least one direction, and shall not exceed 0.005 inch in any direction with the airplane in normal flight. NOTE: An auxiliary panel vibrator or tapper shall be provided if necessary to increase the vibration to 0.003 inch in at least one direction."

2. Instrumentation shall be installed to record the vibration in the longitudinal, vertical, and transverse directions. Pickups should be installed on both the center and R.H. portions of the instrument panel.

3. No special flights are required. The required data may be obtained concurrently with other flights covering a normal range of flight conditions

4. If the vibration is less than .003 inch, a panel vibrator shall be installed, and the tests repeated.

Measurements Required: Record the following:

1. Airspeed
2. Altitude
3. N_2, high rotor speed
4. Vibratory amplitude (3 axes). Two panel positions. Instr. Req. 82-27 (Ref.).
5. Vibratory frequency (3 axes). Two panel positions. This item desirable but not mandatory. Instr. Req. 82-27 (Ref.).

NOTES:

1. Schedule: Soon as possible
2. Requested by AF Ltr. dated 1 June 54. Copy of results requested by Model 8 Proj. Off. AVO dated 20 June 54, from C. B. King and W. W. Fox (cc:J.C.Hoover)

Approved by
H. K. Cheney
Chief Flight Test Engineer
Experimental Flight

HKC:JHS:gh

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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 20

Item Description: Airspeed Calibration

Purpose:

To obtain data for airspeed calibration by three separate procedures. This new calibration is necessitated by drag reduction modifications of the aircraft.

Airplane Configuration:

Airplane modified by sharp nose, yellow canary, and items listed in minutes - Post Flight Meeting, 22 July, 1954.

Procedure A:

Pacer calibration of total and static pressure indication at subsonic speeds using USAF F-86E calibrated pacer aircraft.

1. The following tests should be performed at 15,000 feet altitude.

Configuration	Speed Range (IAS)	Speed Increments
Clean	150-500 MPH	20 MPH
Gear Extended	150-275 MPH	10 MPH

2. For comparison of data, it is suggested that some of the above tests be performed using the same pacer as was used for the previous calibration (F86, Ser. No. 582)

Procedure B:

Static pressure calibration for transonic speeds using three static orifices as reference during performance of the specified test flights.

1. Install 3 fuselage static pressure sources on the fuselage in accordance with drawing 8-00881. Connect each static port to a separate altimeter on the photo panel. Connect the middle static port also to a separate altimeter on the pilot panel.

2. Perform the following flight plan for obtaining the required calibration data for transonic and supersonic calibration:

   a. Climb to about 43,000 feet altitude and dive with a sufficiently large dive angle to reach a maximu Mach No. of about 1.2 during the dive.

   b. Pull out at about 34,000 feet altitude into level flight.

Procedure B: - Continued

2. Continued

   c. Maintain constant altitude using the altitude indicated by the middle static port on the fuselage nose.

   d. During this level flight decelerate slowly to an indicated Mach No. of $M_1 = 0.80$ or less (approximate time to decelerate from $M_1 = 1.08$ to $M_1 = 0.80$ may be 30 seconds)

   e. Record the photopanel data on high speed motion picture camera in the Mach number range between $M_1 = 1.08$ and $M_1 = 0.80$ or less.

   f. Repeat the above procedure with pull out at about 27,000 foot altitude.

3. Evaluation of the data should be based on the method described in Report ZC-8-501-SR2 (ref.).

Procedure C:

Static pressure calibration at subsonic, transonic, and supersonic speeds using radar calibration during level flight decelerations and a dive to supersonic speeds.

1. The following flight plan shall be accomplished, and the flight pattern shall be in accordance with Figure 1:

   a. Climb to altitude in the most convenient manner and do not require radar tracking during the climb. This permits the pilot to make an optimum climb without attempting to navigate within radar range.

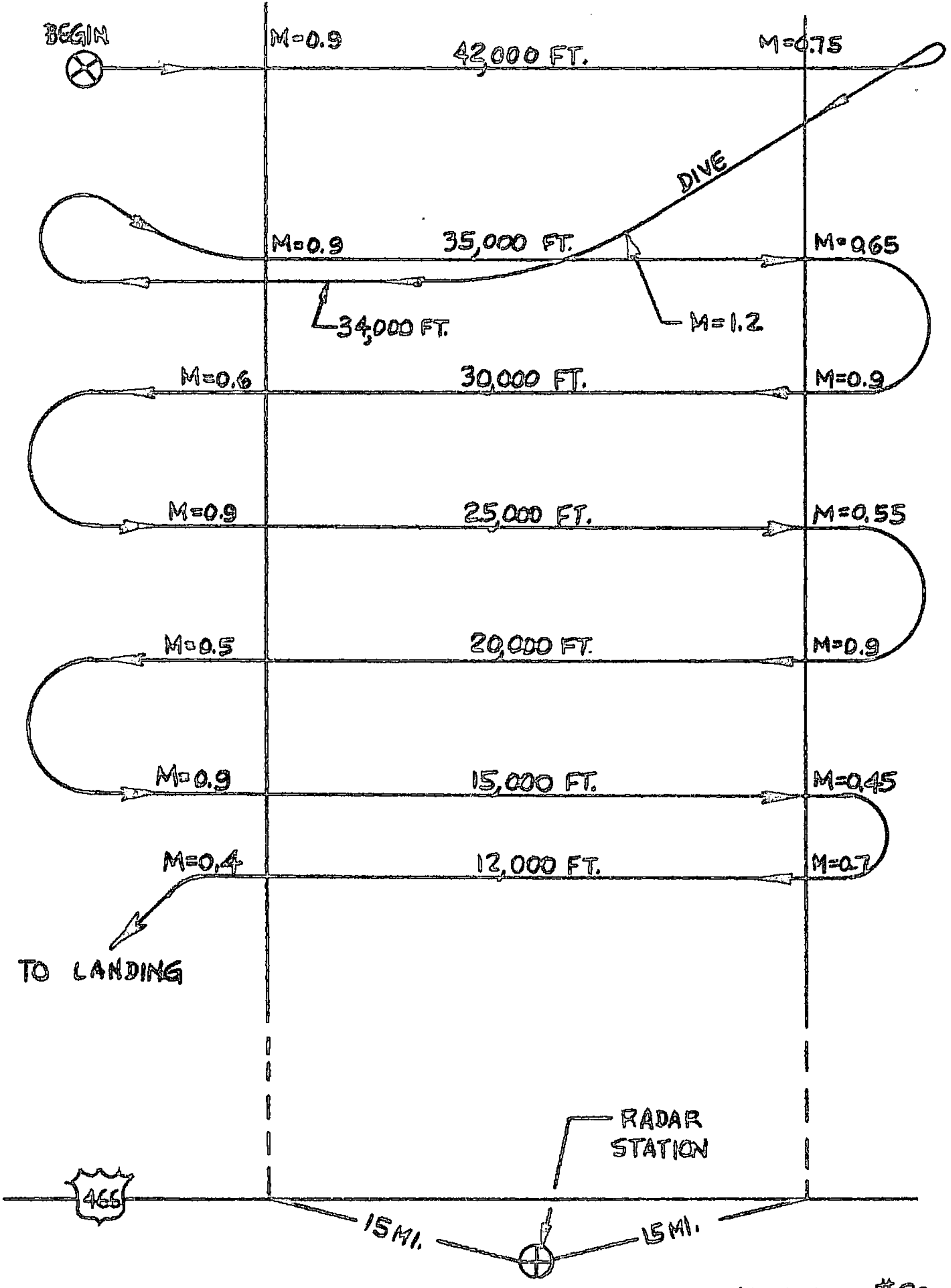
   b. Obtain all radar tracking data in a sequence of passes above highway No. 466 and within 15 miles of the radar installation, as illustrated by Figure 1.

Measurements Required:

Same as for previous calibration tests, except install additional static pressure sources and altimeters as described in paragraph 1 of Procedure B.

Approved by H K Cheney
H. K. Cheney
Chief Flight Test Engineer
Experimental Flight

HKC:JIS:gh



ADDENDUM #20

FIGURE 1

CONVAIR
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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 21

Item Description: Drag Characteristics

Purpose:

To establish a procedure and desirable sequence of operations for the obtainment of pertinent drag and calibration data.

Airplane Configuration:

Airplane modified by sharp nose, yellow canary, and items listed in minutes of Post Flight Meeting, 22 July 1954.

Procedure:

1. First priority is to obtain supersonic drag characteristics. The data required will be obtained during a series of dives. The procedure of these dives is: a) Climb to approximately 43,000 foot altitude and dive at a sufficiently large dive angle to reach a maximum Mach No. of about 1.2 during the dive, b) Pull out at specified altitude to a level flight condition, c) Maintain constant altitude using the altitude indicated by the middle static port on the fuselage nose, d) During this level flight region, decelerate to subsonic speeds approaching $M_i = 0.80$.

   Three dives are requested. The pullout altitudes are 34,000 feet and 27,000 feet and approximately 20,000 feet. The first two dives are as requested under procedure B of Addendum 20.

2. Second priority is to determine the subsonic drag level. This is best accomplished by low altitude flight (15,000 feet) in the region of $M = 0.60$ to $M = 0.85$ (W/δ = 46,000Lb. or less) in increments of $\Delta M = 0.05$.

   The evaluation of these data will be based on pacer type airspeed calibration which must be obtained concurrently or prior to this schedule.

3. Third priority is evaluation of the drag rise region. This requires stabilized data from maximum level flight Mach numbers to approximately $M = 0.85$ in increments of $\Delta M = 0.025$. Such data should be obtained from flight at three altitudes, preferably 15,000, 25,000 and 35,000 feet. (Corresponding W/δ values are 46,000; 78,000 and 110,600 lbs.)

Procedure: - Continued

4. Fourth priority is to fly progressive stabilized speed - power data at approximately 35,000 feet from Vmax to $M = 0.55$ in increments of $\Delta M = 0.05$ ($W/\delta \cong 110{,}600$). In addition two maneuvering flight conditions are requested at 1.6 g's for: a) $M = 0.70$ and b) $M = 0.80$. The altitude for these turns is 25,000 feet at a gross weight of 26,000 lbs. ($n\,W/\delta \cong 112{,}000$).

Measurements Required:

Refer to Experimental Flight Memo EF-8-257, subject "Measurements Required for First Flights After Modification of S/N 7995".

Symbols:

n = Maneuvering load factor, g's
W = Gross weight, lbs.
δ = Ambient pressure ratio

Reference:

1. Addendum 9-A, "Yellow Canary Evaluation".

2. Addendum 11, "Drag Data Test".

3. Addendum 20, "Airspeed Calibration".

Approved by T. L. Maloy
T. L. Maloy
Group Engineer
Experimental Flight

TLM:JMS sgh

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F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 22

Item Description: Vibration and Acceleration Characteristics

PURPOSE

Vibration and acceleration data is necessary to establish design factors based on actual tests, for the Hughes MG-3 system weight saving study.

AIRPLANE CONFIGURATION

Elongated nose.

PROCEDURE

1. Data shall be recorded over a full range of flight conditions, until a satisfactory pattern is established, with special emphasis on the analysis of conditions such as buffeting, gusts, landing and taxi.

2. At least one landing should be made with as much impact as pilot feels is reasonably safe.

3. Special flights shall be made only in cases where a study of the data indicates that more exploration is necessary in certain areas.

MEASUREMENTS REQUIRED

1. Acceleration, vertical and lateral, at approx. Sta. 47. Analysis to include double amplitude and frequency.

2. Vertical acceleration at C.G.

3. Basic flight control measurements.

NOTES

1. Instrument Requirements No. 82-30.

Requested by Project Office
F. B. Wozniak

Approved by T. L. Maloy
T. L. Maloy

TLM:JS:gh

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Date: 1 Oct., 1954

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A Division of General Dynamics Corporation
(San Diego)

F-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. 2C-8-103
Addendum No. 23

ITEM DESCRIPTION: Cable Tension Values

PURPOSE:

To obtain control cable tension values from tests for design purposes.

PROCEDURE:

Elevator, Aileron and rudder cable tension variations versus time should be recorded at the following conditions.

1. A flight at high altitude and minimum power setting to obtain maximum cold conditions.
2. A flight at 20,000 feet and maximum Mach number with afterburner to obtain a high heat condition. (Level Flight)

MEASUREMENTS REQUIRED:

1. Rudder cable tension.
2. Elevator cable tension.
3. Aileron cable tension.

NOTES:

1. Instrument Requirement No. 82-29 (ref.).
2. Install during the next engine change. Instruments should be removed for use on AF53-1780 after the flight plan of this addendum has been accomplished.

Requested by Controls Group

Approved by D. H. Bennett
for E. L. Shannon
Chief of Experimental Flight

ELS:JFS:gh

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(San Diego)

YF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Airplane No.: 52-7995

Report No. ZC-8-103
Addendum No. 24

ITEM DESCRIPTION: Buffet characteristics and Wing Fence Evaluation.

PURPOSE: To determine buffet characteristics for various airplane configurations, and to determine the possibility of weight saving through elimination or reduction of wing fences.

AIRPLANE CONFIGURATION:

A. Install dummy rudder actuator fairing (Accomplished)

B. Install dummy fin tip antenna.

C. Remove all wing fences.

PROCEDURE:

1. Conduct flights at altitudes of 15,000 to 35,000 feet in the speed range from M = .7 to M = 1.0 to establish buffet boundary patterns. Separate and successive patterns shall be established for each of the above configurations, for comparison purposes.

2. An attempt should be made to obtain buffeting by dumping cabin pressure, and the counter number noted. Airplane configuration is optional.

3. The buffet boundary should be established for making coarse corrections and fine corrections while tracking at high altitude. It is suggested that this be obtained as a by-product of tracking flights for other purposes.

4. Stick Position/g shall be obtained after removal of the wing fences (configuration C) for normal load factors of 2g and 3g. This is in addition to the buffet pattern, which is ordinarily based on smaller load factors. The wing fences should be restored or altered if necessary after aerodynamic analysis.

5. Directional characteristics tests in accordance with Addendum 25 should be performed after removal of the wing fences.

MEASUREMENTS REQUIRED:

1. Elevon Surface Position
2. Rudder Surface Position
3. Longitudinal Stick Position
4. Vertical Velocity Pickup, Sta. 532, BL 174.
5. Lateral Velocity Pickup, Sta. 566, WL 101.3
6. Normal and Lateral Acceleration at C.G.
7. Vertical and lateral acceleration at Sta. 47.
8. Yaw rate Gyro output.
9. Nose boom yaw angle.

Approved by D. W. Bennett
for E. D. Shannon
Chief of Experimental Flight

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EDS:JNS:gh
Requested by:
Dynamics & Aero

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CONVAIR
A Division of General Dynamics Corporation
(San Diego)
YF-102
FLIGHT TEST PROGRAM ADDENDUM ITEM

Sheet 1 of 2

28 October 1954
Report No. ZC-8-103
Addendum No. 25-A

Airplane No.: 52-7995

ITEM DESCRIPTION: Adverse yaw and directional characteristics.

PURPOSE:

1. Adverse yaw tests are required to determine the effect of changes in yaw damper gyro orientation upon gyro output, during rolling maneuvers. Adverse yaw characteristics have been experienced during aileron alone rolls with damper on.

2. Directional characteristics tests are to determine the effect of the rudder fairing on directional trim. Present experience indicates that approximately 1° of left rudder trim is required throughout the normal speed range up to M = .93. At M = .93 a sudden and undesirable directional trim change occurs.

AIRPLANE CONFIGURATION: Noted Below.

PROCEDURE A (Adverse Yaw):

Record yaw conditions during rolls with aileron alone at M = .9 and 35,000 feet, with and without damper, and acquire a roll rate of at least 40°/second, for each of the following gyro configurations:

a. Tilt gyro down 3 to 5 degrees, so that the yaw rate output causes rudder deflections which are less adverse during roll maneuvers.

b. Return gyro to normal, except reverse the normal direction of rotation (See Note 1).

c. Restore normal direction of rotation of gyro.

PROCEDURE B (Directional Characteristics):

1. Install a dummy actuator fairing on the opposite side of the fin from the actuator fairing, to provide symmetry.

2. Test trim characteristics through the normal speed range, including Vmax at 15,000 feet.

"A" Revision, 3 Nov. 1954:
Record pressures on both R.H. and L.H. side of afterbody fillet (Instrument Requirements 82-32)
REASON: To determine whether or not the afterbody pressures are symmetrical, as this may affect directional characteristics.

MEASUREMENTS REQUIRED:

1. Airspeed.

2. Pressure Altitude.

3. Outside Air Temperature.

4. Rudder Surface Position

5. R.H. Elevon Surface position.

Approved by:
D. H. Bennett
for E. D. Shannon
Chief of Experimental Flight

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MEASUREMENTS REQUIRED - Continued

6. L.H. Elevon surface position.

7. Nose boom yaw angle.

8. Yaw rate gyro output.

9. Roll rate gyro output.

10. Normal acceleration at C.G.

NOTES:

1. Sensing gyro of Par. A1 may be reversed by either rotating the gyro case 180° or by reversing the direction of rotation of the gyro. If gyro rotation is reversed the input voltage leads to the sensing potentiometer must also be reversed.

Requested by Dynamics

Approved by
for E. D. Shannon
Chief of Experimental Flight

EDS:JNS:gh

S/N 52-7995 F-102 - ZC-8-103

Date Prepared: 21 Oct 1955

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TITLE FLIGHT TEST ITEM AND MEMO NUMBERS	Future	Completed	Cancelled	Transferred	REMARKS: Including Memos and Reports Written on Flight Test Items.
1. STATIC THRUST CALIB		X			ZC-8-501-1 Flight Operations Report
2. BRAKE OPERATION EVALUATION		X			ZC-8-501-1,5,6,15,18 Flt. Oper. Rpt.
3. TAXI TESTS		X			ZC-8-501-1, 8, 9, Flt. Oper. Rpt. EF-8-313 Nose Landing Gear Loads and Stresses
4. ENGINE CONTROL OPERATION		X			ZC-8-501-9, 18 Flt. Oper. Rpt.
5. SHAKEDOWN FLIGHTS		X			ZC-8-501-2 Flt. Oper. Rpt.
6. SPEED BRAKE OPERATION		X			ZC-8-501-2 Flt. Oper. Rpt.
7. FUEL TANK PRESSURIZATION		X			ZC-8-501-4 Flt. Oper. Rpt.
8. ENGINE EQUIPMENT OPERATION		X			ZC-8-501-2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 13, 14, 15, 16, 17, 18, 19 & 20 Flt. Oper. Rp EF-8-258 A/B Roughness
9. FLUTTER AND VIBRATION		X			ZC-8-501-2, 3, 4, 5, 6, 7, 9, 10, 16, 17 and 20 Flt. Oper. Rpts. ZU-8-013 Results of Flight Flutter Tests
a. EF-8-560 Noise Level Measurement		X			
b. EF-8-591 Cockpit Noise and Vibration		X			ZC-8-501-19 and 20 Flt. Oper. Rpts.
10. AIRSPEED CALIBRATION		X			ZC-8-501-3, 4, 6, 8, 10, 12, 18 and 20 Flt. Oper. Rpts. EF-8-501-SR-2 Summary of Airspeed Calibration Analysis

FLIGHT TEST ITEM AND MEMO NUMBERS
(Continued)

	Future	Completed	Cancelled	Transferred	REMARKS:
11. PRELIMINARY FLIGHT CHARACTERISTICS		X			EF-8-442 Longitudinal Stick Force EF-8-614 Dynamic Stability During Rolls EF-8-416 Vertical Acceleration Activity
a. EF-8-520 Airplane Response to High Roll Rate Maneuvers		X			EF-8-558 Roll Rate versus Aileron Angle EF-8-618 Flight Results During Aileron Rolls
12. ANGLE OF ATTACK AND YAW CALIBRATIONS				X	Transferred to S/N 53-1780
13. PRELIMINARY AIRPLANE PERFORMANCE		X			ZC-8-501-7, 9, 10 & 21 Flt. Oper. Rpts. EF-8-429 Climb Performance EF-8-354 Climb Performance
a. EF-8-475 Wing Tip Structual Load		X			EF-8-522 Wing Tip Load Variation with Mach Number and Altitude EF-8-472 Wing Tip Strain Gage Data
14. USAF PHASE II		X			USAF Phase II Evaluation Report YF-102 S/N 52-7995
15. MISSILE BAY DOOR OPERATION				X	Transferred to S/N 53-1781
16. AIRPLANE PERFORMANCE		X			EF-8-402 Take-off Performance EF-8-245 Take-off Characteristics
a. EF-8-595 Landing Leads and Distances		X			ZC-8-501-2, 3, 8 & 12 Flt. Oper. Rpts.
b. EF-8-597 Wing Tip - Cambered Not Reflexed		X			EF-8-274 & 274A Drag Polars EF-8-353 Drag Evaluation
17. ENGINE AND STRUCTUAL COOLING		X			ZC-8-501-2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 13, 14, 15, 18, 19 & 20 Flt. Oper. Rpts. EF-8-403 Cooling System Analysis EF-8-334 Results of A/B Blanket Installation Cooling System Operation in YF-102 Airplane

FLIGHT TEST ITEM AND MEMO NUMBERS
(Continued)

		Future	Completed	Cancelled	Transferred	REMARKS:
18.	DYNAMIC RESPONSE				X	Transferred to S/N 53-1780
19.	STABILITY AND CONTROL-SPECIFICATION COMPLIANCE				X	Transferred to S/N 53-1791
20.	PILOT ASSIST SYSTEM DEVELOPMENT				X	Transferred to S/N 53-1780
21.	ENGINE OPERATION - AIR STARTS				X	Transferred to S/N 53-1781
22.	DROGUE CHUTE TESTS		X			ZC-8-501-10, 18 & 20 Flt. Oper. Rpts. EF-8-276 Speed Brake Effectiveness
23.	EMERGENCY FUEL CONTROL SYSTEM			X		
24.	RAM AIR TURBINE TESTS				X	Transferred to S/N 53-1781 Addendum 12B Reinstates this test to S/N 52-7995
25.	FLIGHT CONTROL SYSTEM OPERATION		X			ZC-8-501-8, 9, 12, 16, 17, 19, 21, 22, 23 Flt. Oper. Rpts. ZC-8-512-5, 6 & 26 Flt. Oper. Rpts. EF-8-323 Control Stick Travel EF-8-420 Shelly Unit with Modified Cam EF-8-287 High Longitudinal Stick Force EF-8-608 Transient Longitudinal Oscillograph EF-8-617 Longitudinal Stick Force ZC-8-519 Flight Control System
	a. ZC-8-457 Elevon System Force Link		X			ZC-8-501-2 Flt. Oper. Rpt.
26.	ENGINE AND ASSOCIATED EQUIPMENT PHASE III		X			Operation of the J-57-P11 installed in YF-102 and F-102
	a. ZC-8-527 Ground Test Combustion Starter		X			ZC-8-512-15 Flt. Oper. Rpt.

FLIGHT TEST ADDENDUM AND MEMO NUMBERS

		Future	Completed	Cancelled	Transferred	REMARKS:
1.	LANDING GEAR HYDRAULIC PRESSURE		X			ZC-501-4, 12 Flt. Oper. Rpts.
2.	INLET DUCT PRESSURES				X	Superceded by Addendum 16A
3.	DUMMY FUEL TANK VENT PRESSURE			X		
4.	INLET DUCT STRESSES			X		EF-8-234 Engine Inlet Duct Pressures and Stresses
5.	DYNAMIC DIRECTIONAL STABILITY		X			ZC-8-501-10, 15, 16 & 17 Flt. Oper. Rpts.
6.	AIRSPEED CALIBRATION - TRANSONIC				X	Superceded by Addendum 20
7A.	PRIMARY HYDRAULIC SYSTEM OIL TEMPERATURE - WITH AND WITHOUT COOLER		X			ZC-501-18 & 19 Flt. Oper. Rpts. ZC-8-516 Hydraulic System Oil Temperature
8.	PRESSURES ON LONG NOSE		X			ZC-8-512-2 Flt. Oper. Rpt. EF-8-626 Elongated Nose Pressure Survey
	a. EF-8-592 Radome Glycol Dispersion		X			ZC-8-512-15, 16 & 29 Flt. Oper. Rpts. EF-8-637 Radome Glycol Anti-icing System
9B.	YELLOW CANARY PRESSURES		X			ZC-8-501-18 & 20 Flt. Oper. Rpts. EF-8-574 Evaluation of Afterbody Pressure Data
10A.	BLEED AIR CONSUMPTION EFFECT		X			EF-8-247 Thrust Loss Due to Engine Air Bleed EF-8-501-15 Flt. Oper. Rpt.
11.	DRAG EFFECTS IN REGION OF MACH 0.9		X			ZC-8-501-21 Flt. Oper. Rpt.
12B.	RAM AIR TURBINE TESTS		X			ZC-8-512-10 Flt. Oper. Rpt.
13A.	SIMULATED LANDINGS WITH RAM AIR TURBINE				X	Superceded by Addendum 12B
14A.	LONGITUDINAL SENSITIVITY		X			ZC-8-501-21 Flt. Oper. Rpt. ZF-8-537 Ground Hysteresis Test
	a.		X			ZC-8-501-24 Flt. Oper. Rpt.

FLIGHT TEST ADDENDUM AND MEMO NUMBERS
(Continued)

		Future	Completed	Cancelled	Transferred	REMARKS:
15A.	ENGINE OUTER WALL AND NOZZLE ACTUATOR TEMPERATURES		X			ZC-8-501-12, 13, 14, 15, 18 & 20 Flt. Oper. Rpts.
16A.	COMPRESSOR STALL TEST		X			ZC-8-501-14, 20, 21 & 22 Flt. Oper. Rpts. EF-8-513 Compressor Stall Test EF-8-215 Occurrence of Compressor Surge
17.	ENGINE OIL TEMPERATURES WITH AND WITHOUT AIR-OIL COOLER		X			ZC-8-501-8 and 22, Flt. Oper. Rpts. ZC-8-512-4 and 21, Flt. Oper. Rpts.
18.	AFT SECTION COOLING		X			ZC-8-501-19 Flt. Oper. Rpt.
19.	INSTRUMENT PANEL VIBRATION				X	Transferred to S/N 53-1791.
20.	AIRSPEED CALIBRATION		X			ZC-8-501-18 and 20 Flt. Oper. Rpts. EF-8-318 Airspeed Calibration Sharp Nose Aircraft
	a. EF-8-428 Relocation of "Q" Pot. Static Source		X			EF-8-642 Shelly Unit Static Pressure Source Evaluation
21.	DRAG CHARACTERISTICS		X			ZC-8-501-16, 18, 19, 20, & 21 Flt. Oper. Rpts. ZC-8-512-27 Flt. Oper. Rpts. ZC-8-501-SR3
22.	VIBRATION AND ACCELERATION CHARACTERISTICS		X			ZC-8-501-22 Flt. Oper. Rpt.
23.	CABLE TENSION VALUES		X			ZC-8-512-5 Flt. Oper. Rpt.
24A.	BUFFET CHARACTERISTICS AND WING FENCE EVALUATION		X			ZC-8-501-20, 21, 22, and 23 Flt. Oper. Rpts.
	a. EF-8-562 Wing Fence Evaluation		X			ZC-8-512-10, 11, 13, 14, 15, 16 & 18 Flt. Oper. Rpt.
25A.	ADVERSE YAW AND DIRECTIONAL CHARACTERISTICS		X			ZC-8-512-4 Flt. Oper. Rpt.

ZC-8-504-4

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Page No.

Purpose 1

Summary of Results 1

Log of Operations 2

Description of Operations 2 & 3

Airplane Configuration 3

Test Results - Text

Flight Characteristics

General Flight & Handling Characteristics 4

Power Plant Operation

(Effect of Missile Exhaust Gases) 4

Equipment Operation

Landing Gear 4 & 5
Pressure Cabin 5
Armament 6

Airplane Pre-Flight Changes 7 & 8

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Photographs - Nose Landing Gear Assembly Failures 11
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Test Results - Figures

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I	Flight History - Flight No. 15	13
II	Missile Exhaust Dust Pattern - Flight No. 15	14
III	Landing Track Schematic - Flight No. 16	15

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The following changes were accomplished prior to flight release:

Pre-Flight No. 14

1. Engine No. P-420245 replaced with J-57 S/N P-420413 because of excessive vibrations during flight and ground checks.
2. Installed new pressurization valve, P/N 191346 on AJA2 fuel control.
3. Replaced instrumented ram air turbine hydraulic lines with original production lines.
4. Installed afterburner actuator shields on new engine.
5. Replaced G.T.C. check valve.
6. Replaced "Q" switch due to excessive amount of water being used during flight.
7. Added support to secondary hydraulic pump pressure line.
8. Replaced primary and secondary hydraulic check valves.
9. Added one (1) CVAC F1-157 filter between disconnect and hydraulic system per TVA 30191, also added check valve between the filters.
10. Replaced canopy actuator due to faulty operation.
11. Replaced ram air oil cooler seal, 8-27712.
12. Removed inert missiles from all three aft bays.
13. Removed following RAT instrumentation from airplane:

    a. Autosyn (output pressure from RAT)
    b. Autosyn (output pressure of flow control valve)
    c. CEC Transducers (output pressure from RAT)
    d. Statham transducer (reservoir to RAT pressure)
    e. Statham transducer (flow control valve to reservoir)
14. Removed three thermocouples and their matching indicator and harnesses from RAT system.

Pre-Flight No. 15

1. Replaced rudder actuator cylinder.
2. Replaced engine fuel dump valve.
3. Installed new aileron trim actuator.
4. Reworked nose weight bar to relieve interference with nose wheel tire in retracted position.
5. L/H elevon control valve stops adjusted to improve friction level for return of stick to neutral position.
6. Canopy L/H teleflex cable replaced.
7. 245.5 pounds ballast installed on aft electronic bay door per TVA 30145 on Drawing 8-70608.
8. Cabin pressure dump valve teleflex repaired.
9. Anti-icing duct cap installed per TVA 18601-7.
10. Dummy missile installed in R/H forward bay.
11. Replaced right [redacted].

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AIRPLANE PREFLIGHT CHANGES
(Continued)

Pre-Flight No. 16

1. UHF receiver overhauled to improve sensitivity.
2. 1/2 inch rubber cushion installed on bulkhead, station 403.8 to prevent damage from missile ignitor plug.
3. Replaced box assembly on canopy latch control 49-3.
4. Replaced ARC-34 receiver.

REFERENCES

No.	Title
1	Convair Report No. ZC-8-504-1, Flight Operations Report, dated 23 August 1954.
2	Convair Report No. ZC-8-116, Investigation of Nose Landing Gear Failure Incident on YF-102 No. 53-1781, dated 21 September 1954.
3	Convair Report No. ZC-8-504-2, Flight Operations Report, dated 28 September 1954.

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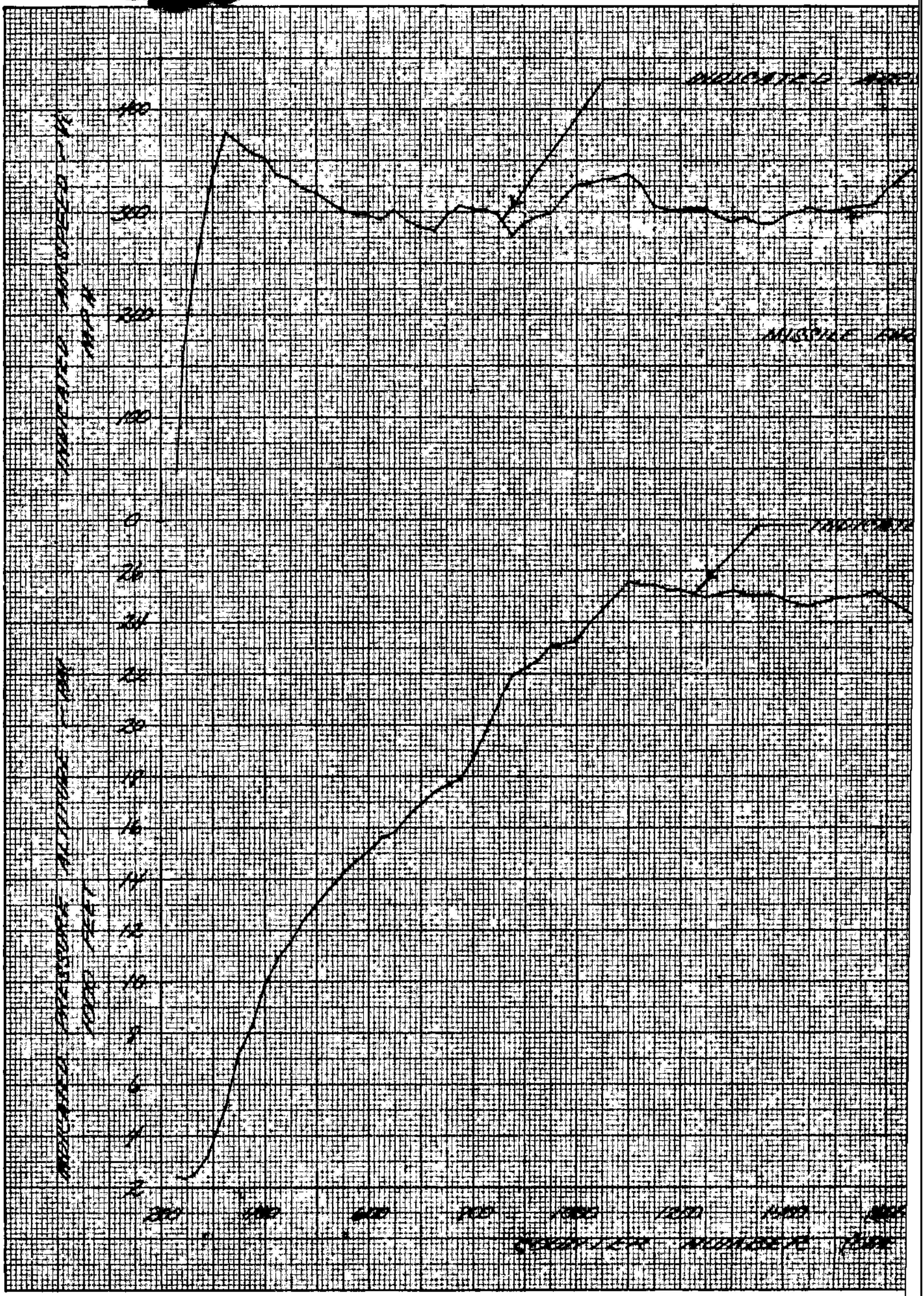

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10 X 10 TO THE CM. 359T-14LG
KEUFFEL & ESSER CO. MADE IN U.S.A.
ALBANENE ®

UNCLASSIFIED REPORT NO. ZC 8-504-4

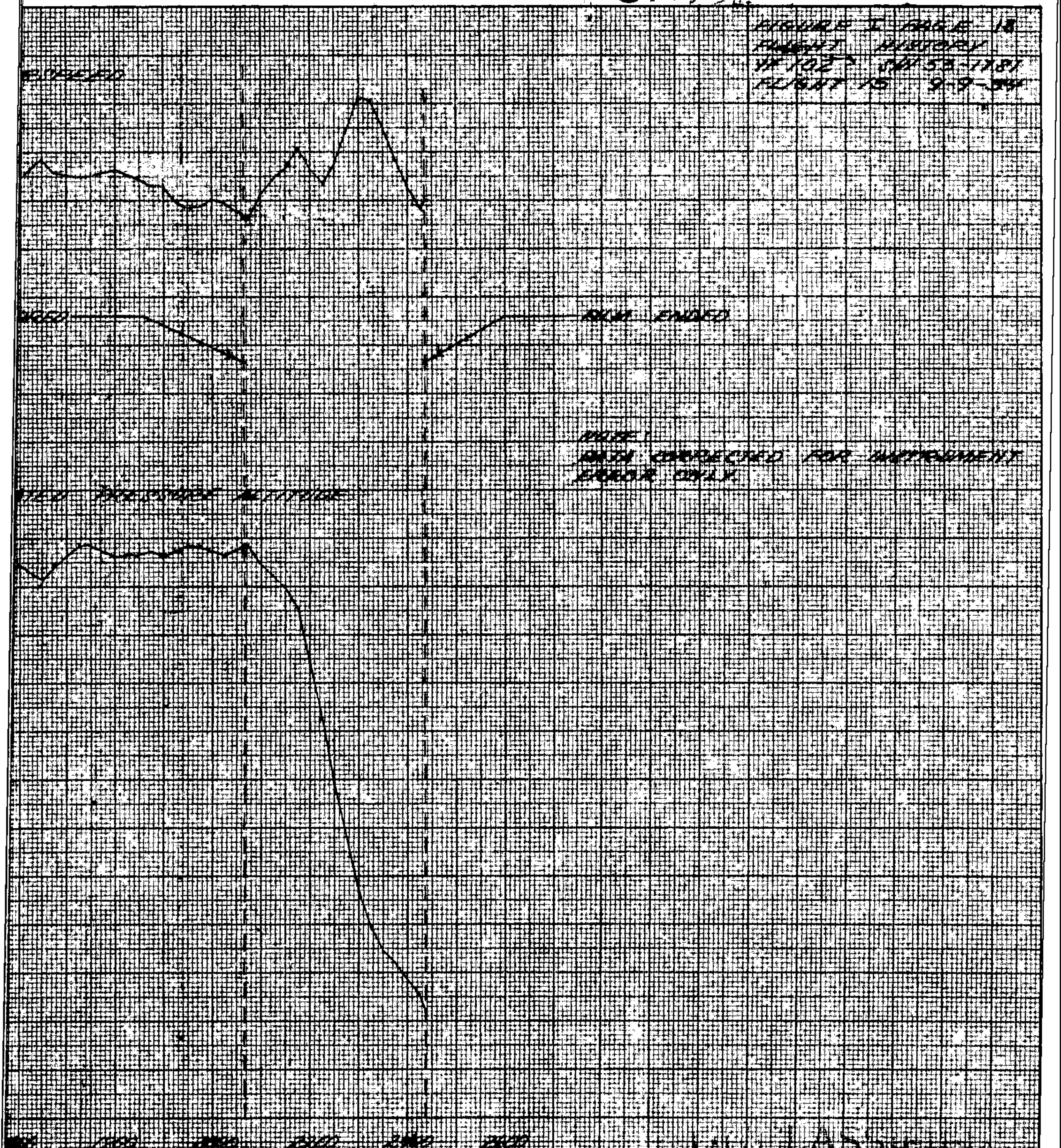

FIGURE I PAGE 18
FLIGHT HISTORY
YF-102 USAF 53-1787
FLIGHT 15 9-9-54

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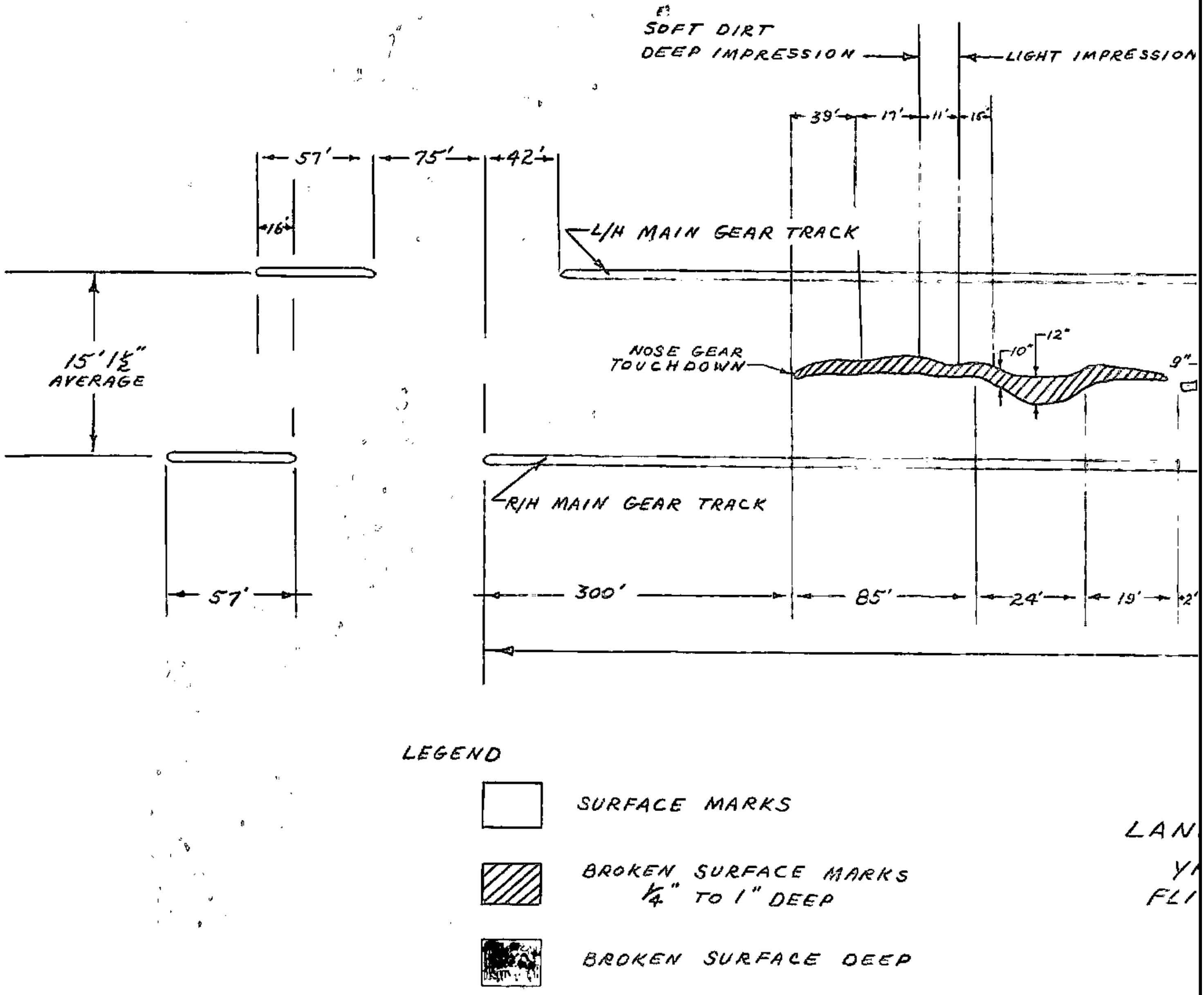
SOFT DIRT
DEEP IMPRESSION
LIGHT IMPRESSION
39'
17'
11'
15'
57'
75'
42'
16'
L/H MAIN GEAR TRACK
NOSE GEAR
TOUCHDOWN
10"
12"
9"
15' 1½"
AVERAGE
R/H MAIN GEAR TRACK
57'
300'
85'
24'
19'
2'
LEGEND
SURFACE MARKS
BROKEN SURFACE MARKS
¼" TO 1" DEEP
BROKEN SURFACE DEEP
LAN
Y
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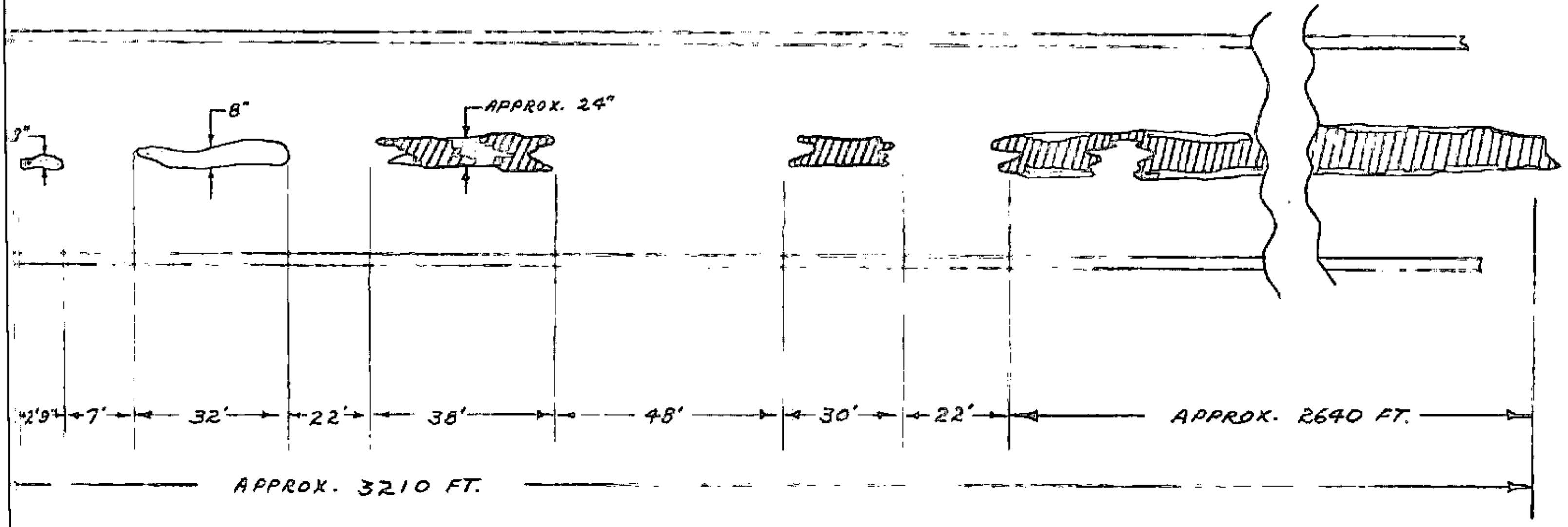
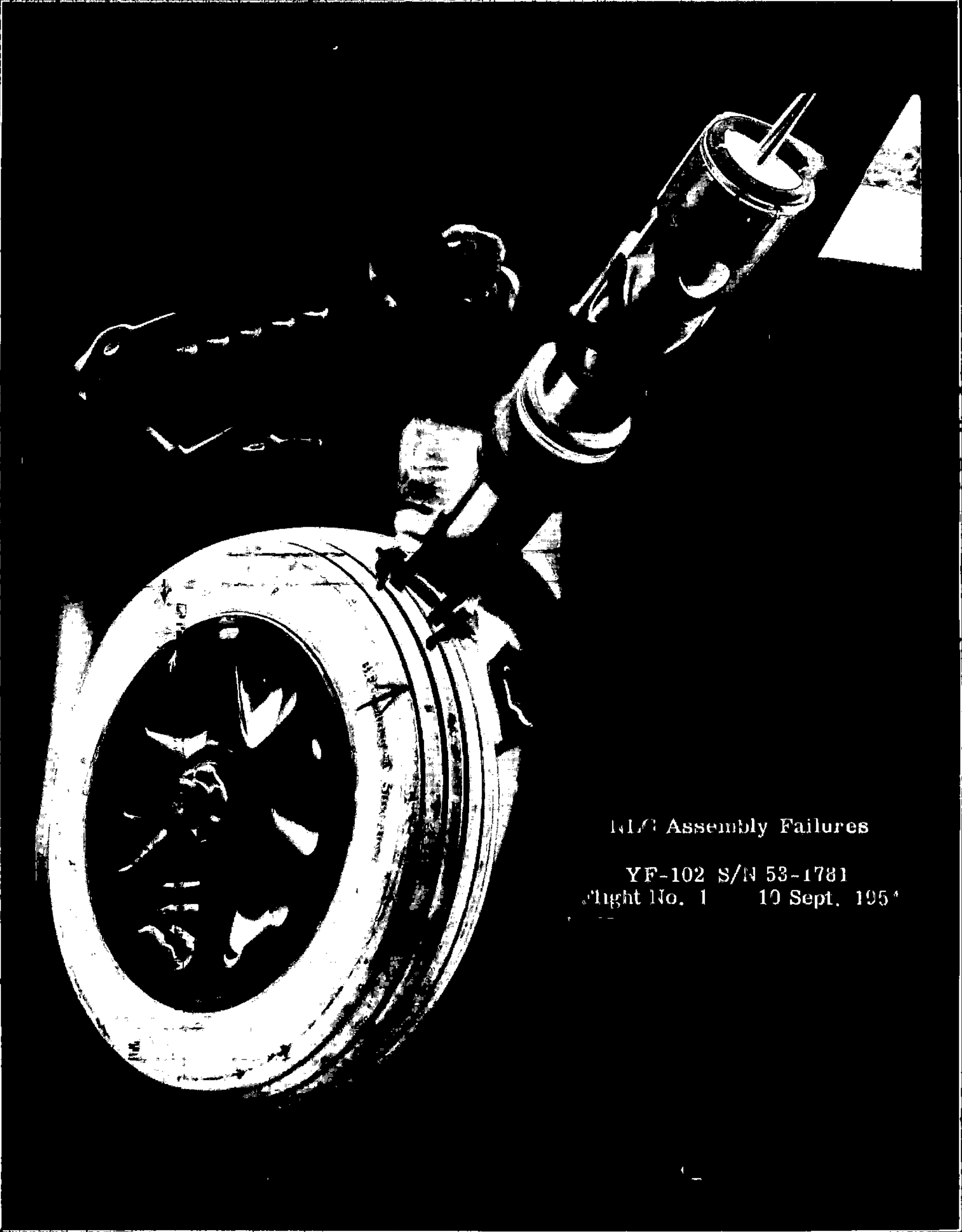
PAGE NO. 15
REPORT NO. ZC-8-504-4
FIGURE NO. 3

ON

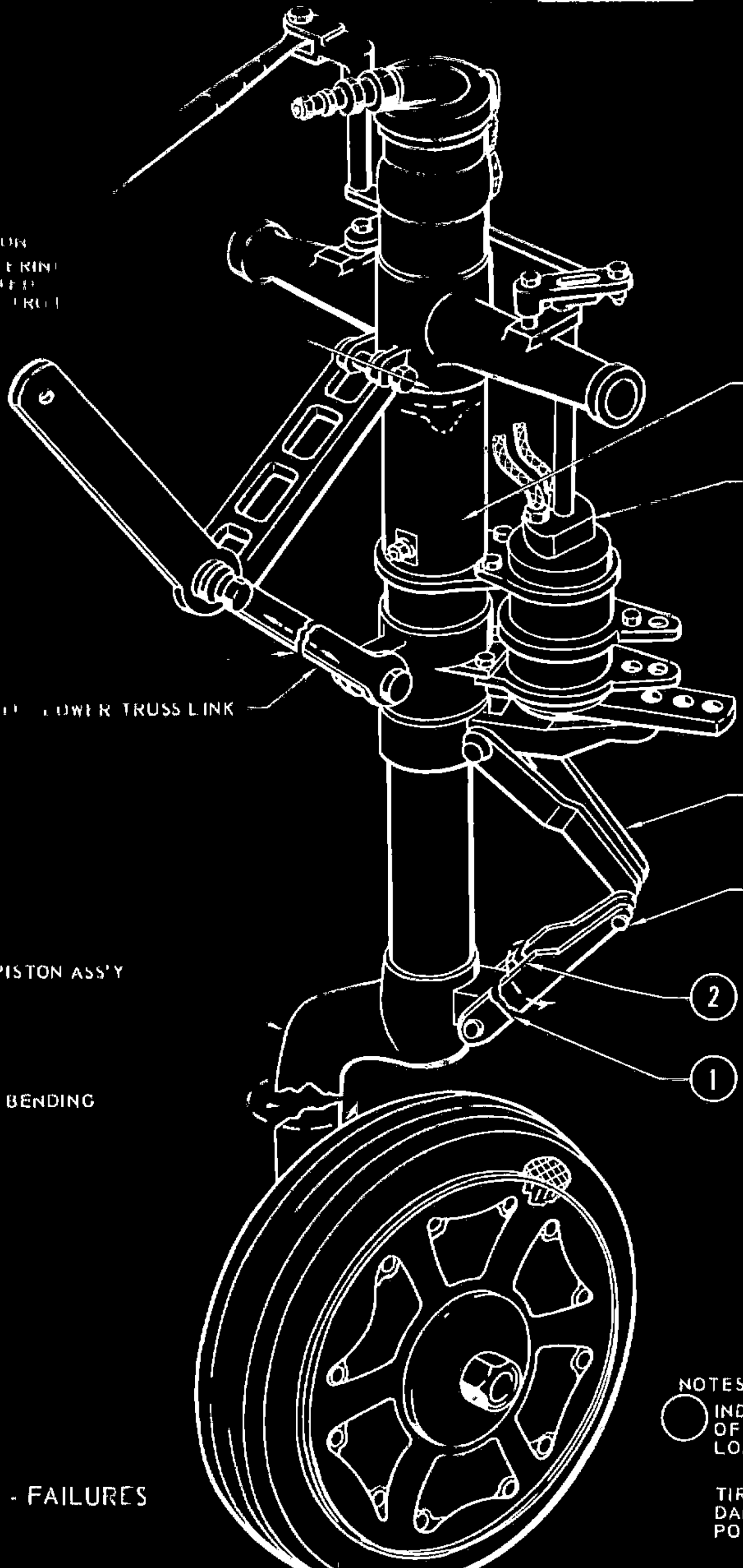


NDING TRACK SCHEMATIC
YF-102 S/N 53-1781
IGHT NO. 16 10 SEPT. '54



N/G Assembly Failures

YF-102 S/N 53-1781
Flight No. 1 10 Sept. 1954

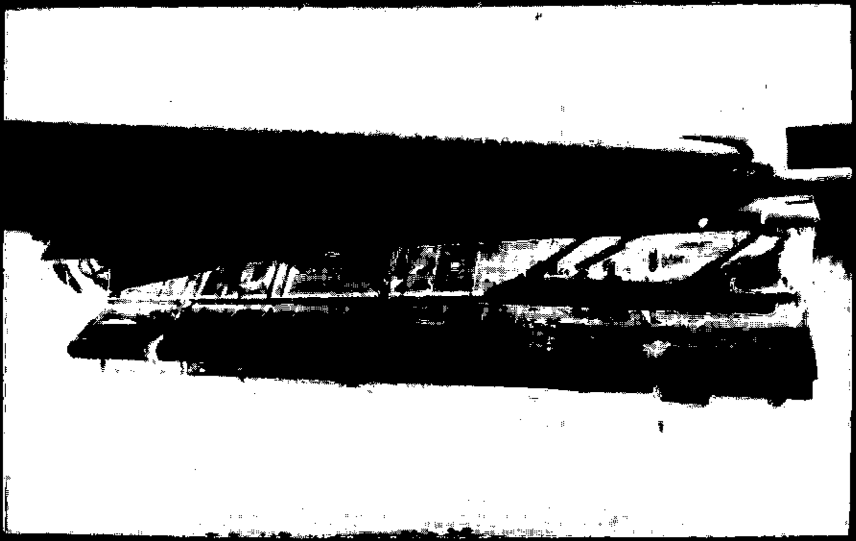


NLG ASSEMBLY - FAILURES

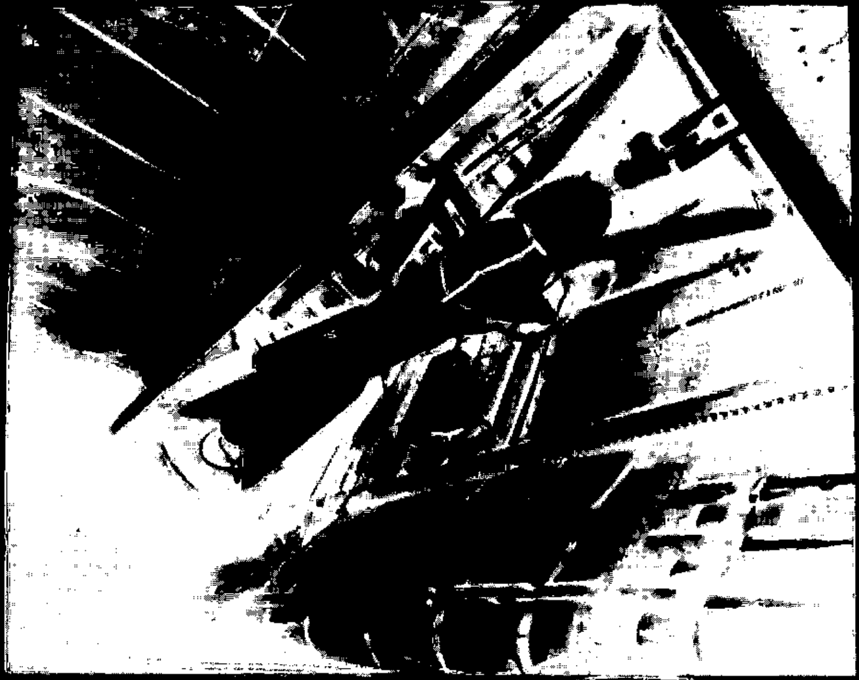
NOTES:

○ INDICATES SEQUENCE OF FAILURES AND LOAD APPLIED.

TIRE & WHEEL CASTING DAMAGED BY BROKEN PORTION OF TORQUE ARM.



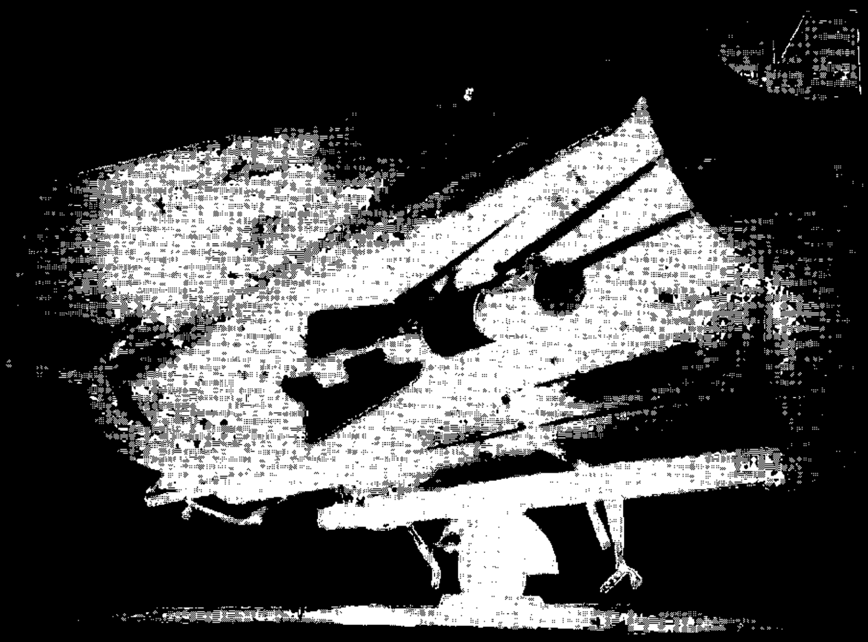
Right Hand Side of Airplane



Right Hand Aft Missile

MISSILE EXHAUST DUST CONTAMINATION
(Dark Streaks are marks made in dust)

YF-102 S/N 53-1781
Flight No. 15 9 Sept. 1954



MISSILE BAY DOORS OPENED
W/ FORWARD MISSILE IN EXTENDED POSITION

MISSILE BAY DOORS OPENED
W/ AFT MISSILE IN RETRACTED POSITION

YF-102 S/N 53-1781

REPORT EF-8-84
MASTER VELLUMS

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DIVISION San Diego

MODEL YF102-1

REPORT EF-8-84

DATE 6 November 1953

[redacted]

AUTHORITY: CMDR, SAAMA
BY: C. L. RIPLEY, PUSO
DATE: 11-6-53

TITLE

FLIGHT TEST SUMMARY

FLIGHT NOS. 1 THROUGH 6

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Declassified by: Dept. Date
130-1 4-6-67

SUBMITTED UNDER

AF33 (600)-594

PREPARED BY Experimental Flight and Technical Groups

GROUP Experimental Flight

REFERENCE

CHECKED BY R. L. Bayless, T. L. Maloy

APPROVED BY W. W. Fox

NO. OF PAGES

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REVISIONS

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A	9 Nov.1953			8,15,16,22,43

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FOREWORD

The Convair YF-102 is a prototype of a single place, single engine, all weather interceptor aircraft, featuring a thin sixty degree delta planform wing and a half delta vertical fin; without horizontal tail surfaces. The production airplane is designed to operate as an all weather interceptor at altitudes up to 60,000 feet with an armament configuration consisting of various combinations of 2.75 in. rockets, and a maximum of six GAR-1 Falcon missiles. The engine (interim) used for the first series of aircraft is a Pratt and Whitney J57-P-11. Additional tests are programmed for a Wright J67-W-1 engine (design) installation at a later date.

A few of the physical features of the airplane are as follows:

1. A six missile bay area with center bay doors incorporating integral tandem rocket launchers.

2. Three hydraulically operated speed (dive) brakes installed between wing and vertical fin (one on each side), and on fuselage bottom centerline.

3. Drogue or braking parachute installed in tail cone at base of vertical fin.

PAGE 2
REPORT # EF-8-84

PHOTO
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Page 3
REPORT # EF-8-84

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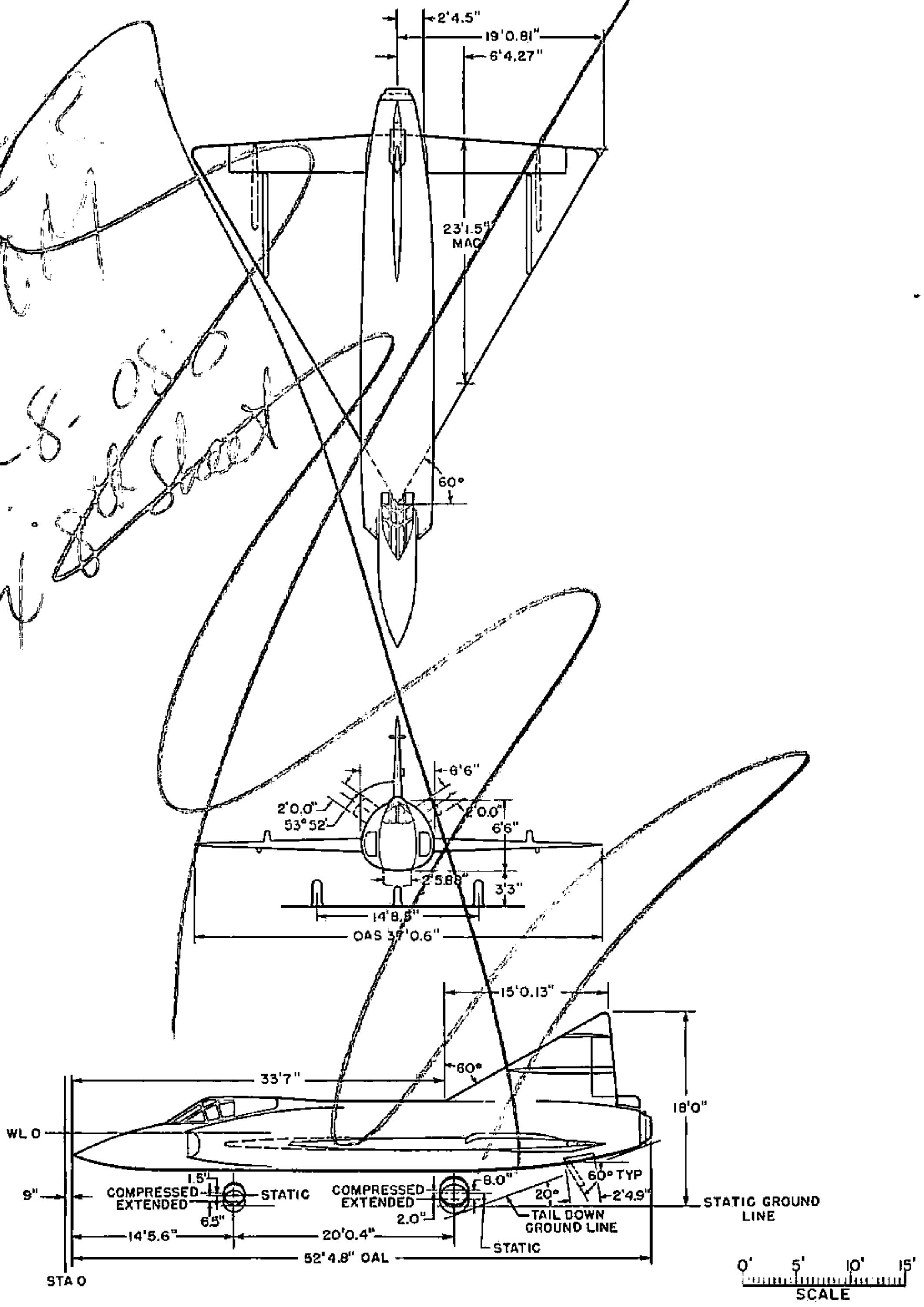

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INTRODUCTION

This report is intended to summarize the flight testing of the YF-102 airplane, USAF S/N 52-7994 during the period from 22 October 1953 to 31 October 1953. The testing was conducted from the Air Force Flight Test Center, Edwards, California, and consisted of low and high speed taxi tests and six flights. The total flight time for the period was one hour and forty-seven minutes (01:47).

These flights were mainly devoted to aircraft systems shakedown and preliminary in-flight flutter investigation. Aircraft systems presenting some difficulty were: the landing gear retraction system, the fuel transfer system, and the artificial feel system for the irreversible flight controls. The flutter program has been carried through a Mach number of 0.826 with no adverse conditions being noted.

The instrumentation installed, for recording of the test data, consisted of three separate recording systems:

1. An automatic photo-recorder located in the nose section of the fuselage, and accommodating up to forty (40) standard size instruments.
2. Four (4) recording oscillographs of eighteen (18) channels each, installed in the nose section just aft of the photo recorder.
3. An FM/FM telemetering system employing two (2) standard RDB bands, located in the missile bay area.

Time correlation was provided for all three (3) systems. Detailed descriptions of all instrumentation and installations may be found in reference 1.

SUMMARY OF RESULTS

FLIGHT AND GROUND HANDLING CHARACTERISTICS

1. The longitudinal stability and control is inherently good, with no adverse characteristics except excessive longitudinal control friction which made precise control of the airplane difficult.

2. The directional-lateral stability and control is good, with no objectionable characteristics except very light rudder and aileron control forces and high lateral control sensitivity -- both a function of the mechanical characteristics of the control system.

3. There are no objectionable trim changes with landing gear retraction or extension. Unsymmetrical extension of the speed brakes results in a directional disturbance to the airplane, and also results in a definite but not particularly objectionable nose-up pitching tendency.

4. The take-off and landing characteristics are very good, again with the exception of the control problem associated with the excessive friction in the longitudinal control system.

5. The ground handling characteristics are generally good with the one qualification that the wheel brake effectiveness is very poor and should be improved.

FLIGHT FLUTTER AND VIBRATION

1. Experimentally determined damping coefficients were obtained from in-flight flutter tests at $M = .675$ and $M = .826$. These damping coefficients were greater than those predicted theoretically, indicating at the speeds tested that, from the flutter viewpoint, the airplane is more stable than predicted theoretically.

PERFORMANCE

Drag points reduced from meager afterburner climb and military level flight data, covering a speed range up to approximately $M = .95$, fall somewhat below the drag curve indicated by PARD model tests. In judging the accuracy of these drag points, it must be kept in mind that there is only an estimated airspeed calibration, that calculated flight thrust data are subject to question and that the number of drag points available may be insufficient to establish a trend.

Take off distance measured on one flight was slightly more than predicted.

POWER PLANT OPERATION

Powerplant handling characteristics were generally satisfactory. A low military and maximum afterburning power engine speed and turbine discharge temperature condition were present throughout the flight test program, and could not be adjusted to specified temperature and speed.

EQUIPMENT OPERATION

Equipment operation in general has been very satisfactory for an airplane in this stage of development.

PILOTS COMMENTS

The #1 YF-102 made two taxi runs and six flights. During these operations the airplane proved to be exceptional during take-off, approach and landing. The aircraft possessed positive stability at all speeds flown. No unusual flying characteristics were noted, and pilot transition from conventional aircraft to this type of airplane should be no problem. Take-off and landing characteristics were normal. Pilot visibility for take-off and landing has been adequate. Stability and control at take-off, approach and landing speeds was excellent. Only minor mechanical difficulties have been encountered to date.

CHASE PILOT COMMENTS

The chase plane pilot's report covering all flight operations of the YF-102 is included on the following page.

REVIEW OF TEST PROGRAM PROGRESS

The flight test program of the Convair YF-102 was initiated at the Edwards AFFTC facility on 22 October 1953 with the advent of taxi operation No. 1. Although both main landing gear tires were blown at high speed during the first taxi run, no serious directional control problems were indicated. Additional satisfactory high speed taxi operations, including a sustained lift-off of several hundred yards, were accomplished on 23 October 1953. The first flight was made on 24 October 1953, and was of 16 minutes duration. The landing gear could not be fully retracted during this flight despite repeated cycling. Flight two was made on the same day following minor landing gear system changes; gear retraction was still not achieved. Flight three was made on 28 October, and the main landing gear again failed to completely retract. During flight three, the left hand low fuel level warning light came on during the climb to operating altitude, and caused a discontinuance of additional flight.

Flight four was flown 29 October, and considerable quantitative data were obtained.

Although gear up lights showed an unlocked condition, the chase plane pilot reported all door fairings were flush. A full afterburning power climb was made to approximately 35,000 feet. Level flight flutter and vibration instrumentation sweeps were achieved at this altitude with a nominal Mach number of 0.8.

Flight five was made on 29 October, and was terminated after repeated efforts to fully retract the gear had failed. Only 95% engine speed could be obtained at maximum and military power for this flight.

Flight six was made 31 October. The addition of a hydraulic booster jack resulted in successful main gear retraction and locking on the first attempt. The left low level fuel warning light came on shortly after take-off, and remained on for the duration of the flight. Flutter investigation was accomplished at 25,000 feet at indicated mach numbers of 0.65, 0.70 and 0.80. Moderate buffet was noticed during speed brake extension at approximately 0.55 mach number.

DETAILED LOG OF TESTS -- YF-102 No. 52-7994

Operation No.	Date	I.G.W (lbs.)	I.C.G. (% mac)	Flight Time	Flight Time (to date)
Taxi 1	22 October 1953	27,947	30.2	00:14 (Taxi)	--
Taxi 2	23 October 1953	27,947	30.2	00:16 (Taxi)	--
Flight 1	24 October 1953	27,988	30.3	00:16	00:16
Flight 2	24 October 1953	27,978	30.3	00:08	00:24
Flight 3	28 October 1953	27,967	30.2	00:07	00:31
Flight 4	29 October 1953	27,967	30.2	00:33	01:04
Flight 5	29 October 1953	27,957	30.2	00:15	01:19
Flight 6	31 October 1953	27,947	30.2	00:28	01:47

Pilot for all operations was R. L. Johnson of Convair.

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A Revision
9 November 1953

STATEMENT

Chase Pilot Observations - YF-102

All flights of the YF-102 have been escorted by Major Yeager and Captain Nash of the Air Force Flight Test Center. The following is an account of chase pilot observations of the YF-102 concerning the stability, control and performance of the aircraft.

A. Take Off - All take-offs were accomplished with afterburner, and the take-off roll appeared to be 5500 to 6000 feet. On the first flight it was noticeable that the pilot tended to over control and attain a steep angle of attack shortly after nose wheel lift off, but this condition was immediately corrected. This was not noticed after the first flight. No lateral DIRECTIONAL ~~differential~~ or longitudional instability was evident in the YF-102 during the take off or during the acceleration to climb speed. The aircraft accelerated very rapidly after gear retraction.

B. In Flight Stability and Performance - The YF-102 appeared to be very stable in flight and there was no evidence of any serious control deficiencies. The dive brakes were opened differentially and the actuation caused the aircraft to pitch-up. Control of pitch up was rapid. The dive brakes were very effective. The differential actuation of dive brakes should be easily corrected by balance of hydraulic flows to each of the units.

On flight No. 6 the pilot of the YF-102 did not use afterburner for the climb to 25,000 feet. The chase pilot was able to keep up with the YF-102 in the F-86E during the climb. When the pilot of the YF-102 leveled off at 25,000 feet the aircraft pulled AWAY ~~out~~ from the F-86E which was using maximum power. The F-86E airspeed system indicated a Mach Number of .895. This period of level flight was for a very short period and the YF-102 pilot stated that he was returning to the Base for a landing because of a fuel flow deficiency that was giving him a wing heavy condition.

On another occasion the YF-102 climbed from 5000 feet to 15000 feet without afterburner. The F-86E was indicating a rate of climb of 4000 feet per minute. At 15000 feet the YF-102 lit the afterburner and climbed to 38000 feet. When the YF-102 reached this altitude the F-86E was at 23800 feet. The best climb speed was used on the F-86 E.

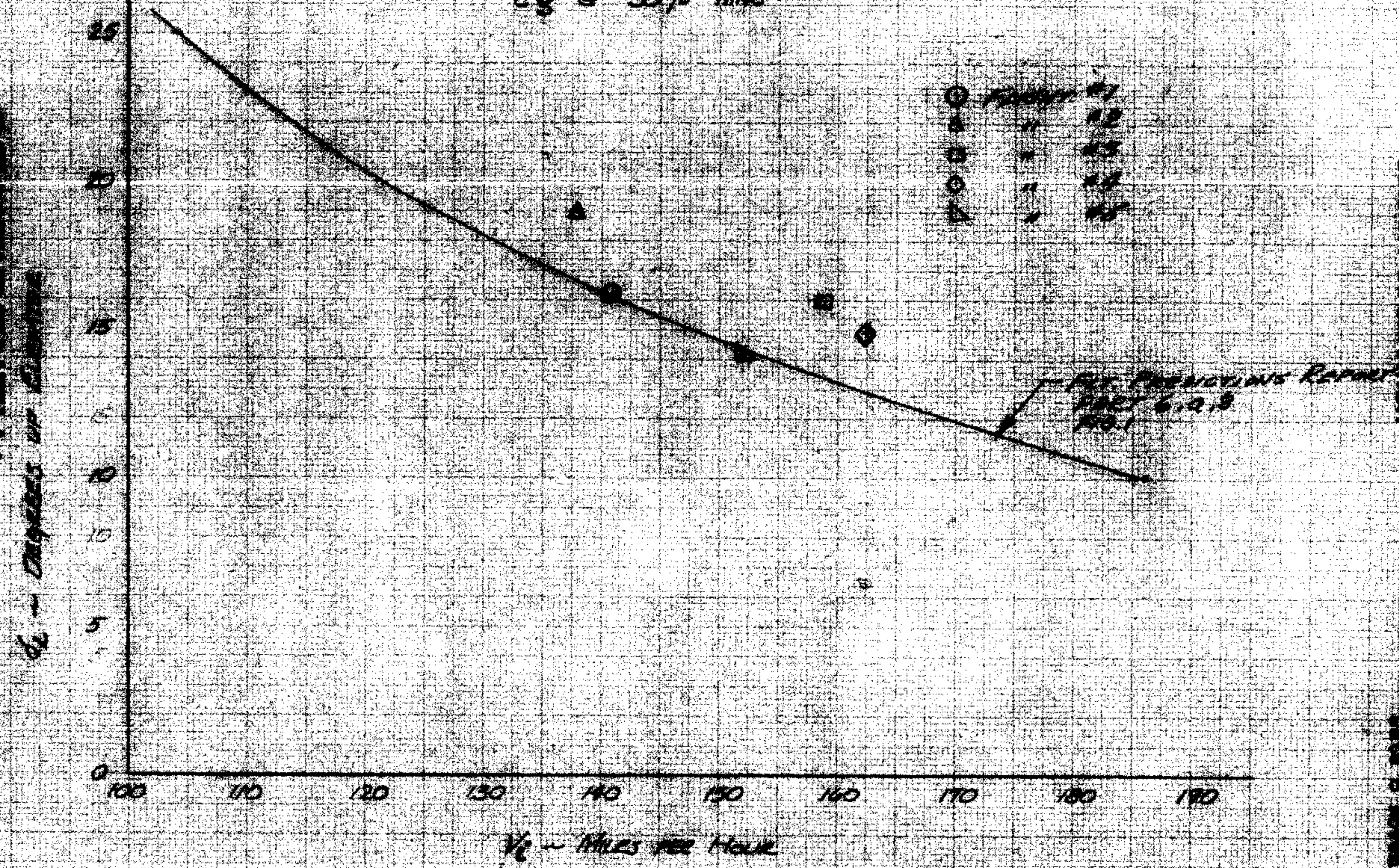
At all times during flight the YF-102 appeared to be stable and did not have any control difficulties.

C. Landing - All landings in the YF-102 were accomplished on the lake bed. The YF-102 appeared to be very stable on the approach with the gear extended. The landings were very smooth and the aircraft appeared to be under very good control.

Charles E. Yeager
CHARLES E. YEAGER
Major, USAF
Assistant Chief,
Flight Test Laboratory

FIG. 1

ELEVATOR REQUIRED TO RAISE NOSE GEAR
YF-102 AIRPLANE
GROSS WEIGHT - 27,750#
CG @ 30% MAC



FLIGHT AND GROUND HANDLING CHARACTERISTICS

LONGITUDINAL STABILITY AND CONTROL

A preliminary qualitative evaluation of staic longitudinal stability indicates satisfactory characteristics. Generally, push forces are required to increase airspeed and pull forces required to reduce airspeed.

ELEVATOR EFFECTIVENESS

Raising Nose Gear Off Ground: The following plot (page12) shows the correlation between the predicted values of speed for nose gear lifting and those obtained from flight test data. In general, there is good correlation, but high friction forces prevented obtaining accuracy on elevon angles.

Elevator Effectiveness: Page 13 shows the correlation of elevator angle and angle of attack between predicted and flight test data for level flights at 25,000 and 35,000 feet. These data are for the right elevon only (photo panel) and may not be exact since some aileron was used during these tests. Angle of attack from flight has not been corrected for upwash effects.

Page 14 shows elevator angle and angle of attack during climb from flight #4 for two flight segments. In general, again, good correlation exists between flight test and predicted data.

From the initial taxi tests, longitudinal control of the airplane was encumbered by excessive friction in the control system. As a result, precise control was difficult. This was particularly noticeable on take-off and landing wherein there was a marked tendency to over-control. It was felt that even the take-off performance of the airplane was affected adversely by this condition.

The airplane appears to have positive damping for short period dynamic longitudinal stability. No rigorous evaluation for this has yet been made.

DIRECTIONAL AND LATERAL STABILITY AND CONTROL

The directional-lateral stability and control characteristics appear to be inherently good. However, the rudder and aileron control forces are extremely light at the present time. The rudder forces, in particular, have been light to the degree of being unsatisfactory. The aileron control, besides incorporating light forces, have been over-sensitive from the standpoint that an objectionable small lateral control stick movement is involved in the normal control of the airplane. It is believed that decreased sensitivity, especially near the neutral stick position, would be highly desirable.

DIRECTIONAL AND LATERAL STABILITY AND CONTROL (Continued)

In the very preliminary evaluation accomplished to date, there appeared to be a definite positive effective dihedral throughout the speed range tested.

At low speed in the landing approach, the poor lateral-directional characteristics ported for the XF92 airplane has not been encountered in the YF-102 airplane.

The airplane appears to have positive damping for short period directional-lateral stability. No rigorous evaluation for this has been made.

TRIM

The trim rate about all axes is objectionably slow for the airspeed range investigated to date.

TRIM CHANGES

There is no objectionable trim change with gear retraction or extension.

With extension of speed brakes, a longitudinal nose-up pitching and directional oscillation was noticed. This is believed due to the fact that the speed brakes were observed to not extend simultaneously. This points out the desirability of providing synchronized extension of the speed brakes.

TAKE-OFF AND CLIMB

The take-off and initial climb characteristics are good in that a smooth breakaway occurs without any lateral-directional instability in evidence. The longitudinal control friction resulted in serious over-controlling of the airplane during take-off for the C.G.'s tested (C.G. at take-off is approximately 30%).

APPROACH AND LANDING

The handling characteristics of the airplane during the landing approach were good in the smooth air encountered thus far, up to the point of touchdown at which time the longitudinal control friction again became objectionable.

NOSE GEAR STEERING

The nose gear steering functions very well, with the qualification that it appears to be somehwat over-sensitive . The restricted turn radius (imposed by a temporary rudder travel restriction) is somewhat inconvenient.

WHEEL BRAKES

The wheel brake effectiveness is poor, but the braking action is smooth and even. The brake effectiveness should be improved for satisfactory taxi and landing run operation. In one landing, full brake application at approximately 70 MPH resulted in very little additional deceleration.

DIRECTIONAL CONTROL

The rudder becomes effective in accomplishing directional control at approximately 100 MPH indicated airspeed.

On blowing the main gear tires in quick succession at 195 MPH indicated on the ground, there was no serious directional control difficulty.

Operation of the airplane in cross winds is considered very good. The take-off and landing characteristics in a 20 MPH cross-wind are considered equivalent to those of a conventional airplane.

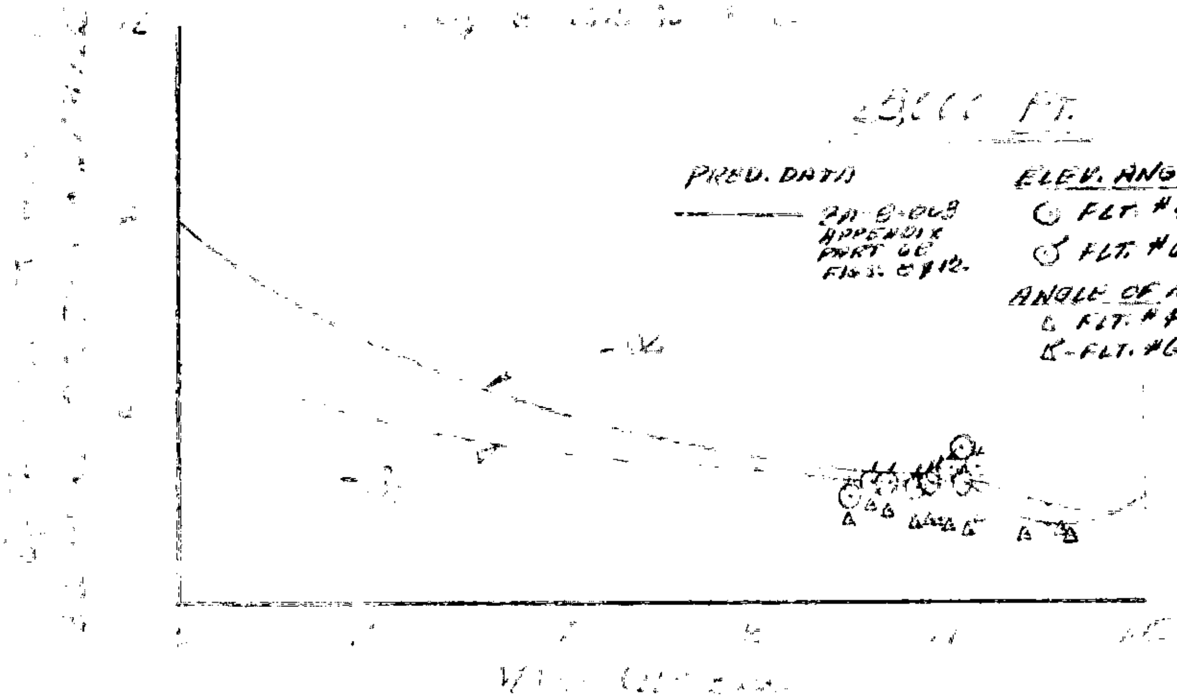
THRUST FOR TAXIING

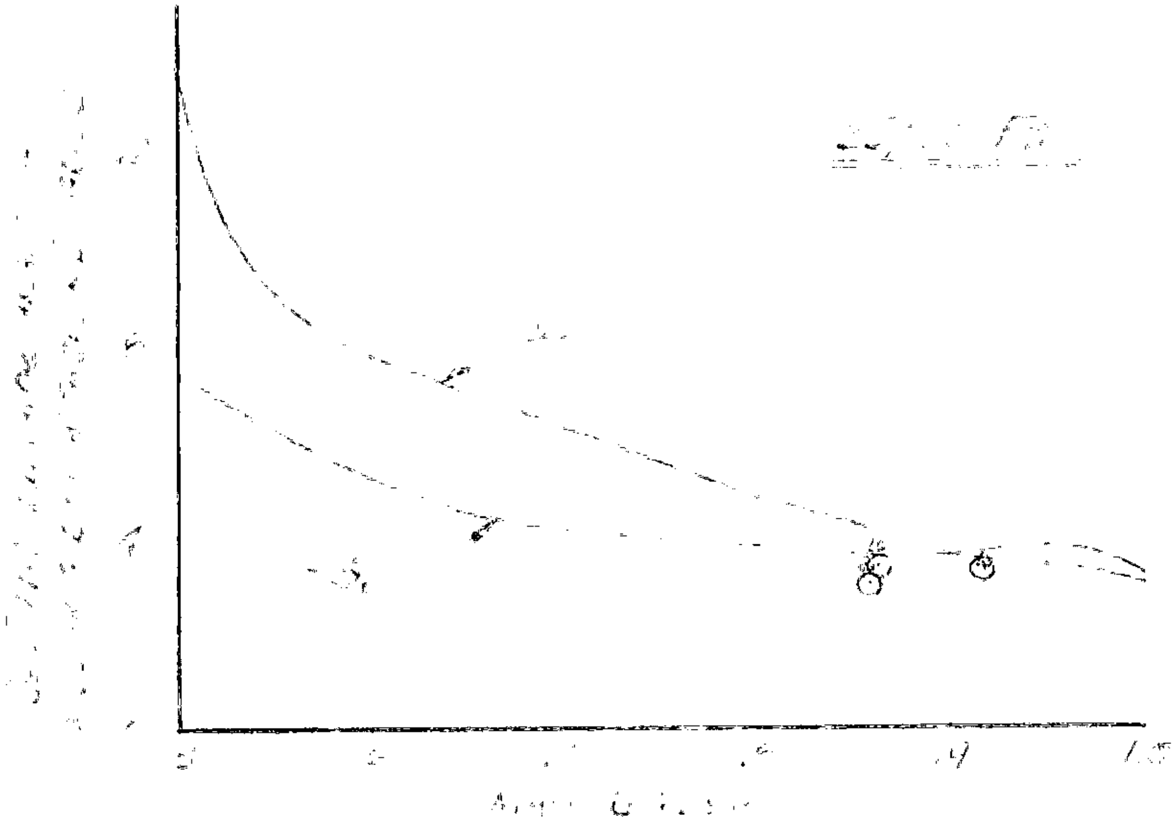
Idling engine power was found sufficient to taxi the airplane satisfactorily.

VISIBILITY

It was felt that, while generally good, improved visibility would be an advantage in both ground and flight operation.

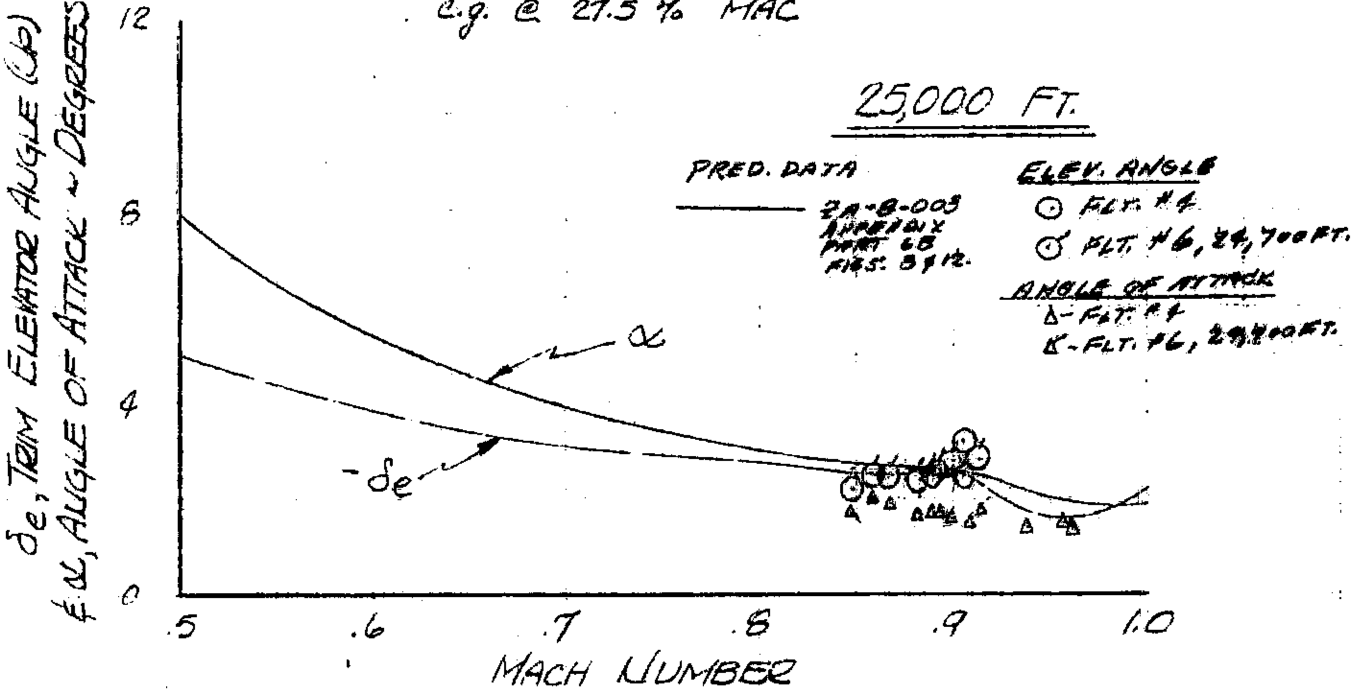
[illegible]

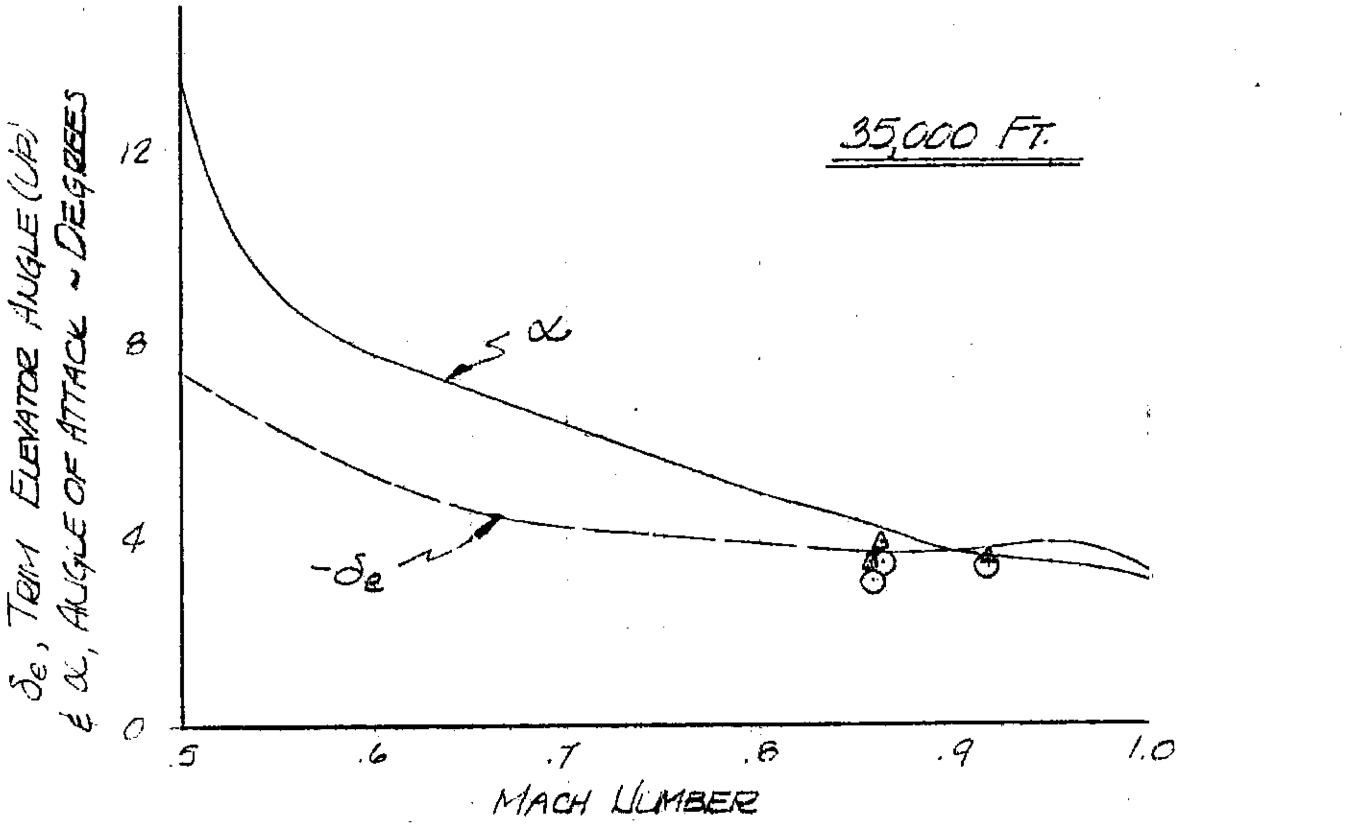

PRED. DATA
ZA-8-008
APPENDIX
PART 6C
FIGS. 6 & 12.
ELEV. ANGLE
FLT. #4
FLT. #6, 24,700 FT.
ANGLE OF ATTACK
FLT. #4
FLT. #6, 29,400 FT.



TRIM ELEVATOR ANGLE & ANGLE OF ATTACK
LEVEL FLIGHT
YF-102 NO. 1 AIRPLANE

GROSS WEIGHT = 26,000 LBS.
C.G. @ 27.5 % MAC





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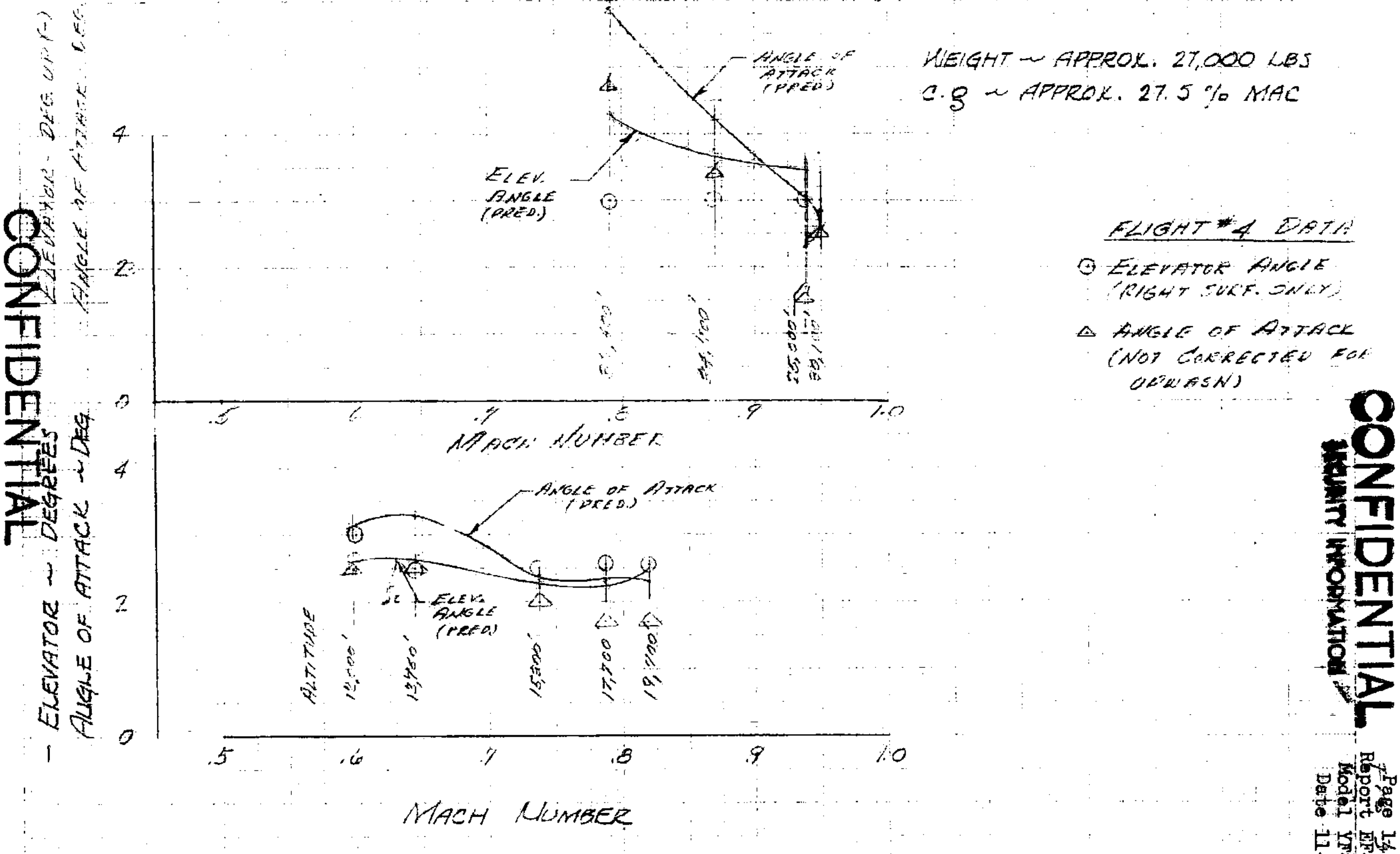
YF-102 AIRPLANE NO. 1
ELEVATOR AND ANGLE OF ATTACK DURING CLIMB



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9 November 1953

FLIGHT FLUTTER TESTS

The purpose of the flight flutter investigation was to verify the safety of the airplane from the flutter viewpoint, and to correlate the results of theoretically determined damping coefficients with experimentally determined damping coefficients.

Useable measurements were only obtained on Flight #6, and these are reported herein.

The results of theoretical flutter calculations are given in Reference 2. The results of ground vibration tests are given in Reference 3. Results of these investigations are utilized in the discussion of the flight flutter measurements.

Provision was made to furnish excitation to the wing by means of two Convair developed hydraulic shakers located symmetrically within the wing as shown on page 18. The shaker force varied parabolically from 0# at 0 cps to ±50# at 8 cps, and was ±50# from 8 cps to 70 cps. Provision was made to excite the wing both symmetrically and anti-symmetrically. Electronic controls were used to control the vibrator for the purpose of obtaining frequency "sweeps" or decay records.

Accelerometer and deflection pickups, as shown on page 18, were installed to measure responses. The accelerometer pickups were Statham A-17-5-120. The deflection pickups were Galetronics SF-6 high resolution potentiometers.

The test procedure consisted of making symmetric and anti-symmetric frequency sweeps from 7 cps to 70 cps and recording the resulting responses.

The sweeps were made at $M = .675$, $h = 18,250$ ft., and $V_t = 491$ MPH, and at $M = 0.826$, $h = 27,950$ ft. and $V_t = 570$ MPH. Records were obtained only for Oscillograph #1 pickups, since the galvanometer lamps on Oscillographs #3 and #4 burned out after takeoff.

The damping coefficients were obtained from the response curves in the manner outlined below.

Page 19 shows typical acceleration responses obtained for three channels. The responses obtained in flight were unsymmetrical with either symmetric or anti-symmetric excitation. Utilizing these responses and the calibration factors, amplitude versus frequency plots were obtained as shown on page 20 for Channel 4 of Oscillograph #1 (See page 18). The damping coefficient, g, is then readily obtained as shown on page 20.

Proceeding in the same manner as noted above for the speeds mentioned previously results in the damping coefficients shown in the table on page 17. This table shows the damping coefficients which were obtained from channels having readable responses.

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FLIGHT FLUTTER TESTS CONT'D

Discussion

The critical (i.e., lowest) calculated damping coefficients obtained for symmetric and anti-symmetric vibration modes are shown on page 21. These were obtained from Reference 2. It is noted that the damping coefficient at zero speed is conservatively taken as g = -0.04; this is the lowest value obtained from decay measurements on the actual airplane. The experimentally determined damping coefficients from the table on page 17 are also plotted on page 21; lines connecting the points pertain to common channels and common frequencies as shown in the table on page 17.

A comparison of the calculated and measured damping coefficients shown on page 21 shows that the calculated damping coefficients are conservative, that is, the experimental damping coefficients are higher than the predicted damping coefficients, which indicates that the airplane is more stable from a flutter viewpoint than predicted theoretically.

It is noted that although the calculated and measured damping coefficients were obtained at different altitudes, the aforementioned comparison is considered valid since theoretical calculations do not show an appreciable difference in damping at sea level and at 35,000 feet (See Reference 2).

Attention is also invited to the fact that the YF-102 airplane reached a Mach number of 0.951 at a 33,670 foot altitude (i.e., a true speed of 644 MPH) during Flight #4 without encountering flutter. No damping measurements were obtained during this flight since the shakers malfunctioned.

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YF-102 AIRPLANE EXPERIMENTAL DAMPING COEFFICIENTS

FLIGHT #6

M = 0.675 h = 18,250 ft. V_t = 491 MPH			M = .826 h = 27,950 ft. V_t = 570 MPH		
Channel	g	f (cps)	Channel	g	f (cps)
4	-.094	11.84	4	-.098	12.32
6	-.124	11.84	6	-.134	12.32
9	-.098	11.84	9	-.123	12.32
6	-.124	17.17	6	-.127	17.96
9	-.098	17.17	9	-.110	17.96
16	-.064	17.17	16	-.092	17.96
17	-.073	17.17	17	-.085	17.96
6	-.081	22.30	6	-.078	22.2
6	-.112	26.40	6	-.085	26.65
17	-.101	35.0	17	-.152	36.75
17	-.141	46.5	17	-.178	50.65

Notes:

1. Data for M = 0.675 was from Oscillograph Record #01035, and Time Synchronizer #288 through 304.
2. Data for M = 0.826 was from Oscillograph Record #01039 and Time Synchronizer #584 through 602.
3. Damping coefficients are taken as negative to conform to flutter nomenclature.

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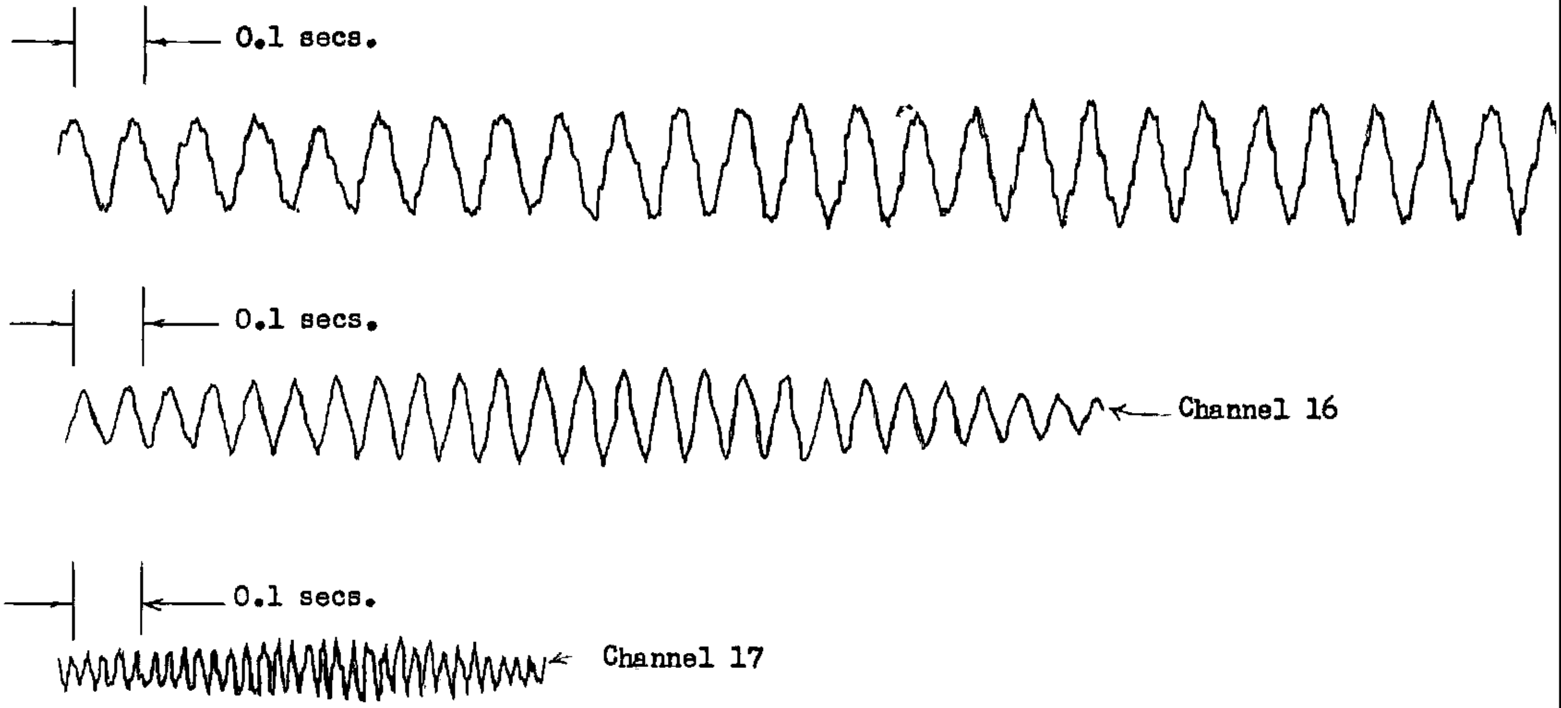
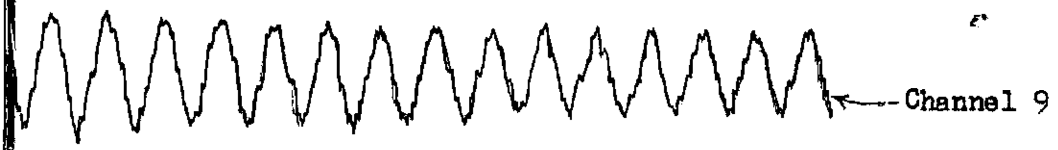
YF-102 AIRPLANE TYPICAL ACCELERATION RESPONSES OBTAINED FROM FLIGHT FL



IGHT FLUTTER TESTS



Notes:

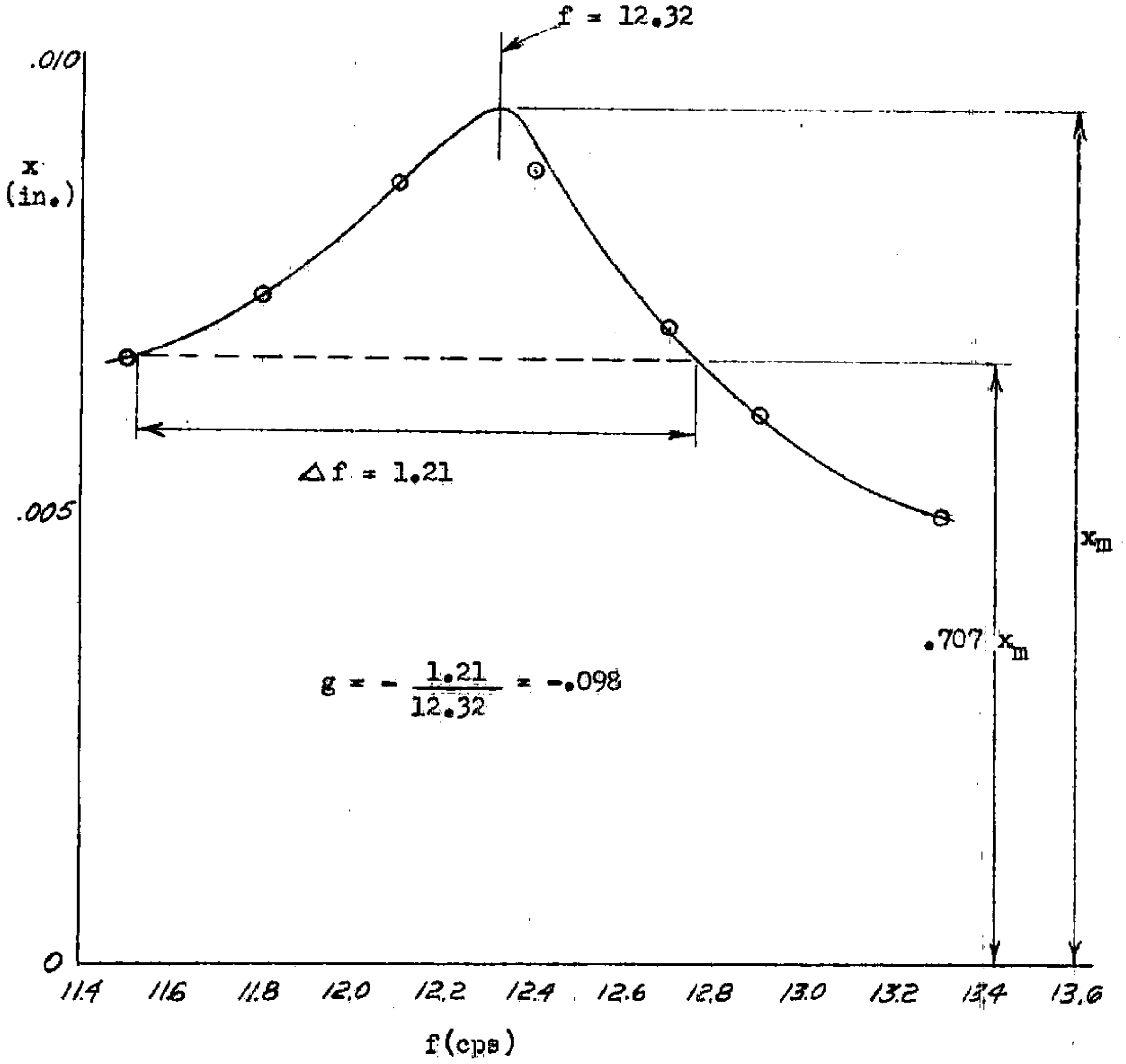
1. Acceleration responses shown were traced from original oscillograph records. Resonant frequency for channel 9 is 12.32 cps. Resonant frequency for channel 16 is 17.96 cps. Resonant frequency for channel 17 is 50.65 cps.

2. Channel numbers are identified on Page 18.

3. Responses shown were obtained on Flight #6 (Oscillograph Record #01039, Time Synchronizer #584 through #602), M = .826, h = 27,950 ft., V_t = 570 MPH.

YF-102 AIRPLANE AMPLITUDE-FREQUENCY PLOT

OSCILLOGRAPH #1, CHANNEL 4

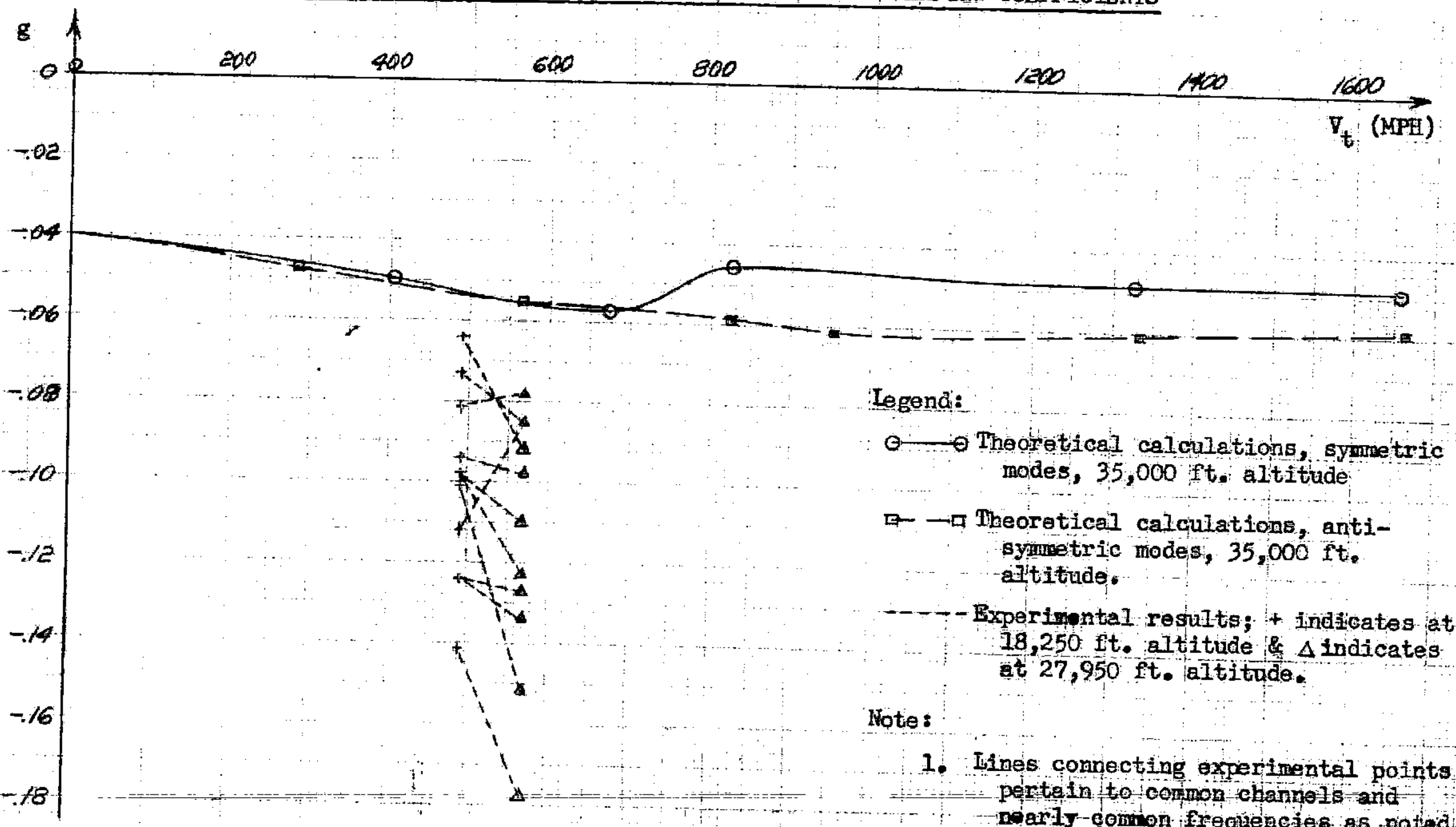


Notes:

1. This plot was prepared for condition given in Note 3 on Page 19.
2. $\pi g = \delta$, where δ = logarthmetic decrement
3. The negative sign for the damping coefficient is taken to conform to standard flutter nomenclature
4. x = Single amplitude

4

YF-102 AIRPLANE THEORETICAL AND EXPERIMENTAL DAMPING COEFFICIENTS



Legend:

○——○ Theoretical calculations, symmetric modes, 35,000 ft. altitude

□— —□ Theoretical calculations, anti-symmetric modes, 35,000 ft. altitude.

- - - - Experimental results; + indicates at 18,250 ft. altitude & Δ indicates at 27,950 ft. altitude.

Note:

1. Lines connecting experimental points pertain to common channels and nearly common frequencies as noted on Page 17.
2. V_t = True speed
3. g = Damping coefficient

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Dive Brake Operation

There was no data obtained on the actual speed brake motion. The observed uneven action of the three speed brake panels is, however, verified by the low frequency disturbance present in the accelerometer data during the speed brake operation on Flight #6. This disturbance, approximately 0.20g normal acceleration (at the c.g. of the airplane) at 0.77 cps, is well damped, becoming undiscernable after two cycles. This observed frequency agrees well with the predicted airplane longitudinal frequency in the speed brake range being considered. It is apparent that position synchronization of the brakes should relieve this disturbance.

The buffet reported by the pilot is evident in the accelerometer data (See table on page 23), with predominant response at the fin tip. It is believed that increased speed brake-fuselage clearance would alleviate these effects.

YF-102 AIRPLANE ACCELERATIONS DUE TO SPEED BRAKE OPERATION

Channel No.	f (cps)	n
2	36.8	.43
3	36.8	.15
4	36.8	.15
5	45	.23
6	12.5	.09
9	10	.16
11	36.8	.64
13	45	.09
17	45	.77

Notes:

1. Data shown in this table were taken on Flight #6 (Oscillograph #1 Record #01042, Time Synchronizer #749 through #761) V_c = 340 MPH at 10,000 foot altitude.
2. For channel locations see page .
3. f = Predominant frequency.
4. n = Single amplitude acceleration in g's due to speed brake operation

PERFORMANCE

DRAG EVALUATION FROM FLIGHT TESTS

Inasmuch as the initial flights were conducted primarily for obtaining overall handling and flutter characteristics, little emphasis was placed on obtaining information which would assist in drag evaluation. However, some meager data were obtained during flights 4 and 6 which give an indication of the drag characteristics at high subsonic speeds. These test points are summarized on the following page (page 26).

Since no calibrations were obtained for the airspeed indicating system including position error for the nose boom static pressures, estimates were made for this correction and are shown on page 34. Utilizing the full afterburner climb test data of flight 4 and the military power level flight data of flight 6, together with estimated calibrations, the data were reduced as shown on the following several pages (pages 27 through 31). Time histories of speed and altitude during the aforementioned portions of flights 4 and 6 are given on pages 32 and 33, respectively.

TAKE-OFF PERFORMANCE

Take-off characteristics as observed from the AFFTC photo-grid for flight no. 3 are shown on page 35.

The take-off distances as observed for the test conditions actually existing are as follows:

Ground Run = 5000 ft.
Total take-off distance over 50 ft. obstacle = 7900 ft.

Correcting these distances to standard conditions at sea level using the method shown in AF Technical Report No. AFFTC 53-11 (Appendix II on page 10) these values for take-off distance correct to:

Ground Run = 3500 ft.
Total take-off distance over 50 ft. obstacle = 4170 ft.

Estimated take-off characteristics for the test gross weight at standard sea level conditions for this airplane are as follows:

Ground Run = 2210 ft.
Total take-off distance over 50 ft. obstacle = 3105 ft.

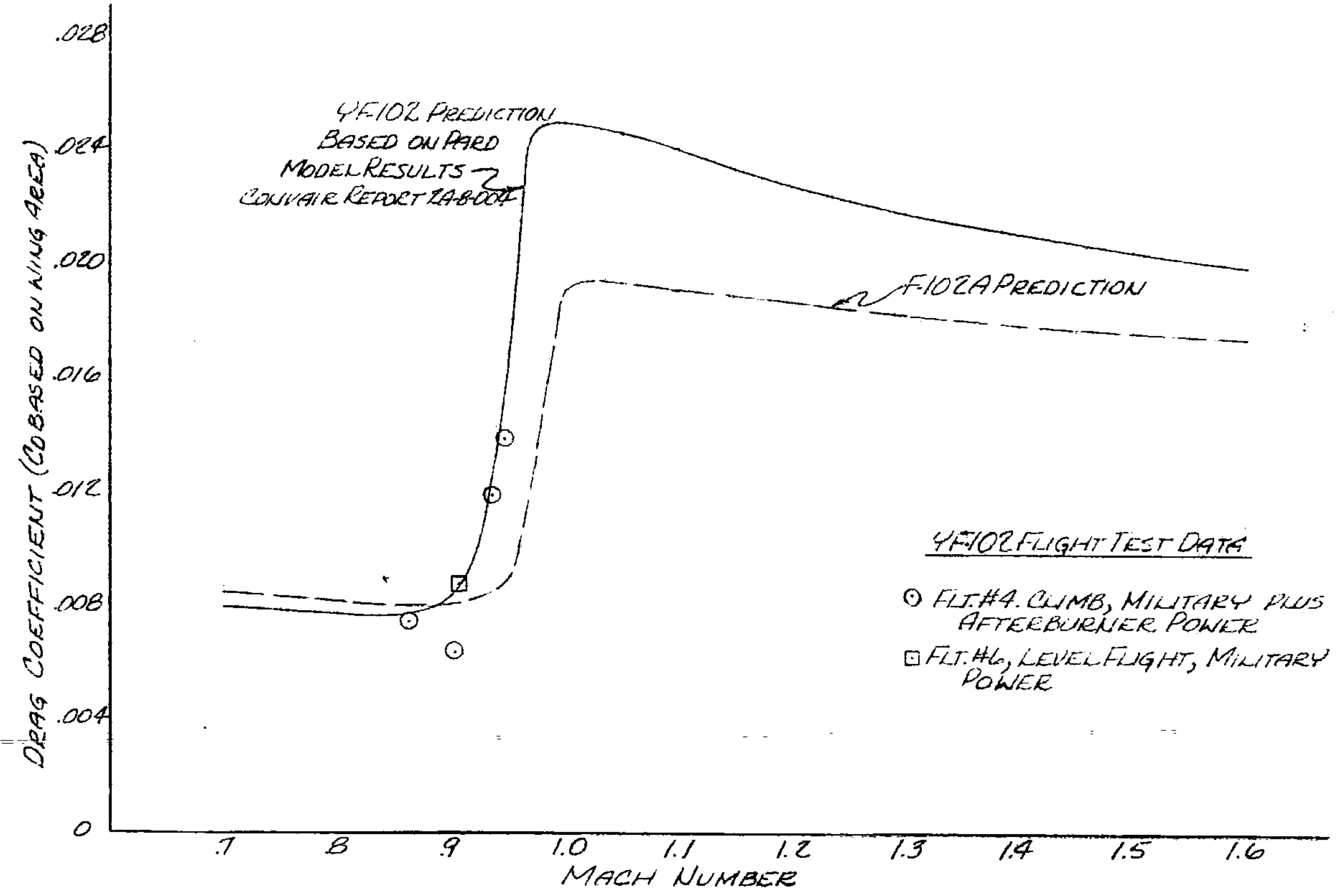
TAKE-OFF PERFORMANCE (Continued)

It was felt by the pilot that distance required to take-off could be reduced considerably if he had longitudinal control without friction. Although thrust was never measured exactly, engine performance parameters which were measured indicate that this particular engine for the first six flights was operating with a deficiency of thrust of the order of 10 to 15 percent. This deficiency alone can account for increased take-off distances of the order of 20 to 25 percent.

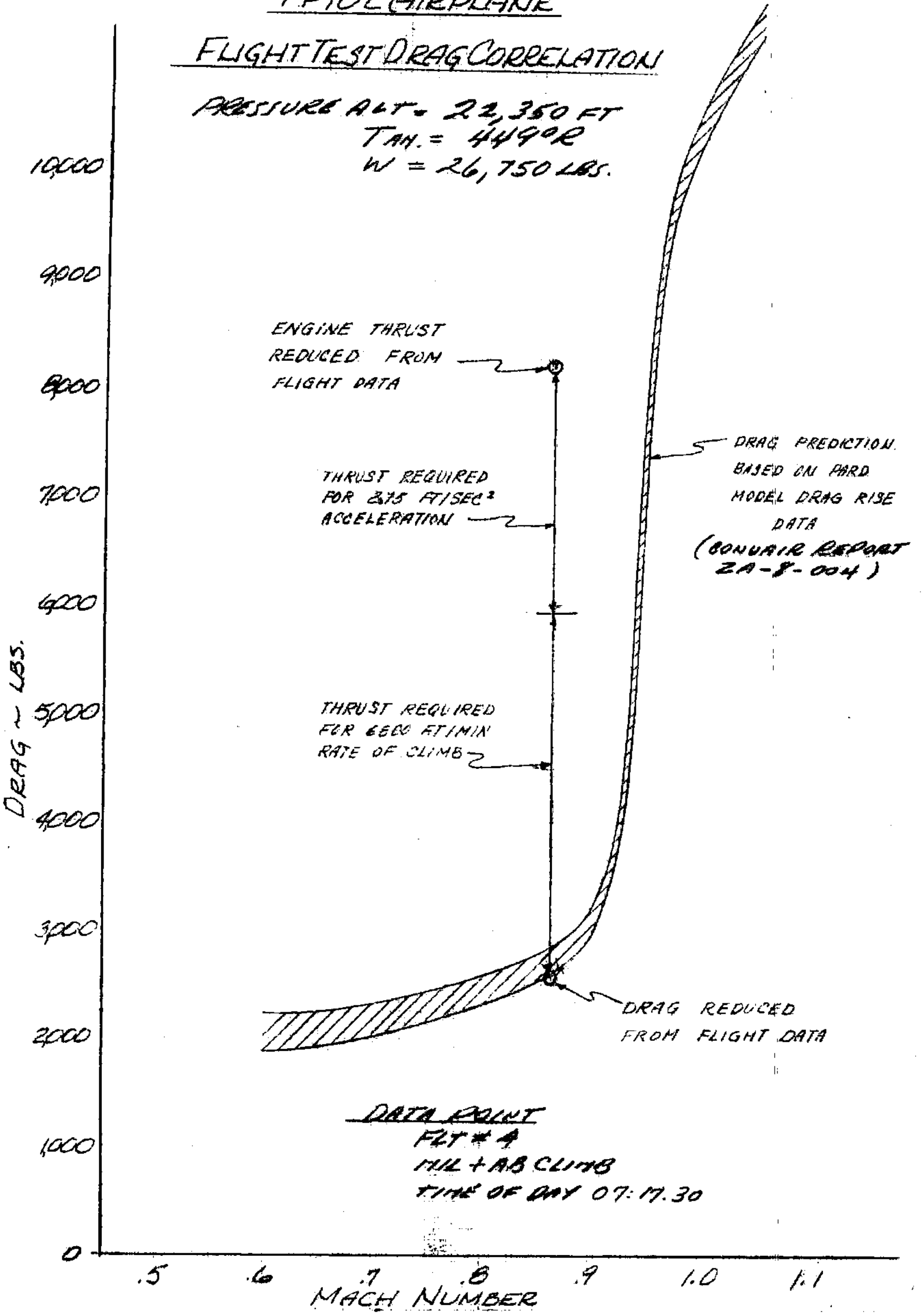
Further, it should be borne in mind that no attempt was made in any of the initial YF-102 take-offs to optimize take-off performance. It is expected that these distances can be considerably improved by optimizing the take-off technique for this airplane.

F-102 AIRPLANE

$C_{D_{MIN}}$ VERSUS MACH NUMBER



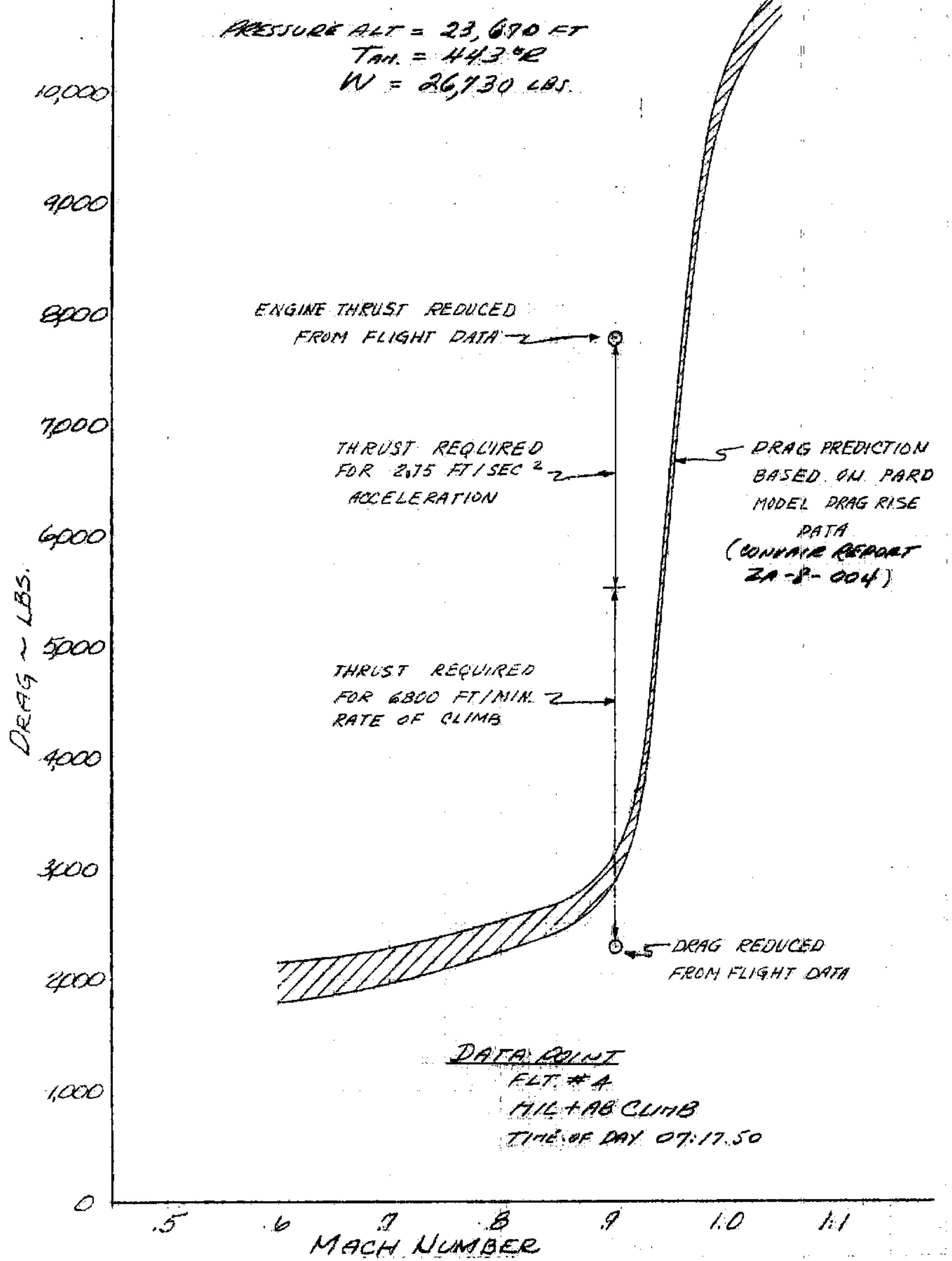
YF102 AIRPLANE



FLIGHT TEST DRAG CORRELATION

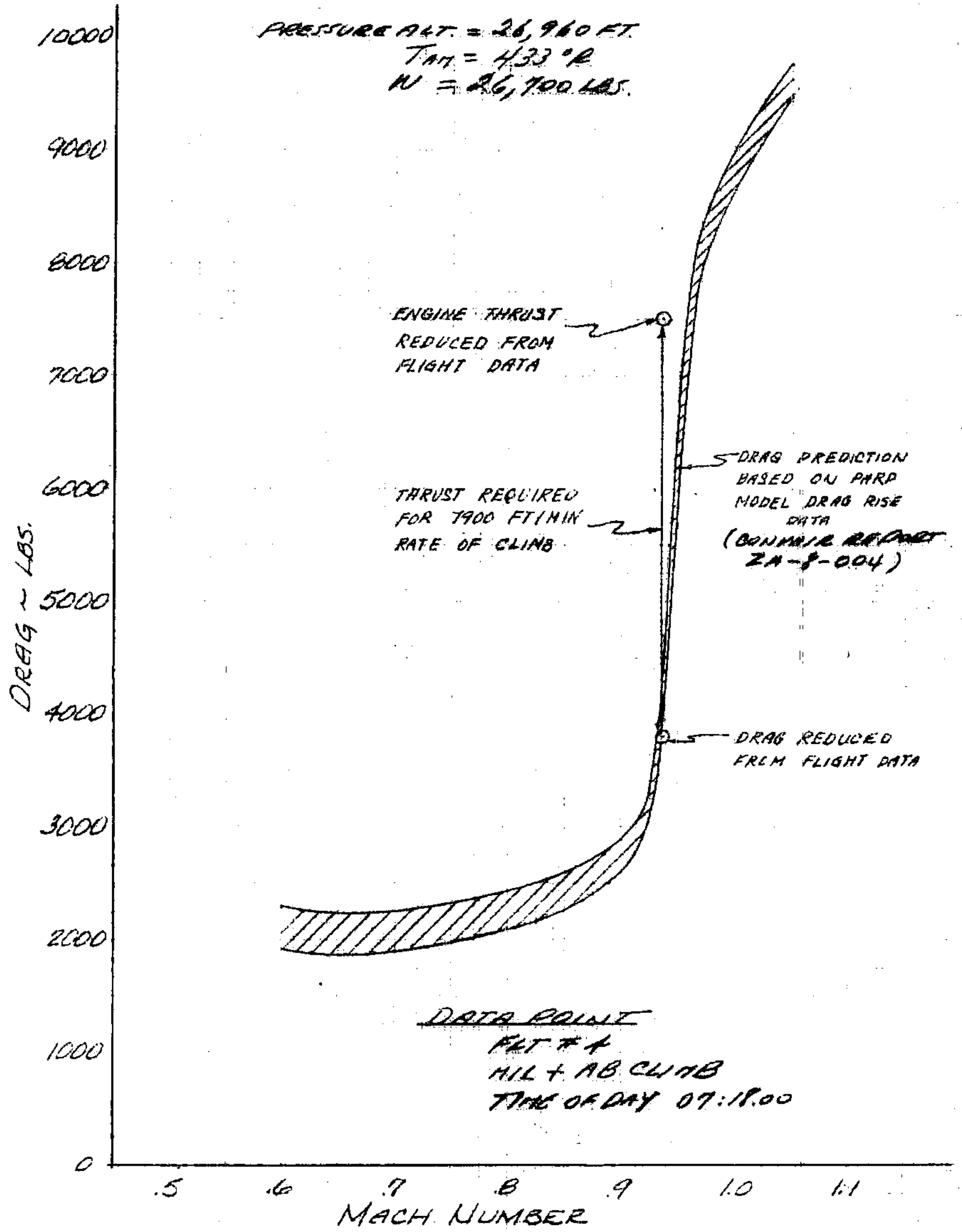
YF-102-AIRPLANE

FLIGHT TEST DRAG CORRELATION



YF102 AIRPLANE

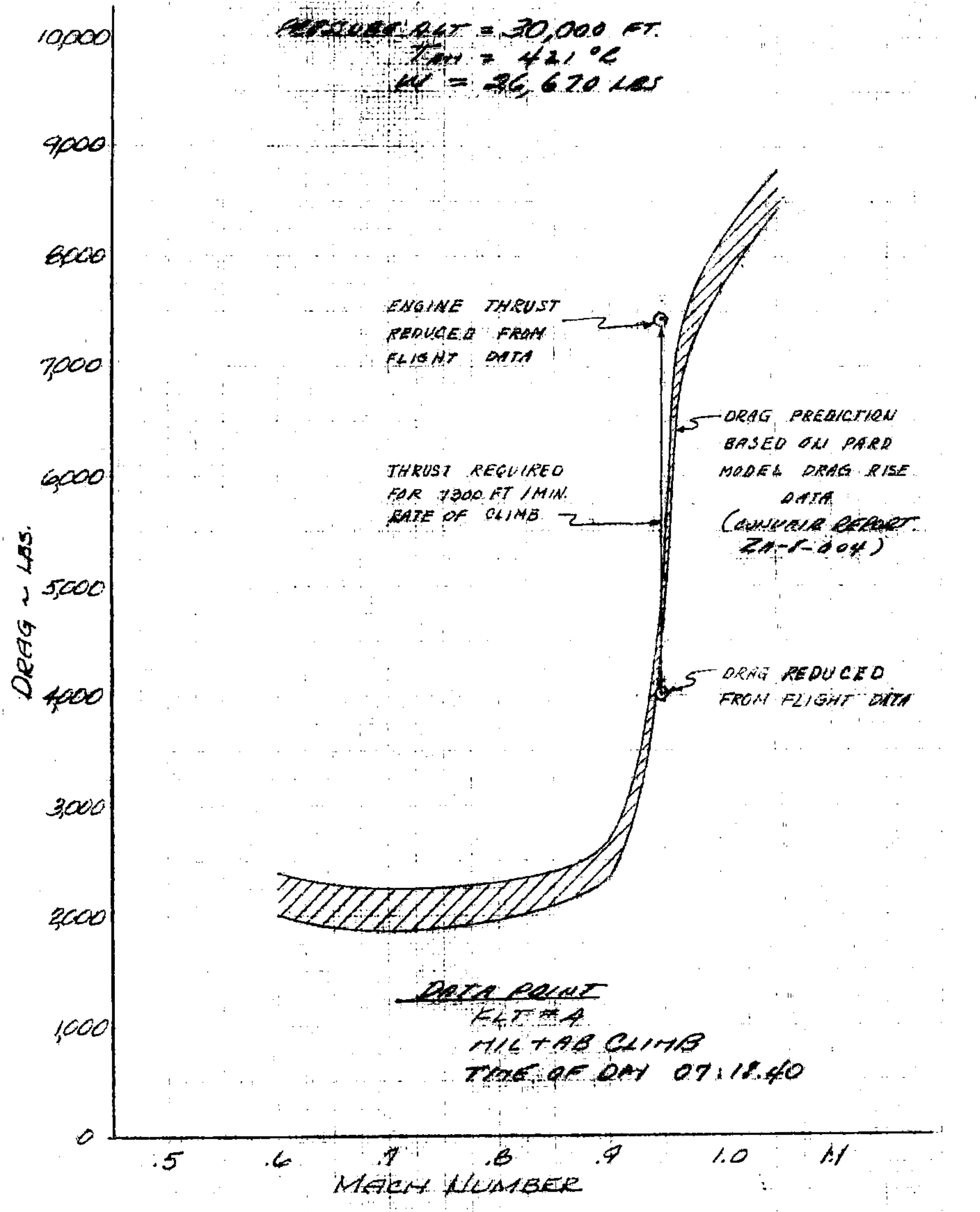
FLIGHT TEST DRAG CORRELATION



YF-102 AIRPLANE

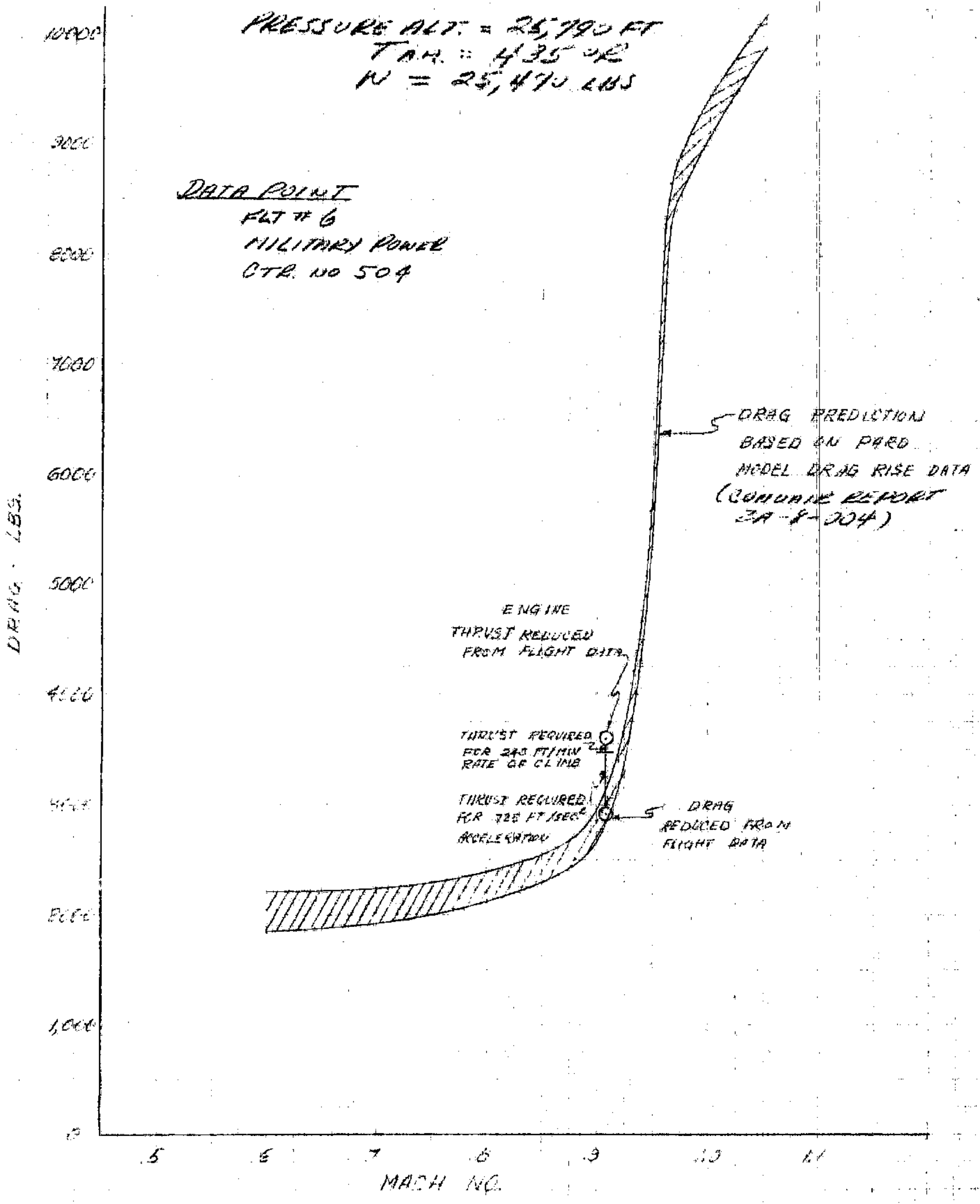
FLIGHT TEST DRAG CORRELATION

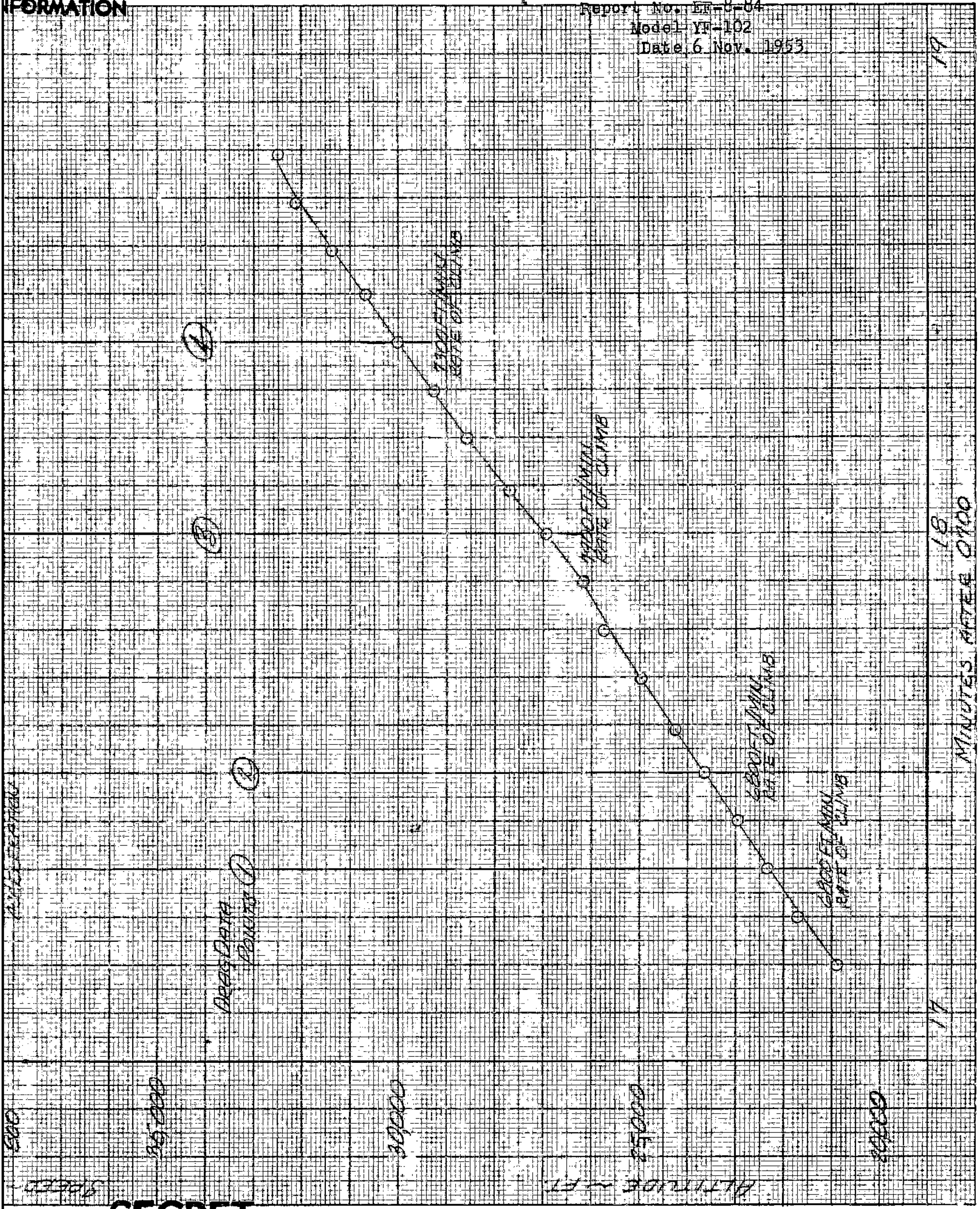
PRESSURE ALT = 30,000 FT.
T_{AM} = 42.1 °C
W = 26,670 LBS



YF102 AIRPLANE

FLIGHT TEST DRAG CORRELATION

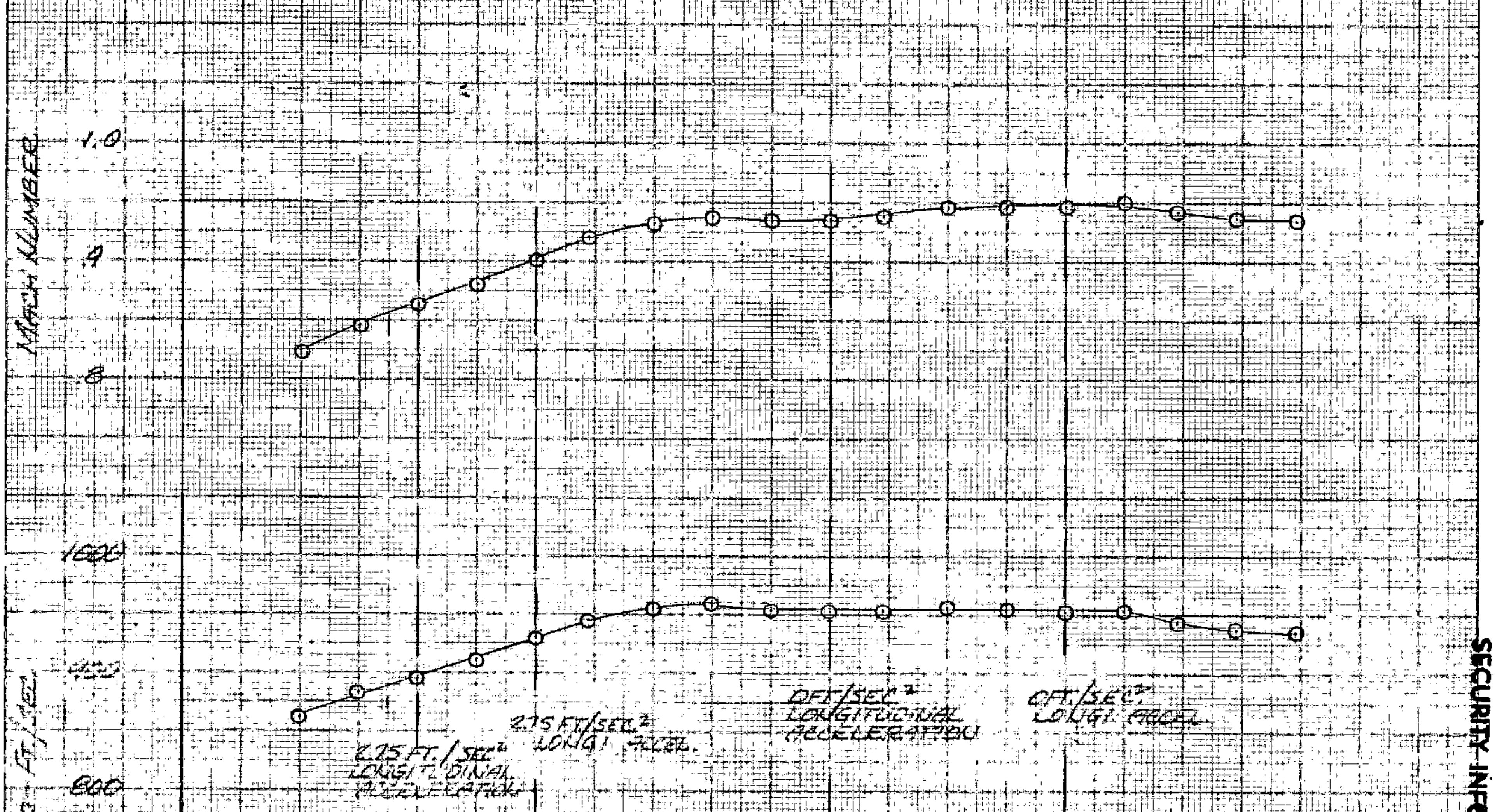



ALTITUDE ~ FT.
20000
25000
30000
35000
SPEED
MINUTES AFTER 0700
17
18
19
DRAG DATA
POINTS ①
②
③
④
7300 FT/MIN
RATE OF CLIMB
7400 FT/MIN
RATE OF CLIMB
6800 FT/MIN
RATE OF CLIMB
6600 FT/MIN
RATE OF CLIMB

359-14L KEUFFEL & ESSER CO
Millimeters, 5 mm lines accented, cm. lines heavy.
MADE IN U.S.A.

YF-102 FLIGHT #4

SPEED AND ALTITUDE VARIATION WITH TIME DURING MAXIMUM POWER CLIMB TO 35,000 FT. ALTITUDE



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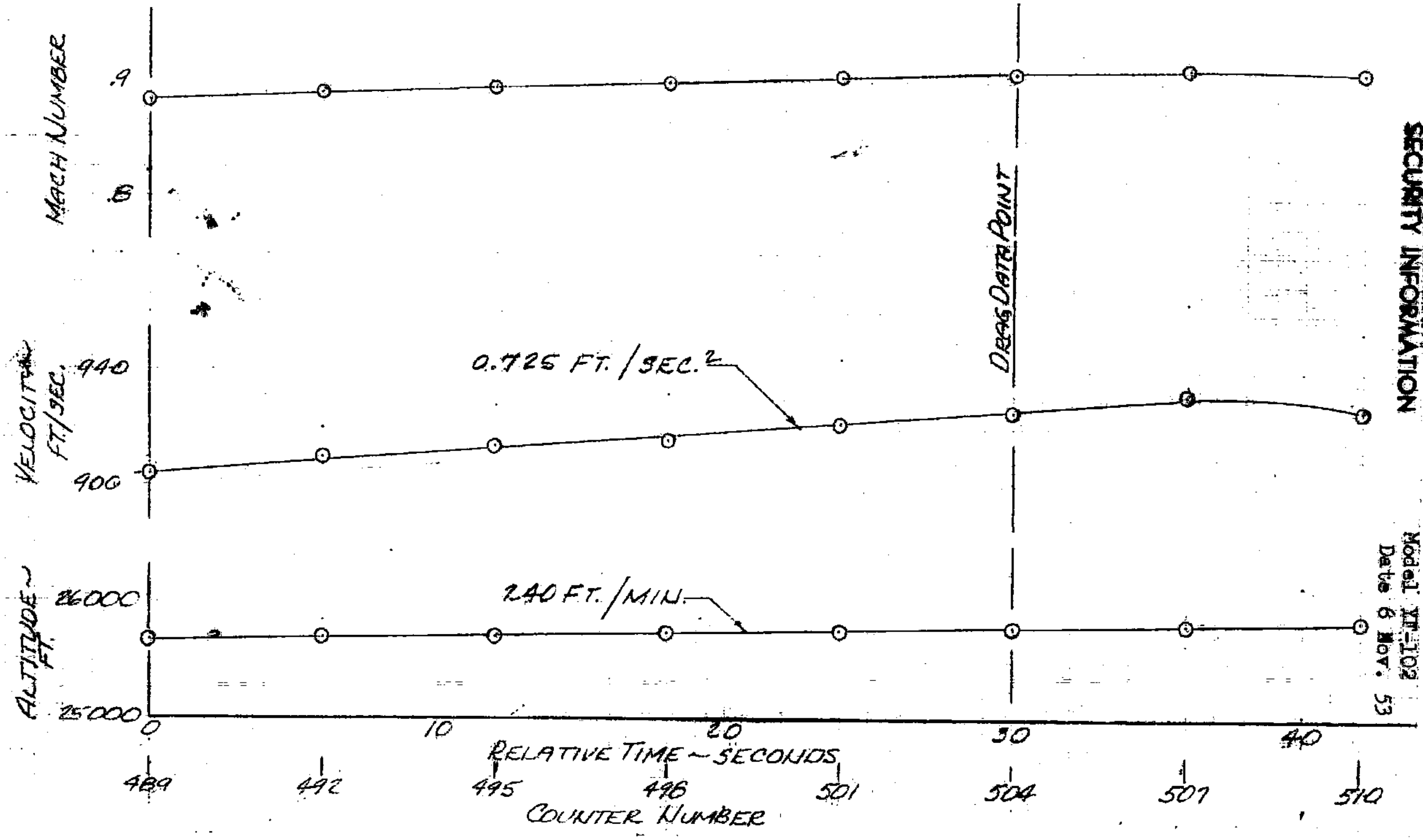
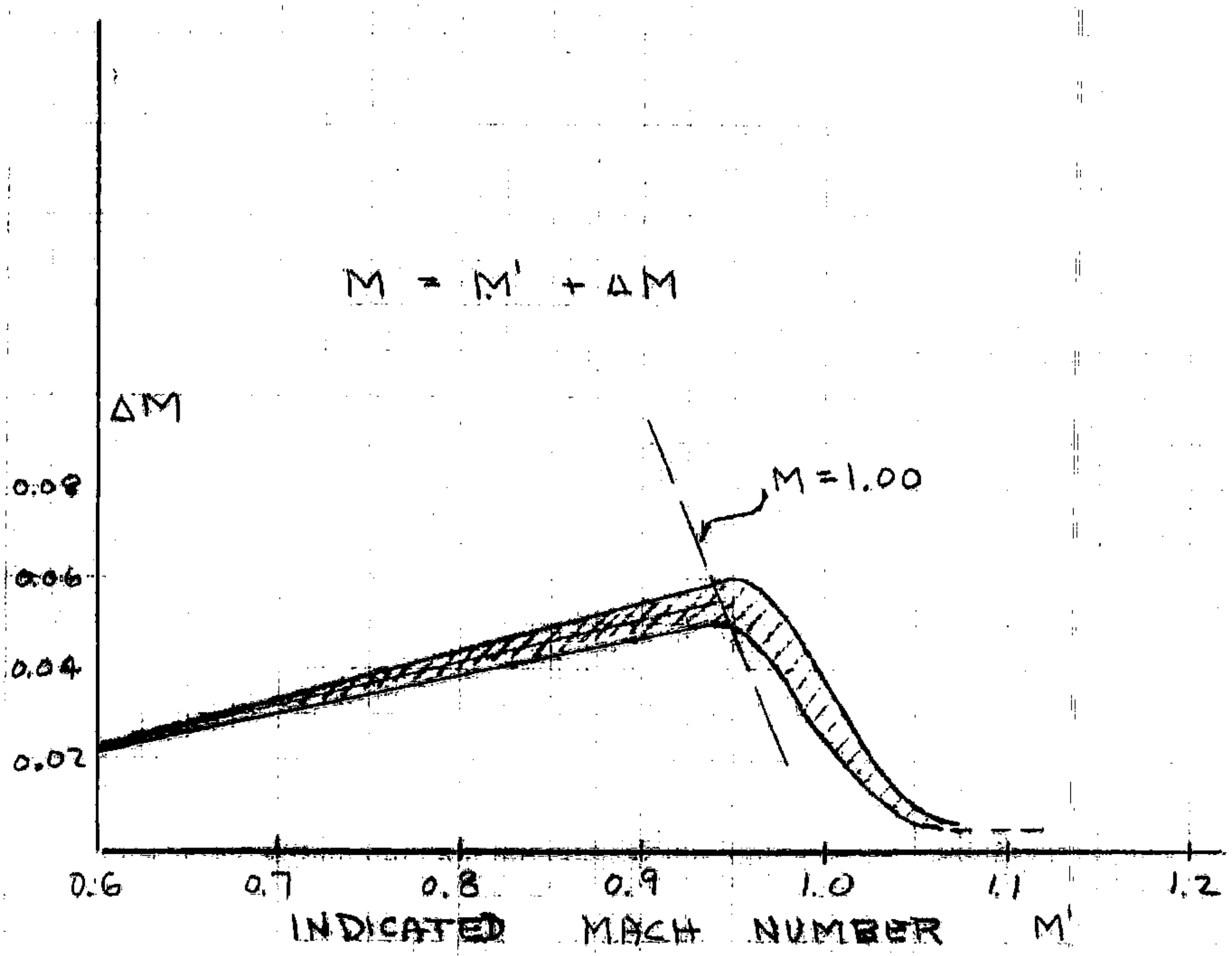
YF-102 FLIGHT #6

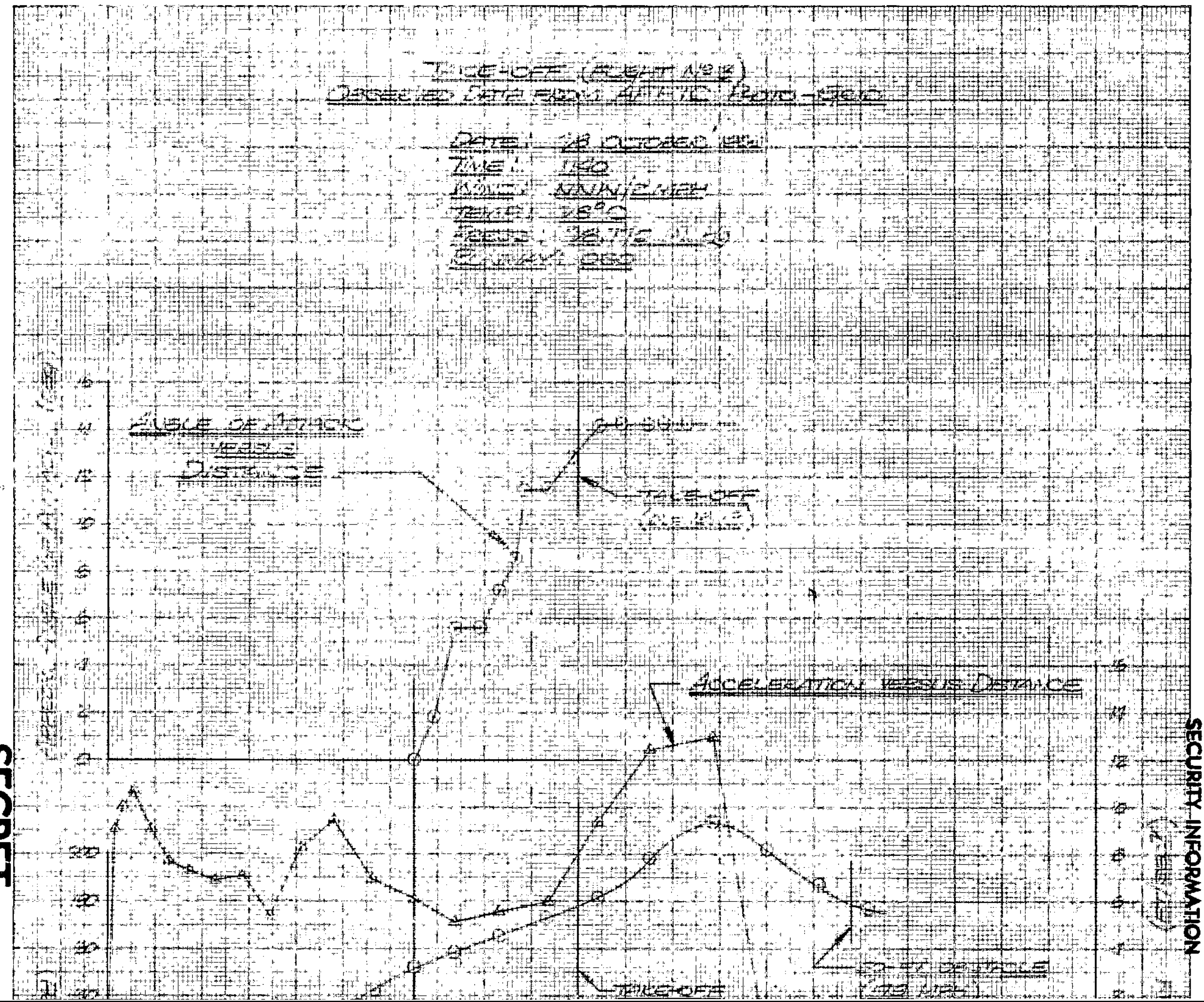
SPEED AND ALTITUDE VARIATION WITH TIME DURING MILITARY POWER SPEED RUN





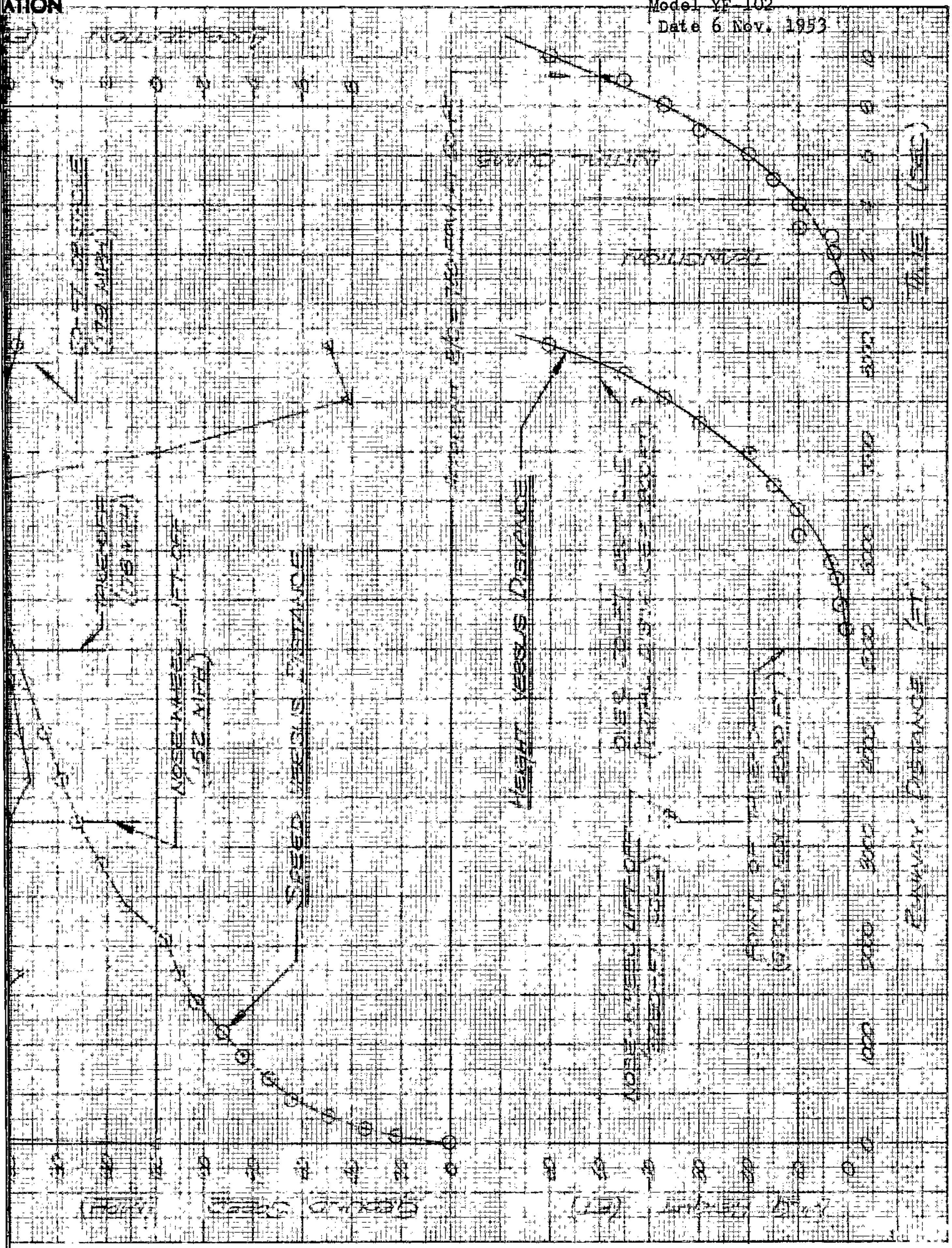
ESTIMATED AIRSPEED POSITION ERROR
YF-102

CONVAIR
29 OCT 1953
G.L.S.



TAKE-OFF (FLIGHT No 12)
CORRECTED DATA FROM AFFTC PHOTO-THEO

DATE: 28 OCTOBER 19[illegible]
TIME: 1150
WIND: NNW/2 MPH
TEMP: 15°C
PRESS: [illegible]
RUNWAY: [illegible]


SPEED VERSUS DISTANCE
HEIGHT VERSUS DISTANCE
NOSE-WHEEL LIFT-OFF (152 MPH)
TAKE-OFF (78 MPH)
50-FT OBSTACLE (173 MPH)
NOSE-WHEEL LIFT-OFF
POINT OF TAKE-OFF
CLEAR 50-FT OBSTACLE
RUNWAY DISTANCE (FT)
TIME (SEC)
0
1000
2000
3000
4000
5000
6000
7000
8000
0
2
4
6
8
10

POWER PLANT CHARACTERISTICS

The powerplant of the YF-102 Serial 52-7994 consists of serial 420111 Pratt and Whitney YJ57-P-11 engine with afterburner equipped with a Bendix type AJ-A2 hydromechanical fuel control. This engine is rated at sea-level static output, NACA standard conditions as: (Pratt and Whitney Aircraft Specification No. 1637)

	Jet Thrust
Maximum Rating (with afterburning)	14,800 lbs.
Military Rating (without afterburning)	9,220 lbs.

and is described as "continuous flow jet propulsion engine with multi-stage (twin-spool) compressor driven by a multi-stage reaction turbine equipped with an afterburner and a two position jet nozzle. Serial 420111 engine was rated at maximum high compressor rotor speed of 9660 RPM (N_2) at sea-level NACA standard day conditions.

OPERATION AND PERFORMANCE

At no time during the flight test program of the subject airplane did the engine reach its rated speed or turbine discharge total temperature in military and/or maximum afterburning power settings. Prior to flights 4 & 5, small adjustments in top RPM were made by Pratt and Whitney representatives with no appreciable results. Engine run No. 5, held on 30 October 1953, consisted of a 45 minute attempt to adjust the AJ-A2 fuel control for maximum engine speed and turbine discharge temperature. No gain in either of these two parameters was realized. The following tabulation presents either the static or take-off engine condition for military and/or maximum afterburning powers.

MAXIMUM POWER ENGINE DATA

Operation	High Comp. Speed N_2 RPM	Turbine Disch. Tot. Temp. T_{t7} °C	Turbine Disch. Tot. Pressure P_{t7} in. Hg.	Approx. Comp. Inlet Tot. Temp. T_{t2} °C	Indicated Airspeed V_{io} MPH	EAFB Sta. Outside Air Temp. OAT °C
Eng. Run #1	9270	550	--	--	Static	9.0
Eng. Run #2	9450	500	53.0	--	Static	--
Taxi Run #1	9450	--	54.2	--	Static	16.0
Taxi Run #2	No Data					
Flt #1	9280	--	60.30	5	150	14.3
Flt #2	9145	--	55.20	10	150	19.7
Flt #3	9350	445	54.65	10	150	21.1
Flt #4	9250	490	61.94	-5	150	7.8
Flt #5	9410	450	58.85	15	150	25.6
Flt #6	9130	470	60.40	-5	150	7.2

It is noted that rated engine speed and turbine discharge temperature of 9660 RPM and 580°C respectively were never reached at any ground condition. Altitude data further verifies the fact that the engine never realized its rated exhaust temperatures and engine speed at military and/or maximum afterburning power settings. A number of items of interest covering the operation of the engine are noted below. During flight #4, at approximately 35,000 feet altitude, an afterburning light up was attempted from military power; following an afterburning climb to approximately 37,000 feet altitude. The pilot reported that two pronounced shocks were felt in the whole airplane resulting from what appeared as a rough light-up. The pilot immediately reduced power to idle and returned to the field. Post-flight inspection revealed no damage to the engine or component parts.

Flight 6 climbout was conducted in afterburning followed by military power with maximum engine RPM of approximately 9100 (N_2) with fuel fed from both wing tanks. At approximately 7000 feet, the pilot switched to right wing tank only because of left wing fuel low level warning, and felt a power surge. A photo panel investigation revealed an increase in N_2 speed to approximately 9400 RPM close to the time of the disturbance. On engine runs one and two, pilot retarded power lever in afterburning and noted a "rough" burning indication at approximately 77% N_2, prior to shifting to a non-afterburning power setting.

On flights 1, 2 and 6, a marked decrease was noted in military and subsequent maximum afterburner take-off engine speeds during the take-off run.

TAKE-OFF ENGINE SPEED COMPARISON

	Approx. Ind. Airspeed V_{lo} (MPH)	High Comp. Rotor Speed N_2 (RPM)	Approx. IAS V_{lo} (MPH)	High Comp. Rotor Speed N_2 (RPM)
Flight No. 1	0	9350	185	9150
Flight No. 2	0	9410	160	9150
Flight No. 6	0	9200	185	9080

During this time, compressor discharge total pressure varied ±2%.

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POWERPLANT PERFORMANCE CHARACTERISTICS

Flight	Time	Pressure Altitude	True Air-Speed-Knots	Mach. No.	Flt. Test Calculated Net Thrust	Specification Net Thrust	Test Thrust Spec. Thrust	N_2 RPM	% Rated N_2 (9950 RPM)
4	0717.30	22350	530	.863	8220	9180	.896	9435	94.8
AB	0717.50	23670	551	.903	7800	9120	.855	9445	94.9
Climb	0718.00	26960	563	.936	7510	8450	.889	9465	95.1
	0718.40	30000	563	.947	7440	7710	.965	9465	95.1
4	0713.10	9220	280	.433	5210	5450	.955	9357	94.0
Mil. Power	0715.00	11860	386	.600	4890	5430	.900	9405	94.5
Slow Climb									
4	0705.51	2285	—	—	6600 MIL	6480	1.02	9262	93.1
Take-	0705.60	2285	66	.102	10270 AB	10150	1.01	9262	93.1
off Mil	0705.70	2285	100	.153	10540	10150	1.04	9262	93.1
& AB	0705.96	2285	161	.244	10940	10250	1.07	9262	93.1
Powers	0706.21	2600	187	.278	10620	10300	1.03	9272	93.2
	Approx.								
6	0859.80	25770	546	.904	3630	4400	.825	9425	94.7
Mil.	0900.00	25790	549	.908	3600	4400	.818	9425	94.7
Power									
Level Flight									

NOTE: J57 specification 100% high compressor speed =9950 RPM
J57-P-11 Serial 420111 rated speed 9660 RPM (97.1%) at sea level standard day.

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It is to be noted that actual net thrust was less than specification net thrust (corrected for duct losses) in all cases except flight 4 take-off performance. The discussion of low temperature and speed described above is considered as applicable.

METHOD OF ESTIMATING NET THRUST FROM FLIGHT TEST DATA FOR THE YF-102 AIRPLANE WITH THE J57-P-11 ENGINE

It was planned to determine in-flight thrust by a method described by reference 1. The accuracy of this method is dependent on having a good calibration of the over-all thrust coefficient (Cg) of the nozzle including burner losses, as well as a good airflow calibration. As discussed in reference 1, it was intended to make calibration runs on a thrust stand with airplane ducts installed. This has not been done to date because it was desired to conserve engine time for flight purposes. It was also expected that Pratt & Whitney would supply accurate thrust coefficient data. Pratt & Whitney has supplied data for the JT-3J non-afterburning engine, however the accuracy of these data is questionable for use with the J57-P-11 engine. Therefore a thrust coefficient was estimated based on Pratt and Whitney static test data of the P-11 for use in the non-afterburning regime as shown by Figure 1. The P & W curve labeled JT3J is shown for comparison. The afterburning thrust coefficient also shown in Figure 1 was supplied by P & W for the P-11 engine and was used to estimate afterburning thrust. An airflow curve for the JT3J non-afterburning engine was used for the P-11 engine. Its accuracy is not known.

Procedure for Estimating Thrust:

From Pratt & Whitney Installation Handbook the following equation was given for determining gross thrust:

$$F_g = C_g \psi \, P_{ambient} \, A_{cold} \, \frac{A_{hot}}{A_{cold}}$$

Net thrust is defined as:

$$F_n = F_g - \frac{W_a V_o}{g}$$

Gross thrust parameter, ψ, is found from the jet nozzle expansion ratio as follows:

$$\psi = 1.26 \frac{P_{t7}}{P_{ambient}} - 1$$

Nozzle area for military

$$A_{cold} \frac{A_{hot}}{A_{cold}} = (2.68)\ (1.006) = 2.70 \text{ sq. ft.}$$

Nozzle area for afterburning power

$$A_{cold} \frac{A_{hot}}{A_{cold}} = (4.27)\ (1.065) = 4.55 \text{ sq. ft.}$$

Corrected airflow is a function of effective low rotor speed $\frac{N_1}{\sqrt{\theta_{t2}}}$ from Pratt & Whitney curve INST. 17147.

By definition

$$\sqrt{\theta_{t2}} = \sqrt{\frac{T_{t2}}{520°R}}$$

$$\delta_{t2} = \frac{P_{t2}}{29.92" \text{ Hg}}$$

T_{t2} and P_{t2} are stagnation conditions at the compressor inlet.

JET NOZZLE GROSS THRUST COEFFICIENT

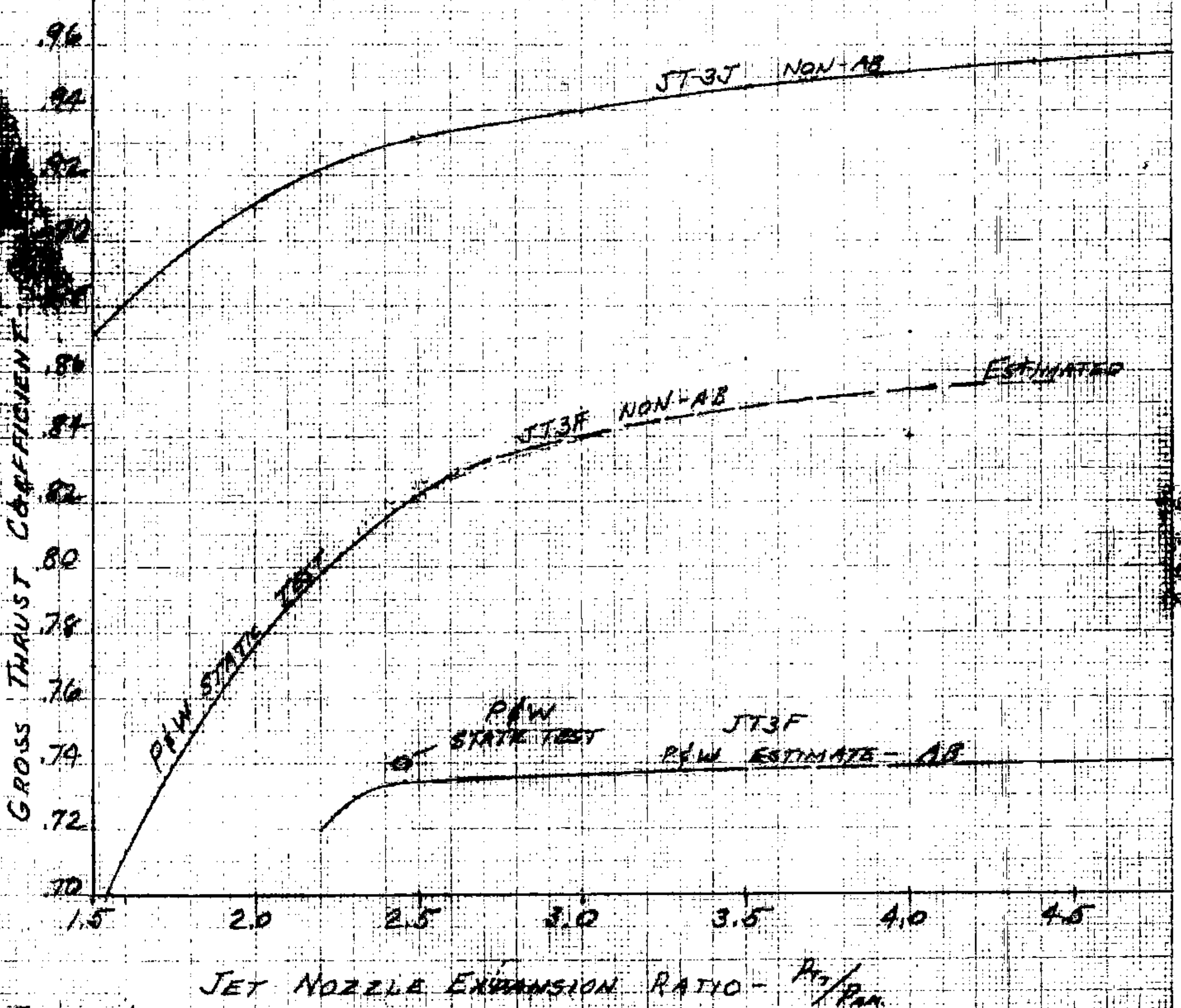

FIGURE 1

EQUIPMENT OPERATION

Equipment operation in general has been very satisfactory for an airplane in this stage of development. Low pressure and low fuel quantity warnings and failure of the main landing gear to retract were the principle difficulties encountered.

FUEL SYSTEM

NOTE: In addition to the normal fuel gages, the fuel system incorporats warning lights, which indicate low tank pressure (air), low fuel booster pump pressure (pump outlet) and low fuel level in the #3 or tank to engine wing compartment. Separate lights are provided for the left wing and right wing since each wing is actually a distinct, separate fuel system; either of which has been proved, by full scale laboratory tests, capable of supplying fuel for full afterburner power operation. Normally both wings supply fuel to the engine, but a shut-off is provided in each to permit pilot selection if desired.

In the initial flights, the left wing low level warning light, left wing tank low pressure warning light and left wing booster pump low pressure warning light came on during or shortly after take off. The low level warning light indicates that 88 gallons of useable fuel remains in the tank. The difficulty was believed to be a fuel transfer problem at high flow rates, and an operating procedure was devised in which the pilot would select the right wing tank only when left wing tank low level warning light came on. This procedure was used on flights 4, 5 and 6. On flight 4, the left wing tank low level warning light went out about five and one-half minutes after the pilot switched to the right tank only. On flight No. 6, however, the left wing tank low level warning light remained on even though the right wing tank only was selected by the pilot. Post flight inspection of the left wing tank low level warning switch revealed a possible mal-functioning switch and a new switch was installed.

The main landing gear either failed to lock up or failed to indicate locked up on flights 1 through 5. Prior to flight 6, a boost cylinder was installed to increase the hydraulic pressure to the main landing gear retraction cylinders and satisfactory landing gear retraction was obtained.

A Revision
9 November 1953

REFERENCES

1. Convair Report ZC-8-051, "Test Instrumentation"

2. Convair Report ZU-8-010, "YF-102 and F-102A Flutter Analysis"

3. Convair Report ZU-8-011, "Results of Ground Vibration Tests on the YF-102 Airplane"

4. Convair Report ZA-8-004, "Predicted Performance Characteristics of the YF-102 Airplane"

5. Convair YF-102 Flight Status Reports for Flights 1 through 6.

6. Convair Report ZA-8-003, "Predicted Flight Characteristics of the YF-102 Airplane"

7. Memo Dunholter to Shannon 10/9/53, "Thrust Cooling Drag & Base Drag Evaluation of the F-102 Airplanes Serial #52-7994 and #52-7995."

8. CVAC Report RT-143, Model 8 Powerplant Test Thrust Cooling and Performance Tests, Pratt & Whitney Aircraft YJ57-P-11 Turbojet, dated 9/18/53.

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CONVAIR

EFT-A-8-[illegible]

A Division of General Dynamics Corporation
(San Diego)

UNCLASSIFIED

Date: 18 February 1957

To: T. L. Maloy

From: Engineering Flight Test Reporting and Analysis Group

Subject: Reporting and Analysis Group Weekly Progress Report for Week Ending 15 February 1957

CURRENT DATA IN WORK

1. External wing tank jettison tests (S/N 53-1791)
2. External wing tank venting and fuel transfer tests (S/N 53-1791)
3. Combustion starter tests (S/N 54-1380)
4. Hydro-Electric fuel flow joiner evaluation (S/N 54-1380)
5. Afterburner relight tests (S/N 54-1380)
6. Edison fire detector system (S/N 54-1380)
7. Emergency fuel control system operation (S/N 54-1380)
8. Speed brake effectiveness evaluation (S/N 54-1390)
9. Take-off and landing performance (S/N 54-1390)
10. Engine response during missile firing (S/N 54-1354)

11. Radar tracking during spins (S/N 54-1401)
12. Evaluation of engine parameters (S/N 56-451)
13. Take-off and landing performance (S/N 56-451)
14. YJ75-P-1 Engine flight thrust determination (S/N 56-451)
15. Engine thrust run and engine performance (S/N 56-45[illegible])
16. Spin entry tests (S/N 54-1401)
17. δ_e/g and F_s/g tests (S/N 54-1390)
18. Buffet investigation with aft duct fairing (S/N 54-1351)
19. Instrument panel vibration survey (S/N 54-1351 and 54-1388)
20. Telemetered data - data station filter utilization study (S/N 54-1390)

21. Aircraft vibration during taxi and take-off (C.G. accelerometer) (S/N 56-451)
22. Stratos unit vibration (S/N 54-1388)
23. Armament door flutter investigation with debris deflectors (S/N 53-1793)
24. In-flight flutter tests (S/N 56-451)
25. Spin maneuver flight data (S/N 54-1401)
26. Wing camera film analysis of wing twist (S/N 53-1794)
27. Preliminary digital test problems for shear moment and torque (S/N 53-1794 & 56-455)
28. Instrumentation evaluation on S/N 53-1794, Flight 53, and S/N 54-1398 Flights 9, 10, 12, 13, and 15.
29. Drag chute deployment tests (S/N 54-1390)
30. Hamilton Standard Refrigeration Package Tests (S/N 54-1390)

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
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Reclassified by: Dept. 130-1 Date 4-4-67

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EFT-A-8-366

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

Date: 15 February 1957

To: T. L. Maloy

From: Engineering Flight Test Reporting and Analysis Group

Subject: Reporting and Analysis Group Weekly Progress Report for Week Ending 15 February 1957

CURRENT DATA IN WORK

1. External wing tank jettison tests (S/N 53-1791)
2. External wing tank venting and fuel transfer tests (S/N 53-1791)
3. Combustion starter tests (S/N 54-1380)
4. Hydro-Electric fuel flow joiner evaluation (S/N 54-1380)
5. Afterburner relight tests (S/N 54-1380)
6. [illegible] fire detector system (S/N 54-1380)
7. Emergency fuel control system operation (S/N 54-1380)
8. Speed brake effectiveness evaluation (S/N 54-1390)
9. Take-off and landing performance (S/N 54-1380)
10. Engine response during missile firing (S/N 54-1354)
11. Radar tracking during spins (S/N 54-1401)
12. Evaluation of engine parameters (S/N 56-451)
13. Take-off and landing performance (S/N 56-451)
14. YJ75-P-1 engine flight thrust determination (S/N 56-451)
15. Engine thrust run and engine performance (S/N 56-452)
16. Spin entry tests (S/N 54-1401)
17. δ_e/g and F_s/g tests (S/N 54-1380)
18. Buffet investigation with aft duct fairing (S/N 54-1351)
19. Instrument panel vibration survey (S/N 54-1351 and 54-1393)
20. Telemetered data - data station filter utilization study (S/N 54-1389)
21. Aircraft vibration during taxi and take-off (C.G. accelerometer) (S/N 56-451)
22. Stratos unit vibration (S/N 54-1388)
23. Armament door flutter investigation with debris deflectors (S/N 53-1793)
24. In-flight flutter tests (S/N 56-451)
25. Spin maneuver flight data (S/N 54-1401)
26. Wing camera film analysis of wing twist (S/N 53-1794)
27. Preliminary digital test problem for shear moment and torque (S/N 53-1794 & 56-455)
28. Instrumentation evaluation on S/N 53-1794, Flight 53, and S/N 54-1393 Flights 9, 10, 12, 13, and 15.
29. Drag chute deployment tests (S/N 54-1380)
30. Hamilton Standard Refrigeration Package Tests (S/N 54-1390)

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CURRENT DATA IN WORK
(Continued)

31. Hydraulic system pressures and temperatures (S/N 56-451)
32. Stratos Refrigeration Unit heat and vent tests (S/N 54-1388)
33. Engine inlet duct anti-icing tests (S/N 54-1354)
34. 2.75 Inch rocket jump angle (S/N 53-1793)
35. Launch Error at missile motor burnout (S/N 54-1354)
36. MB-1 rocket trajectory (S/N 53-1797)
37. Eject -10 qualifying tests with over-center latch (S/N 53-1806)

DISCUSSION OF DATA

1. Engine Trim Check With Inlet Screens Installed (S/N 54-1380) - Engine runs were made on the F-102A airplane S/N 54-1380 to determine the effect upon engine trim of adding screens at the air inlet duct. Data were obtained throughout an ambient air temperature range of -3.5°C to 8°C. The exhaust nozzle pressure ratio was reduced approximately 10% to 20% due to the addition of the inlet screens.

   Previously obtained data cannot be used with these data since the engine was retrimmed prior to this test.

2. Flight Flutter Tests (S/N 56-451) - Test point numbers 5, 6, 7, 11, 12, 13, and 14 of Dynamics' Memorandum DF-8B-114, Revision II were flown on Flights 4, 5, and 6. Satisfactory damping characteristics were observed for these flight conditions.

3. Preliminary Digital Test Problems (S/N 53-1794 and S/N 56-455) - Actual flight pressure data on S/N 53-1794, Flight 49 (simulating S/N 56-455) has been converted to wing shear, moment, torque, and hinge-moment. Data has been checked to within 0.2% of hand calculated values on the IBM 650 Digital Computer.

4. Heating and Ventillation Tests (S/N 54-1388) - During all high speed dives telemetering has been very noisy, obscuring the data.

   Stratos unit speed has reached approximately 80,000 rpm.

   The cabin and forward electronic compartment airflows are still questionable.

5. Hamilton Standard Refrigeration Unit Test (S/N 54-1390) - Preliminary analysis of turbine speed indicates that the unit operates at approximately 80,000 rpm for most flight conditions.

6. Hydraulic System Temperatures and Pressures (S/N 56-451) - Data presently indicates that the normal pressure transients that occur in the primary hydraulic system are well within limits. The secondary system has an operating pressureof 3500 psi which is about 400 psi higher than normal. The transients that occur during operation often exceed the limit of 4050 psi. A need for reducing the secondary hydraulic pressure to 3000 psi is indicated.

CURRENT DATA IN WORK
(Continued)

31. Hydraulic system pressures and temperatures (S/N 56-451)
32. Stratos [illegible] Refrigeration Unit heat and vent tests (S/N 54-1388)
33. Engine inlet duct anti-icing tests (S/N 54-1354)
34. 2.75 inch rocket [illegible] angle (S/N 53-1793)
35. Launch effect on missile motor harness (S/N 54-1354)
36. MB-1 rocket trajectory (S/N 53-1797)
37. Eject -10 qualifying tests with over-center latch (S/N 53-1808)

DISCUSSION OF DATA

1. Engine Trim Check With Inlet Screens Installed (S/N 54-1380) - Engine runs were made on the F-102A airplane S/N 54-1380 to determine the effect upon engine trim of adding screens at the air inlet duct. Data were obtained throughout an ambient air temperature range of -3.5°C to 8°C. The exhaust nozzle pressure ratio was reduced approximately 10% to 20% due to the addition of the inlet screens.

   Previously obtained data cannot be used with these data since the engine was retrimmed prior to this test.

2. Flight Flutter Tests (S/N 56-451) - Test point numbers 5, 6, 7, 11, 12, 13, and 14 of Dynamics' Memorandum MF-55-114, Revision II were flown on Flights 4, 5, and 6. Satisfactory damping characteristics were observed for these flight conditions.

3. Preliminary Digital Test Problem (S/N 53-1794 and S/N 56-455) - Actual flight pressure data on S/N 53-1794, Flight 49 (simulating S/N 56-455) has been converted to wing-shear, moment, torque, and hinge-moment. Data has been checked to within 0.2% of hand calculated values on the IBM 650 Digital Computer.

4. Heating and Ventilation Tests (S/N 54-1388) - During all high speed dives telemetering has been very noisy, obscuring the data.

   Stratos unit speed has reached approximately 80,000 rpm.

   The cabin and forward electronic compartment airflows are still questionable.

5. Hamilton Standard Refrigeration Unit Test (S/N 54-1390) - Preliminary analysis of turbine speed indicates that the unit operates at approximately 80,000 rpm for most flight conditions.

6. Hydraulic System Temperatures and Pressures (S/N 56-451) - Data presently indicates that the normal pressure transients that occur in the primary hydraulic system are well within limits. The secondary system has an operating pressure of 3400 psi which is about 400 psi higher than normal. The transients that occur during operation often exceed the limit of 4050 psi. A need for reducing the secondary hydraulic pressure to 3000 psi is indicated.

DISCUSSION OF DATA
(Continued)

7. Eject -10 Qualifying Tests - Analysis of thirty-one consecutive ejections with the redesigned latch mechanism at San Diego shows all pressures to be within acceptable limits and ejection system operation to be normal and repeatable. The average mean peak cylinder pressure was 4272 psi and all but five ejections were within the 4250 ± 200 psi band required for 1σ (68%) of the qualifying ejections. All pressures fell within the 3800 - 4700 psi absolute limit. The forward hook released first in all cases by 0.5 - 2.0 milliseconds. The time interval from latch first movement to rocket first movement varied only from 12.0 - 17.0 milliseconds. Instrumentation for these 31 operations, conducted in 3 days, was 100% effective.

WORK PUBLISHED

Reports

Airplane S/N	Title	Report Number
54-1380	Afterburner Relight Tests	*ZC-8-512-101
54-1380	Emergency Landing Gear Operation and Evaluation of Main Landing Gear Fairing Link Loads	EFT-A-8-362
54-1390	F-102A S/N 54-1390, Landing Gear Load Tests	EFT-A-8-361
54-1390	Drag Chute Deployment Times for Airplanes With Large Speed Brakes	*ZC-8-512-101
54-1401	Film - Deployment of the 24' Ring-and-Slot Spin Recovery Parachute	
54-1401	Spin Test	*ZC-8-512-101
56-451	Results of Flight Flutter Tests	*ZC-8-512-101
	Qualification of Eject -10 Ejection System With the Redesigned Latch Mechanism	EFT-A-8-364

*Flight Operations Report F-102 and F-106 Development Programs for Week Ending 10 February 1957, dated 15 February 1957

Preliminary Graphs

EF-8-1793-124, GAR-1 Missile Relative Separation Distance
EF-8-1793-125, Summary of GAR-1 Missile Launches
EF-8-1793-94 , Missile Trajectory - Zero Wind Conditions, Flight 47
EF-8-1793-95 , Missile Trajectory - Zero Wind Conditions, Flight 46
EF-8-1794-602,-608, 650 - 658, 664 - 667, 668, 669, 670 - 679, 683 - 685, Time Histories Flights 36, 38, 39, 49, and 50
EF-8-1794-661, 683 - 685, Maneuver Point Data Sheets, Flights 49 and 52
EF-8-1794-662, 663, Accelerometer Check, Flight 52

DISCUSSION OF DATA
(Continued)

7. Eject-10 qualifying Tests - Analysis of thirty-one consecutive ejections with the redesigned latch mechanism at San Diego shows all pressures to be within acceptable limits and ejection system operation to be normal and repeatable. The average mean peak cylinder pressure was [illegible] psi and all but five ejections were within the 4250 ± 200 psi band required for [illegible] of the qualifying ejections. All pressures fell within the 3600 - 4700 psi absolute limits. The [illegible] fired first in all cases by [illegible] - 2.0 milliseconds. The time interval from [illegible] first movement to rocket first movement varied only from 15.0 - 17.0 milliseconds. Instrumentation for these 31 operations, conducted in 3 days, was 100% effective.

WORK PUBLISHED

Reports

Airplane S/N	Title	Report Number
54-1360	Afterburner Relight Tests	ZC-8-512-101
54-1360	Emergency Landing Gear Operation and Evaluation of Main Landing Gear Fairing Link Loads	EFT-A-8-362
54-1360	F-106A S/N 54-1360, Landing Gear Load Tests	EFT-A-8-361
54-1390	Drag Chute Deployment Times for Airplanes With Large Speed Brakes	ZC-8-512-101
54-1401	Film - Deployment of the 24' Ring-and-Slot Spin Recovery Parachute	
54-1401	Spin Test	ZC-8-512-101
56-451	Results of Flight Flutter Tests	ZC-8-512-101
	Qualification of Eject-10 Ejection System With the Redesigned Latch Mechanism	EFT-A-8-364

Flight Operations Report F-102 and F-106 Development Programs for Week Ending 10 February 1957, dated 15 February 1957

Preliminary Graphs

EF-8-1793-124, GAR-1 Missile Relative Separation Distance
EF-8-1793-126, Summary of GAR-1 Missile Launches
EF-8-1793-94, Missile Trajectory - Zero Wind Conditions, Flight 47
EF-8-1793-95, Missile Trajectory - Zero Wind Conditions, Flight 46
EF-8-1794-602 - 608, 630 - 658, 664 - 667, 668, 669, 670 - 679, 683 - 685, Time Histories Flights 36, 38, 39, 49, and 50
EF-8-1794-661, 683 - 685, Maneuver Point Data Sheets, Flights 49 and 52
EF-8-1794-662, 663, Accelerometer Check, Flight 52

WORK PUBLISHED
(Continued)

Preliminary Graphs

EF-8-1806-91, Comparison of Ejection System Operating Parameters and Allowable Tolerance, Pit Drops, Live and Inert Air Drops
EF-8-1806-92, Eject -10 Qualifying Tests at San Diego, 11 February through 14 February 1957
EF-8-1806-93, Redesigned Aft Release Latch Mechanism
EF-8-1806-94, Summary of Ejection System Operation During Qualifying Tests at San Diego , 11 February through 14 February 1957
EF-8-1380-174, Afterburner Relight Tests
EF-8-1388-791, Heat and Vent Test Preliminary Data Summary
EF-8-1388-792,-795, Flight Histories
EF-8-1388-796, V_{max} Dive
EF-8-1388-797, Aft Electronics Compartment Airflow
EF-8-1398-56A, 59A, 90 - 92, Maneuver Point Data Sheets, Flights 15 and 12
EF-8-1398-78- 79, 83 - 89, 93, Accelerometer Time History, Flights 10, 12, 13, & 15
EF-8-1401-61 , Airspeed and Altitude During Spin Test, Flight 20
EF-106-451-44, 46, Take-off Performance
EF-106-451-45, 47, Landing
EF-106-451-67, 68, Altitude and Airspeed - Comparison of Photopanel and Telemetering
EF-106-451-73, 74, Engine Data - Flight 4 and 2
EF-106-451-76, Flight History - Flight 5
EF-106-451-69, through 72, Flight Flutter Response Data From Aileron Pulse Excitation
EF-106-451-79, Flight Flutter Response Data From Wing Shaker Sweep Excitation
EF-106-451-80, Flight Flutter Response Data From Wing Shaker Decay Sweep
EF-106-452-10, Estimated Calibration for Upper Rudder Rotational Velocity Pick-up

CURRENT REPORTS IN WORK

Airplane Number	Subject
53-1787	Elevon Position Measurement Techniques (Memo)
53-1787 53-1791	Summary of Buzz Investigation (Final Report)
53-1788	Two-Inch Rocket Jump Angle (Memo)
53-1791	External Wing Tank Venting and Fuel Transfer Evaluation (Memo)
53-1793	Missile Launcher Air Loads (Memo)
53-1794	December Monthly Progress Report
54-1398	January Monthly Progress Report
53-1797	MB-1 Jump Angle (Memo)
53-1797	Procedure for Rocket Launcher Rail Alignment Determination
53-1799	Weapon System Evaluation (Final Report - Joint HAC and CV)
53-1799	Snap-up Performance (Final Report)
53-1806	MB-1 Ejection System Development Testing (Memo)

WORK PUBLISHED
(Continued)

Preliminary Graphs

EF-8-1506-91, [illegible] of Ejection System Operating Parameters and Allowable Tolerance, Pit Drops, Live and Inert Air Drops [illegible]
EF-8-1506-92, [illegible] -10 Qualifying Tests at San Diego, 11 February through 14 [illegible] 1957
[illegible] a
[illegible] Prediction [illegible]
[illegible] Qualifying Tests at
[illegible] February through 14 February 1957
[illegible] Tests
EF-8-1388-751, [illegible] and Vent Test Preliminary Data Summary
EF-8-1388-7?2, [illegible] Flight Histories
EF-8-1388-796, [illegible] Dive.
EF-8-1388-797, Aft Electronics Compartment Airflow.
EF-8-1398-88A, 89A, 90 - 92, [illegible] Flight Data Sheets, Flights 15 and 12 [illegible]
EF-8-1398-78 - 79, 85 - [illegible], 93, Accelerometer Time History, Flights 10, 12, 13, & 15
EF-8-1401-61, Airspeed and Altitude During Spin Test, Flight 20
EF-108-451-44, 46, Take-off Performance
EF-108-451-45, 47, Landing
EF-108-451-67, 68, Altitude and Airspeed - Comparison of Photopanel and Telemetering
EF-108-451-73, 74, Engine Data - Flight 4 and 2
EF-108-451-76, Flight History - Flight 5
EF-108-451-69, through 72, Flight Flutter Response Data From Aileron Pulse Excitation
EF-108-451-79, Flight Flutter Response Data From Wing Shaker Sweep Excitation
EF-108-451-80, Flight Flutter Response Data From Wing Shaker Decay Sweep
EF-108-452-10, Estimated Calibration for Upper Rudder Rotational Velocity Pick-up

CURRENT REPORTS IN WORK

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53-1787 53-1791	Summary of Buzz Investigation (Final Report)
53-1788	Two-Inch Rocket Jump Angle (Memo)
53-1791	External Wing Tank Venting and Fuel Transfer Evaluation (Memo)
53-1793	Missile Launcher Air Loads (Memo)
53-1794 53-1808	December Monthly Progress Report January Monthly Progress Report
53-1797	MB-1 Jump Angle (Memo)
53-1797	Procedure for Rocket Launcher Rail Alignment Determination
53-1799	Weapon System Evaluation (Final Report - Joint HAC and CV)
53-1799	Snap-up Performance (Final Report)
53-1806	MB-1 Ejection System Development Testing (Memo)

CURRENT REPORTS IN WORK
(Continued)

Airplane Number	Subject
53-1813 53-1787	Summary of Investigations Concerning the Roll-Coupling and Lateral Response Characteristics of the F-102 Airplane
53-1813	Sidestick Evaluation (Memo)
54-1383	Roll Coupling Tests (Memo)
54-1384	TF-102A Dynamic Response to Missile Firings (Memo)
54-1380	Landing Gear Loads and Emergency Extension Times (Memo)
54-1380	Status of Cabin Pressure Surge Testing on F-102A S/N 54-1380) (Memo)
54-1380	Edison Fire and Overheat Detector Systems Temperatures (Memo)
54-1380	Combustion Starter Tests (Memo)
54-1380	Hydro-Electric Fuel Flow Joiner Evaluation (Memo)
54-1380	Emergency Fuel Control System Operation (Memo)
54-1386	Evaluation of the Engine Inlet Duct Anti-icing System
54-1386	Aft Electronics Bay Cooling
54-1380	Turn Coordinator Flight Tests With the Production Large Tail Airplane (Memo)
54-1380	Lateral-Directional Stability Tests on Production Large Tail F-102A (Memo)
54-1380	Flight Test Determination of Elevon Hinge Moments (Memo)
54-1380	Estimated Resonant Rudder Oscillations of a Large Tail F-102A Airplane
54-1401	Proposed Method for Evaluation of the MG-10 Fire Control System in the Missile Firing Mode
54-1401	Aiming of Strike Cameras Snap-up Attack (Memo)
54-1401	In-flight Evaluation of Spin Recovery Equipment (Memo)
54-1401	In-flight Deployment of Spin Recovery Parachute (Memo)
54-1401	Stalls and Spin Entry Characteristics (Memo)
R3Y-2	Flight Test Operations for Two Week Peridd Ending 20 January 1957
F-102A	Analysis Techniques of Filtered Telemetery Data as Utilized for Flight Flutter Testing (Memo)

M. I. Edelstein

M. I. Edelstein
Flight Test Group Engineer

MIE:ee

CURRENT REPORTS IN WORK
(Continued)

Airplane Number	Subject
[illegible]-1013 [illegible]-1787	[illegible] of Investigations Concerning the Roll-Coupling and Lateral [illegible] and Characteristics of the F-102 Airplane
[illegible] 1013	[illegible] (Memo)
[illegible]-1388	[illegible] (Memo)
[illegible]-1364	[illegible] to [illegible] (Memo)
[illegible]-1380	[illegible] S/N 53-1380) (Memo)
[illegible]-1380	[illegible] Temperatures (Memo)
[illegible]-1360	[illegible] (Memo)
[illegible]-1360	[illegible] Evaluation (Memo)
[illegible]-1380	[illegible] Control System Operation (Memo)
[illegible]-1398	Evaluation of the Engine Inlet Duct Anti-Icing System, Air [illegible] Cooling
[illegible]-1380	Turn [illegible] Flight Tests With the Production Large Tail Airplane (Me
[illegible]4-1380	Lateral-Directional Stability Tests on Production Large Tail F-102A (Memo
[illegible]4-1380	Flight Test Determination of Elevon Hinge Moments (Memo)
[illegible]4-1380	[illegible] Resonant Rudder Oscillations of a Large Tail F-102A Airplane
[illegible]-1401	Proposed Method for Evaluation of the MG-10 Fire Control System in the [illegible]
[illegible]-1401	[illegible] Shake-up [illegible] (Memo)
[illegible]-1401	In-Flight [illegible] of Spin Recovery Equipment (Memo)
[illegible]-1401	In-Flight Deployment of Spin Recovery Parachute (Memo)
[illegible]-1401	Stall and Spin Entry Characteristics (Memo)
[illegible]Y-2	Flight Test Operations for Two Week Period Ending 20 January 1957
F-102A	Analysis Techniques of Filtered Telemetry Data as Utilized for Flight Flutter Testing (Memo)

M. I. [illegible]
Flight Test Group Engineer

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SECURITY INFORMATION

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CONSOLIDATED VULTEE

DIVISION San Diego

MODEL 8 (YF-102)

REPORT ZD-8-101

DATE 7 August 1953

TITLE

MODEL YF-102 SERIAL NO. 52-7994

FLIGHT TEST PROGRAM

DOWNGRADED GR. 4
AT 3 YEAR INTERVALS; DECLASSIFIED AFTER
12 YEARS
DOD DIR 5200.10

SUBMITTED UNDER

CONTRACT NO. AF 33(600)-5942

PREPARED BY R. W. Bonney
R. W. Bonney
Flight Test Engineer

GROUP Experimental Flight

REFERENCE ZD-8-001

CHECKED BY R. W. Bonney
for G. W. Hofeller
Sr. Flight Test Engineer

APPROVED BY T. L. Maloy
T. L. Maloy
Group Engineer

Approved By
E. D. Shannon
Chief of Experimental Flight

NO OF PAGES

NO OF DIAGRAMS

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FOREWORD

This flight test program describes the testing to be conducted by Convair on the first YF-102 airplane, Serial No. 52-7994. The tests will be conducted during the period of October 1953 through March 1955 at the Air Force Flight Test Center at Edwards Air Force Base, Edwards, California and at Miramar Naval Air Station at San Diego, California. A total of 114 flight hours will be required to accomplish this program. The purpose of these tests is to demonstrate and evaluate the basic flying and operating qualities of the YF-102 airplane. This program is authorized by Air Force Contract AF33(600)-5942, Supplemental Agreement No. 3. This program will, when approved, supersede all previous proposed programs for flight testing of YF-102 airplane Serial No. 52-7994.

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SCOPE

The F-102 airplane is a single place, land based, high altitude, all weather interceptor. The primary mission of the airplane is the interception and destruction of attacking enemy aircraft and airborne missiles which fall within the performance capabilities of the aircraft, its armament and associated electronics equipment under all conditions of weather and visibility and to fulfill the requirements for general, local, and free lance defense operations for aircraft of this type.

Unique features of the F-102 airplane are the 60° delta wing planform, the Hughes MX-1179 or MG-3 fire control system, and the irreversible, fully hydraulic-powered control system having only three movable surfaces.

The YF-102 airplane, Serial No. 52-7994, for which this flight test program has been compiled is the first prototype F-102 airplane to be constructed. It is powered by a Pratt & Whitney J-57-P-11 turbojet engine with afterburner. Adequate instrumentation has been installed in the airplane to measure and record the necessary data for evaluation of the various tests which will be planned. This airplane will not have a fire control system installed.

This flight test program complies with Phase I tests in accordance with AFR 80-14 of 11 September 1951. It has been designed to demonstrate and evaluate the air-worthiness of the airplane, equipment and systems. Preliminary airplane performance data will also be obtained concurrently with the above tests.

A list of the flight tests included in this program are presented on page 5 of this report.

A further breakdown of the test items stating the purpose, requirements procedure for each test has been made and is presented on pages 6 through 39 of this report. The airplane instrumentation required to measure the necessary data during each test has been tabulated in functional groups according to the type of test involved and is presented in this manner on pages 40 through 46. A tabulation of the various tests, stating the configuration to be tested, measurement groups required to accomplish the test, the estimated flight hours, and the calendar period during which the test is scheduled to be conducted is presented on pages 47 through 51.

Scope (Continued)

In addition, Phase II tests will be conducted on this airplane by the Air Force Flight Test Center as early as practical. The Phase II testing is not included in this program.

SPECIAL TEST EQUIPMENT

Flutter and Vibration Tests

A hydraulic powered vibrator is installed in each wing to produce structural vibrations during flutter and vibration tests. Each vibrator has a frequency range of 5 - 70 cps. The force produced by the vibrators is constant and not a function of frequency.

The vibrators have an automatic sweep over the frequency range and are operable either in phase or 180° out of phase. A stop control is capable of stopping the vibrators within one cycle of operation. The vibrator frequency, phase relationship, and stop control are all controllable from the cockpit. The vibrators are completely submerged within the wing.

Dynamic Response Tests

An electronic scheduling device referred to as "programmer" is installed in the airplane to supply an input signal to the control valve servo amplifier which causes a pre-determined flight control system operation during dynamic response tests. The various flight parameters are set into the "programmer" on the ground to produce a scheduled series of control systems movements of prescribed rate, amplitude and duration during flight. The "programmer" may also be set by the pilot during flight to produce a given control system operation.

The "programmer" will be used during the dynamic stability demonstration tests to produce specified elevon and rudder deflections.

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CONFIGURATIONS AND SYMBOLS

The following configurations and symbols are used in this report and are identical to those used in Air Force Requirements R1815-B.

Configurations:

CR	Cruise	Thrust for level flight at approximately speed for maximum range, speed brakes and gear up.
D	Dive	25% NRT, speed brakes extended.
G	Glide	Idling thrust, unless otherwise specified, speed brakes and gear up.
L	Landing	Idling thrust, gear down.
P	Power-on Clean	NRT, speed brakes and gear up.
PA	Power Approach	Thrust for level flight at $1.20V_{S_L}$, gear down.
WO	Wave-Off	T.O. thrust, gear down.

Symbols:

V_S	True indicated stalling speed for $C_L = 0.75$ at the gross weight corresponding to the specified condition. Subscripts (PA, L, etc.) refer to the airplane configurations as defined above.
Vmax.	Level flight speed with maximum available thrust (full afterburner operation).
Vmax.NRT	Level flight speed with normal rated thrust.

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TESTS TO BE ACCOMPLISHED

1. Low Speed Taxi Tests
2. High Speed Taxi Tests
3. Shakedown Flight Tests
4. a. Airspeed Calibration - Low Speed
   b. Airspeed Calibration - High Speed
5. Angle of Attack and Angle of Yaw Pickup Calibration
6. Flutter and Vibration Tests
7. Preliminary Airplane Performance
8. Dynamic Response Test
9. Preliminary Flight Characteristics Tests
10. Fuel Tank Pressurization System Tests
11. Emergency Ram Air Hydraulic Power Unit Tests
12. Drogue Chute Deployment Tests
13. Fuel Consumption Tests
14. Emergency Fuel System Operation Tests
15. Speed Brake Tests
16. Stability and Control Tests

It is expected that the test program will be conducted in the above listed order if practicable. Some of these tests will be conducted concurrently with others in order to expedite the program.

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Test: Low Speed Taxi Tests CVAC Index No. 1.5

Purpose: To determine ground handling, engine operation engine and structure cooling, and equipment operation.

Requirements:
Engine thrust for "breakout" from standstill.
Engine thrust for low speed taxi.
Maximum engine thrust withheld by brakes.
Maximum engine thrust for nose wheel scrubbing.
Minimum turn radius.
Nose wheel steering and main brake wheel brake operation.
Engine handling characteristics.
Engine and structure pressure and temperatures within design limits.
Acceptable operation of hydraulic, pneumatics, electrical, and other equipment operable on ground.
Airplane handling acceptable in crosswinds up to 30 mph.

Procedure: Taxi airplane at low speeds up to 60 mph V_i. Check items listed in requirements above. Procedure details at pilot's discretion.

Test: High Speed Taxi Tests CVAC Index No. 1.5

Purpose: To determine ground handling, engine and structure cooling, engine operation, fuel system, and equipment operation.

Requirements:
Control system operates smoothly and normally.
Directional control satisfactory in crosswinds up to 30 mph.
With the C.G. at the most forward position attainable with maximum gross weight it shall be possible to lift the nose wheel off the ground at a speed of not more than 1.05 V_{S_L}.
Effective lateral control when required.
All temperatures and pressures to remain within design limits.
Satisfactory and smooth engine operation.
Fuel tank pressures not to exceed 4 1/2 psi.
Normal operation of all operated equipment.

Test: High Speed Taxi Tests (Continued) CVAC Index No. 1.5

Procedure: Taxi the airplane at speeds up to take-off speed. The following special items will be determined:

a. Control system operation.
b. Directional control by use of brakes, nose wheel steering and rudder in crosswinds not exceeding 30 mph. Use each type of control singly and in combination in its operational speed range.
c. Minimum practical speed for nose wheel lift off.
d. High speed lateral control and effectiveness.
e. Brake effectiveness in high performance decelerations.
f. Engine cooling during simulated pre-take-off pilot cockpit check, followed by simulated take-off run.

All other requirements will be measured during these operations by instrumentation and pilot's observation.

Test: Shakedown Flights CVAC Index No. 1.12

Purpose: To determine general flight handling characteristics, engine operation and cooling, structural cooling and equipment operation.

Requirements: Airplane response to flight controls displacement.
Engine and structural cooling.
Engine controls operation.
Low and high pressure pneumatic systems operation.
Hydraulic system operation.
Electrical system operation.
Preliminary air speed system check.
Cockpit cooling.

Procedure: The primary purpose of the shakedown flight will be to conduct the initial flutter and vibration tests as outlined on page 10 of this report.
The pilot will qualitatively check the requirements listed above and record data as required.
The preliminary airspeed system check will consist of verbal airspeed checks between the pilot of the chase airplane and the pilot of the YF-102.

Test: Airspeed Calibration - Low Airspeed Range CVAC Index No. 2.1

Purpose: To determine test nose boom airspeed position error.

Requirements: Since the airspeed system installation on this airplane is not a standard system no specification compliance is necessary. It is required that the airspeed position error be reasonable and that repeatable calibration data can be obtained.

Procedure: Calibrate the airspeed system by pacer method in the following speed ranges:

a. Clean 150 - 350 mph V_i
b. Speed brakes extended 150 - 350 mph V_i
c. Landing gear extended 150 - 310 mph V_i

The altitudes used for these tests are at the pilot's descretion.

Test: Airspeed Calibration - High Speed CVAC Index No. 2.1

Purpose: To determine the test nose boom airspeed position error.

Requirements: Since the airspeed system installation on this airplane is not a standard system no specification compliance is necessary. It is required that the airspeed position error be reasonable and that repeatable calibration data can be obtained.

Procedure: The system is to be calibrated by radar plotting during an atmosphere-surveying climb ($M \leq .85$) and a subsequent high speed dive.

Climb the airplane to service ceiling at approximately $M=.80$. Establish a dive at $M=1.3$ until low altitudes require a pull out. The calibration error will be determined by radar, photo panel and radiosonde data.

Test: Angle of Attack and Angle of Yaw Pickup Calibration CVAC Index No. 5.0

Purpose: (a) Calibration of error of fire control system angle of attack and yaw pickups. (b) Determination of optimum location of pickups.

Test: Angle of Attack and Angle of Yaw Pickup Calibration (Continued) CVAC Index No. 5.0

Requirements: The attack and yaw pickups will be located in the position for minimum position error consistent with other design criteria.

It is desired that the calibration error be less than 0.25 degree.

Note: Design specifications for these production instruments are to be furnished at a later date.

Procedure: Special flight operations are not required for these calibrations. The errors will be determined by statistical analysis of photo panel data of pickup readings under various conditions of flight including maximum speed and load factor conditions.

Test: Preliminary Airplane Performance Evaluation CVAC Index No. 2.2

Purpose: Determine performance characteristics of the airplane as concerns drag, high speed, rate of climb, and ceilings.

Procedure: Engine thrust and airplane energy data will be obtained from photo panel and telemeter data throughout all early flights on this airplane. This information will be obtained concurrently with the conduction of primary test items such as wing flutter analysis and preliminary stability and control tests.

Note: The engine and instrumentation installed originally for flight will not have been calibrated for thrust measurement. All engine thrust calculations will be based on engine manufacturer's data supplied for this particular engine.

Test: Flutter and Vibration CVAC Index No. 1.8

Purpose: To demonstrate that the airplane has positive damping throughout the speed range.

Requirements: Vibration induced in the wings and tail surfaces in the frequency range of 0-70 cps shall be positively damped throughout the entire speed range of the airplane.

Test: Flutter and Vibration (Continued) CVAC Index No. 1.8

Procedure:

1. Stabilize the airplane at M=.08 at 20,000 ft. P.A. in level flight and turn on the exitation equipment. Sweep through the frequency range and determine the frequencies for resonant vibration. Vibrate at each of these frequencies and then stop the vibrators. Maintain recording until vibrations have damped.

2. Repeat the above under assymetric conditions, ie, with the vibrators operating 180° out of phase.

3. The gross weight at which the above tests will be conducted will be that with approximately 60% ±10% of the total fuel capacity.

4. After analysis of the data from the foregone tests and upon approval of the WADC representative the above tests will be repeated at higher speeds (in the order of approximately 70 mph (T.A.S.) increments) up to the limiting high speed of the airplane-engine combination for which these tests can be accomplished in shallow dives.

Test: Dynamic Response CVAC Index No. 1.7

Purpose: To determine dynamic stability and response of the airplane and control system in manual and automatic modes of flight to programmed control system input signals.

Requirements:

1. The airplane shall exhibit satisfactory response to input signals.

2. Measurements shall be obtained with sufficient accuracy to provide the fire control system manufacturer with design data for automatic flight control.

3. The various parameters for the pitch damper, yaw damper, and turn coordinator will be determined from these tests.

Procedure: The airplane will be trimmed in level flight. A specified signal will be introduced to the control system. The response of the airplane and control system will be recorded during hands-off flight. These signals will be introduced singly or in a programmed series by selection of automatic or manual operation of the programmer. Details of the configurations of test will be specified at a later date.

Test: Preliminary Flight Characteristics Tests (Preliminary Stability and Control) CVAC Index No. 1.12

Purpose: Determine general flight characteristics of the airplane.

Requirements: The airplane shall exhibit safe and reasonable stability and control characteristics. Compliance with all parts of Specification R 1815-B will not be required during these tests since the airplane control system will not be in final configuration at this stage. Compliance with Specification R 1815-B with authorized deviations will be demonstrated during the stability and control tests on pages 16 through 36.

Procedure: The flight characteristics items listed below will be preliminarily evaluated during the early stages of the flight test program. Evaluation will be principally obtained from qualitiative comments and implemented by recorded data.

The following items will be evaluated during the early flights concurrent with other operations and require no special test operations.

1. Effectiveness of controls.
2. Control forces.
3. Trim control operation (rate and sensitivity).

The following flight characteristic items will require that tests be conducted to determine their values. The desired altitude for the tests is 20,000 ft. with the airplane in the clean configuration.

1. Static Longitudinal Stability

   Determine stick force and position stability at three trim speeds in the level flight speed range.

   Trim in level flight at 1.4 V_{S_L}. Vary speed in increments of 20 mph above and below trim speed to maximum practical by use of elevator control only. Maintain altitude within ±2000 ft. Repeat this above at 330 mph (V_i) and at 250 mph (V_i).

2. Dynamic Longitudinal Stability

   Determine at the three trim speeds called out in Item 1 above. Deflect elevator in small increments and record damping.

Test: Preliminary Flight Characteristics Tests (Continued) CVAC Index No. 1.12

Procedure: 3. Static Directional Stability

Determine in wings level yawed flight at approximately 200 mph V_i and 350 V_i, follow procedure outlined in items 3 and 4 under Static Directional Stability, page 25.

4. Dynamic Directional Stability

Determine at the three trim speeds called out in Item 1 above. Follow procedure outlined in Items 3, 4, and 5 under Dynamic Directional Stability page 26.

5. Longitudinal Stick Force

Determine stick force per "G" during diving turns and pull-outs.

6. Dynamic Lateral Stability

Determine at the three trim speeds called out in Item 1 above, follow procedures outlined in Items 3 and 4 under Dynamic Lateral Stability, page 31.

Note: These tests will be conducted in the "Direct Manual Mode" of flight.

During the early testing the following limits are imposed:
Maximum side load = 0.95
Maximum aileron deflection = ±6°
Maximum steady bank angle = 60°
(above M=0.7)

Test: Fuel Tank Pressurization System Tests CVAC Index No. 4.16

Requirements: Fuel tank pressure must be held to $4.25^{+00}_{-.25}$ psig relative to ambient air pressure.

The bleed air entering the tank shall not exceed 200°C.

Procedure: Record fuel tank pressure and fuel temperature (in tank) during flight.

Test: Fuel Tank Pressurization System Tests (Continued) CVAC Index No. 4.16

Procedure: Periodic readings should be obtained throughout the flight to determine pressure regulation with varying fuel quantity.

The desired data will be obtained concurrently with other tests. Fuel tank pressures and bleed air temps will be automatically recorded throughout flights until adequate data are obtained.

Test: Equipment Operation - Emergency Ram Air Hydraulic Power Unit CVAC Index No. 4.1

Purpose: Determine effectiveness of the emergency ram air hydraulic power unit.

Requirements: The emergency hydraulic pump must supply 1000 psi at 6.3 gals./min. at 125 knots indicated airspeed to the primary hydraulic system.

1. Extend the emergency hydraulic pump at approximately 135 mph during a high speed taxi run.
2. Extend the emergency hydraulic pump in level flight at maximum level flight speed at 35,000 ft. P.A.
3. Make a normal descent and land directly after extending pump.

Note: Pump flow and pressure data can be obtained by measurement of pump outlet pressure.

Test: Equipment Operation - Drogue Chute CVAC Index No. 4.2

Purpose: Determine drogue chute effectiveness and operational technique.

Requirements:

1. The drogue chute shall deploy smoothly and immediately without fouling.
2. The drogue chute shall release immediately upon actuation of the release.
3. The drogue chute shall be useable in cross winds up to 30 mph.

Test: Equipment Operation - Drogue Chute(Continued) CVAC Index No. 4.2

Procedure:

1. Deploy the drogue chute while taxiing in level attitude and take-off speed. Record data on the photo panel and with a theodolite.
2. Deploy the drogue chute with the nose wheel off the ground with angles of attack of 10°, 15°, and/or maximum practical. Conduct tests at 0.9 V_{SL} up to approximately 160 mph.
3. Repeat (1) and (2) with a cross wind component not greater than 30 mph.
4. Jettison the drogue chute at approximately 30, 90, 120 and 160 mph V_1.

Note: The tests shown above will be the final tests. Deployment speeds, release speeds and cross winds for initial pilot familiarization will be at the pilot's discretion.

Deployment and jettisoning of the drogue chute will be done with idling thrust.

Test: Power Plant Operation - Fuel Consumption CVAC Index No. 2.9

Purpose: To determine fuel consumption in flight.

Requirements: The fuel consumption of the engine shall conform to the engine manufacturer's specifications.

Procedure: The fuel consumption of the engine will be determined from automatically recorded data without special procedures. This data will be obtained concurrently with other tests and will cover the range of operation of the airplane.

Test: Equipment Operation (Emergency Fuel System) CVAC Index No. 3.3

Purpose: To determine that the emergency fuel system operates satisfactorily.

Requirements: The emergency fuel system shall provide smooth transition from normal to emergency system operation without excessive surging or tailpipe temperatures.

Test: Equipment Operation (Emergency Fuel System) CVAC Index No. 3.3 (Continued)

Requirements: The emergency system shall provide smooth power regulation at all speeds and altitudes.

Procedure: Change from normal to emergency fuel system according to the engine manufacturer's recommended operating procedures and vary power between idle and afterburner regimes. Accomplish at both 5000 and 35,000 feet approximate pressure altitudes.

Test: Equipment Operation - Speed Brakes CVAC Index No. 4.14

Purpose: To determine speed brake operation characteristics.

Requirements:

1. The speed brakes shall have acceptable smoothness and sensitivity.
2. The speed brakes shall not cause more than .5g deceleration at high speeds.
3. Speed brake operation during gear extension shall not cause unfavorable trim characteristics.
4. Satisfactory static and dynamic directional and longitudinal stability must be maintained with speed brakes extended.

Procedure:

1. The pilot will modulate the speed brakes between full closed and open to determine smoothness and sensitivity of operation.

2. The pilot will obtain maximum practical speed in a dive. The speed brake switch will be held to open as the airplane speed decreases until full speed brake position is obtained.

3. The speed brakes will be fully extended and airspeed stabilized at maximum gear extension speed. The landing gear will then be extended and the speed brakes will automatically close.

4. With speed brakes partially extended at 300 mph V_1, the pilot will execute pull-ups, pushovers, and will yaw the airplane in wings level flight up to 5° of yaw.

Test:	Static Longitudinal Stability	CVAC Index No. 1.1A
Applicable Spec:	R1815B Par 4.2.1	
Purpose:	To demonstrate that the airplane has positive static stick-fixed longitudinal stability for the specified conditions with exception of the noted deviation.	
Requirements:	Demonstrate that the airplane has positive static stick-position and force stick, longitudinal stability at all normal service loadings when flying with constant power settings under conditions given in Table I, page 17. Except in configuration D, the existence of such stability shall be proved by determination of the stick-fixed neutral points. The neutral point determined for the condition of most aft C.G. location must be at least 5% of the MAC aft of the most aft C.G. location. In configuration D, the stability shall be indicated by stable variations of elevator control force and angle with indicated airspeed.	
Deviation:	At Mach numbers above 0.85 with the airplane trimmed at M=0.85, the variation of elevator force and deflection with Mach number may be negative if the force changes involved are not greater than 20 lbs. and occur over a reasonably wide Mach number range.	
Procedure:	1. Conduct at altitudes specified in Table I, page 17. 2. Trim airplane at specified speed in configuration to be tested. 3. Stabilize speed in specified increments through speed range by use of elevator control only. 4. Determine δ_e and F_e vs. airspeed or Mach number.	
Remarks:	Since C.G. position varies during flight, the data should be taken for those loadings which determine the most forward and most aft C.G. position possible during flight, i.e., to obtain C.G. variation necessary to determine the longitudinal neutral point.	

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TABLE I

REQUIRED CONDITIONS FOR STABILITY

Configuration	Symbol	Gear	Speed Brakes	Power	Trim Speed	Speed Range	Altitude (feet)
Cruise	CR	Up	In	M=.85 in L.F.	M=.85	M=.7-V_{max} practical (50 mph increments)	12,000 and 30,000
Power On Clean	P	Up	In	NRT	V_{max} NRT	M=.7 to V_{max} practical (50 mph increments)	12,000 and 30,000
				as necessary	$V_{R/C}$ to be specified later	M=.7 to V_{max} practical (50 mph increments)	
Dive	D	Up	Out	25% NRT	All speeds in the specified speed range	All permissible speeds speeds above V_{S_G}	From 30,000
Glide	G	Up	In	Idle	1.4 V_{S_G}	All permissible speeds above V_{S_G} (50 mph increments) All permissible speeds above V_{S_G} (25 mph increments)	12,000 and 30,000
Power Approach	PA	Down	In	L.F. at 1.2 V_{S_L}	L.F. at 1.2 V_{S_L}	All permissible speeds above V_{S_G} (25 mph increments)	12,000
Land	L	Down	In	Idle	1.4 V_{S_L}	All permissible speeds above V_{S_G} (25 mph increments)	12,000

NOTE: Maintain specified altitudes to ± 2000'.
A plot of stalling speed vs gross weight is presented on page 18.

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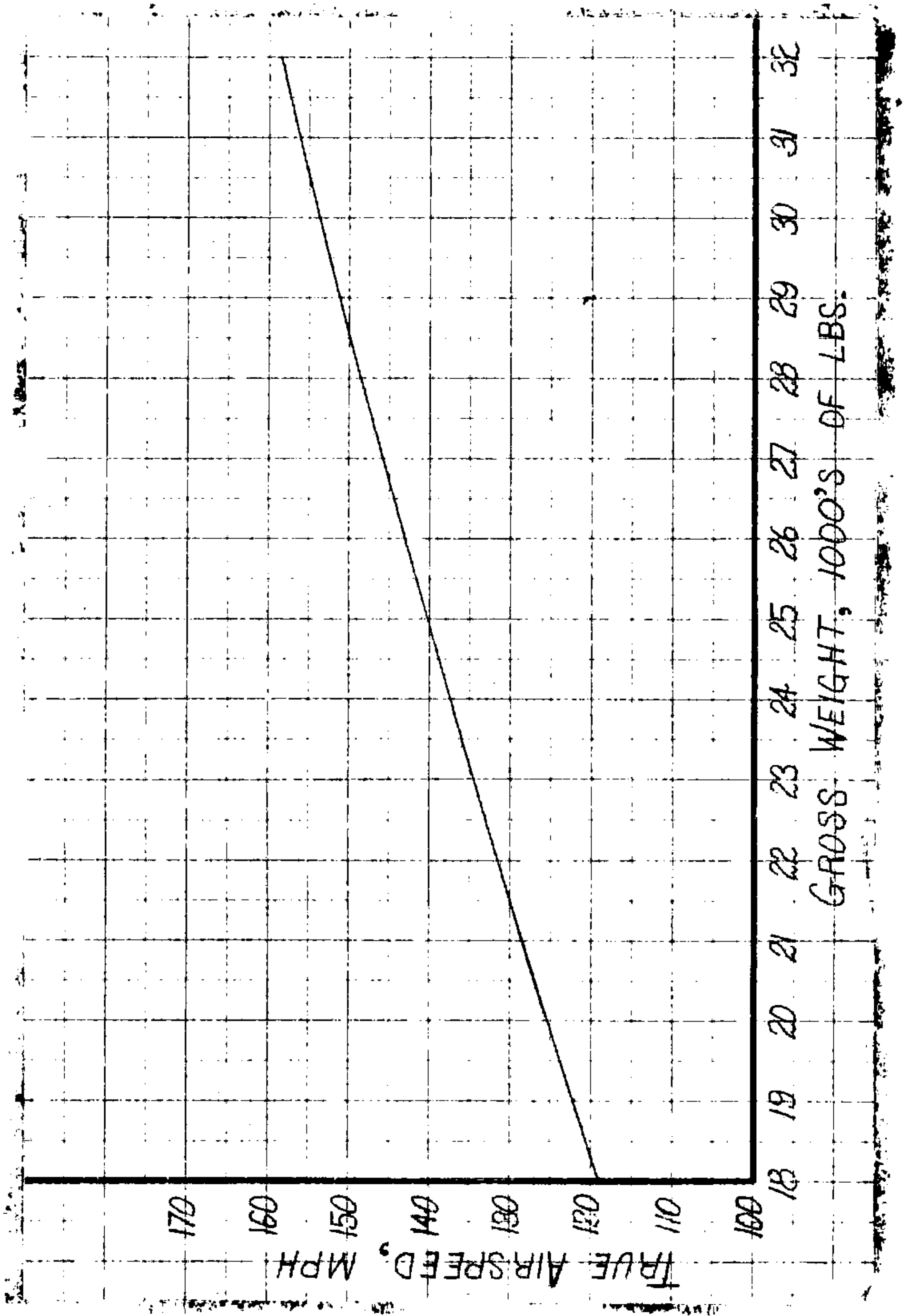

STALLING SPEED VS. GROSS WEIGHT, SEA LEVEL

BASED ON $C_L = 0.75$

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Test: Static Longitudinal Stability CVAC Index No. 1.1A

Purpose: To determine the effect of C.G. shift with airplane attitude on static longitudinal stability

Requirements: None

Remarks: Center of gravity shift due to attitude is maximum when No. 2 fuel tank is approximately 1/2 full (No. 3 full) Fuel quantity of 3114 lbs.

Procedure:

1. Conduct at 12,000 ft.
2. Trim airplane in steady level flight with 50% NRT.
3. Vary airplane attitude to 12° nose up and 12° nose down in stabilized 3° increments. Use increasing or decreasing power to maintain trim speed.
4. Determine δ_e vs. C.G. position.

Test: Dynamic Longitudinal Stability CVAC Index No. 1.1C
Applicable Spec. R1815-B Par. 4.1

Purpose: To demonstrate that elevator induced longitudinal oscillations are damped out.

Requirements: Short period dynamic oscillations produced by sudden elevator deflection shall be damped out. Damping oscillations must be stabilizing but number of cycles required to dampen to be approved by WADC (Spec. Deviation).

Procedure:

1. Conduct at altitudes specified in Table I, page 17.
2. Trim airplane in steady level flight in configurations and trim speeds specified in Table I, page 17.
3. Deflect elevator abruptly and release. Record data until short period oscillations cease. Both pull-ups and push-overs should be accomplished. The Programmer will be used to supply specified elevon deflections when practical.
4. Repeat at high and low speeds of each configuration.
5. The above tests are to be conducted with manual and direct manual mode control system configurations.

Test: Longitudinal Control Force — CVAC Index No. 1.1D
Applicable Spec. R 1815-B Para. 4.4.1, 4.4.2, 4.4.4

Purpose: To determine elevator control force gradient (stick force per "g").

Requirements: Show that elevator control force gradient is within the specified limits: 12.9 lbs/g max.; 4.35 lbs/g min.

In steady turning flight and in pull-outs, increases in pull force shall be required to produce increases in normal acceleration, throughout the range of permissible accelerations.

Elevator control force gradient for maximum normal acceleration in sudden pull-ups from trimmed straight flight shall never be less than the gradient in steady turning flight under the same conditions of gross weight, configuration, altitude, C.G. and airspeed.

Procedure:

1. Conduct at altitudes specified in Table 1, page 17.
2. Trim airplane in configurations P and D listed in Table 1, page 17.
3. Climb to 2000 ft. above test altitude.
4. Make steady diving turns in which the steepness of the turn is used to control the acceleration and the airspeed is held constant at trim value by losing altitude.
5. Develop maximum positive load factor in turns.
6. Make sudden pull-ups from straight trimmed flight during which maximum positive load factor is developed for configuration P only. Conduct at most rearward C.G. and at trim speed of M=0.7, speed for NRT, and at maximum practical.

Test: Longitudinal Control Force — CVAC Index No. 1.1D
Applicable Spec. R 1815-B Para. 4.5.4

Purpose: To determine elevator control force during high speed dive.

Requirements: With airplane trimmed in steady level flight with NRT, configuration P, it shall be possible to hold the airplane in dives up to the maximum practical speed with an elevator push force not to exceed 56 lbs. If a pull force is required to maintain a steady dive, this pull force shall not exceed 35 lbs. and the additional force required to execute a 3g pull-out shall not exceed 75 lbs.

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Test: Longitudinal Control Force (Continued) CVAC Index No. 1.1D
Applicable Spec. R 1815-B Para. 4.5.4

Procedure:

1. Trim airolane in steady level flight with NRT at service ceiling (approximately 54,000 ft.).
2. Dive airplane and stabilize speed at approximately 50 mph increments during dive until maximum practical speed is reached. The maximum practical speed is expected to be less than maximum permissible speed designed for the J-67 engine installation.
3. Execute a 3g pull-out with recovery at not less than 15,000 ft.

Test: Longitudinal Control Effectiveness CVAC Index No. 1.1E
Applicable Spec. R 1815-B Para. 4.31.

Purpose: To prove that sufficient elevator control exists to maintain steady flight for the given conditions.

Requirements: With the airplane trimmed at the speeds specified in Table I, page 17, it shall be possible in each configuration to maintain steady flight over the specified speed range.

Procedure: This requirement will be demonstrated in tests on static longitudinal stability.

Test: Longitudinal Control Effectiveness CVAC Index No. 1.1E
Applicable Sped R-1815-B Para. 4.3.2.

Purpose: To prove that sufficient elevator control exists to develop maximum positive or negative load factor at trim speed.

Requirements: Develop maximum positive or negative limit load factor at trim speed by means of elevator control only for each configuration listed in Table I, page 17.

Above requirement is subject to following limitations:

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Test: Longitudinal Control Effectiveness (Cont'd) CVAC Index No. 1.1E
Applicable Spec. R-1815-B Para. 4.3.2

Requirements: Angle of attack of 20°

Elevator deflection of $\begin{matrix}-25^\circ \\ +10^\circ\end{matrix}$

Procedure:

1. Data to comply with above positive load factor requirements will be obtained from tests for elevator control force gradient.
2. Develop maximum negative limit load factor at the specified trim speeds by making push-overs from straight, steady trimmed flight.

Note: Tests on this airplane will be of a preliminary nature only as demonstration of compliance with this specification will be obtained during flight loads testing on F-102A Serial No. 53-1780.

Test: Longitudinal Control Effectiveness CVAC Index No. 1.1E
Applicable Spec. R 1815-B Para. 4.3.3

Purpose: To demonstrate that sufficient elevator control exists to hold airplane in close proximity to the ground at the given speeds.

Requirements:

1. With C.G. in most forward position and airplane trimmed at $1.4V_{S_L}$ or at maximum allowable ground angle, whichever attitude corresponds to the higher speed.
2. Longitudinal stick force shall not exceed 35 lbs. during (1).

Note: Maximum allowable ground angle = 20°.

Procedure: As outlined in Requirements above.

Test: Longitudinal Control Effectiveness — CVAC Index No. 1.1E
Applicable Spec. R 1815-B Para. 4.3.5
Longitudinal Control Forces — CVAC Index.No. 1.1F
Applicable Spec. R1815-B Para. 4.4.7 & 4.4.8.

Purpose: To prove that sufficient elevator control exists to attain take-off attitude at 90% V_{S_L} and to show that elevator stick force during take-off does not exceed the given limits.

Requirements: Airplane shall be able to assume take-off attitude at 90% V_{S_L} for all normal service loadings.

With airplane trimmed for 1.25 V_{S_L}, the nose wheel lift off at 90% V_{S_L} shall not require a stick pull force greater than 35 lbs.

With airplane trimmed for 1.25 V_{S_L} with C.G. in most aft position, the transition from take-off speed to 1.25 V_{S_L} shall not require a stick push force greater than 10 lbs.

Procedure: See taxi tests.

1. Set elevator trim control for trimmed flight at 1.25 V_{S_L} and record data during take-off with maximum G.W. for most aft C.G. position.

Note: Trim position for 1.25 V_{S_L} can be set with pilot's test cockpit instrumentation.

Test: Longitudinal Trim Changes — CVAC Index No. 1.1F
Applicable Spec. R 1815-B Para. 4.61, 4.6.2 & 4.6.4.

Purpose: To determine longitudinal trim change due to effects of power and configuration.

Requirements: With the airplane trimmed for zero elevator control force at any given speed and using any combination of engine thrust, gear position or speed brake setting, it shall be possible to maintain the given speed without exerting push or pull forces on the stick greater than 35 lbs. when the power, gear or speed brake settings are varied in any manner.

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Test: Longitudinal Trim Changes CVAC Index No. 1.1F
Applicable Spec. R 1815-B Para. 4.61, 4.6.2. & 4.6.4.

Requirements: With airplane trimmed in steady level flight with maximum power, it shall be possible to maintain the trim speed for one second following full extension of speed brakes with an elevator push force not to exceed 20 lbs. or a pull force not to exceed 16 lbs.

Procedure:
1. Conduct at 12,000 ft.
2. Trim airplane in steady flight at trim speeds given below and perform operation listed under "variable."

	Trim Speed	Gear	Speed Brakes	Power	Variable
1.	1.4 V_{SG}	Up	In	Idle Power	Gear Down
2.	1.4 V_{SG}	Up	In	50% NRT	Gear Down
3.	1.4 V_{SL}	Down	In	50% NRT	Idle Power
4.	1.2 V_{SL}	Down	In	Max. Power	Gear Up
5.	V_{max} NRT	Up	In	Level Flt.	Max. Power
6.	1.2 V_{SL}	Down	In	Level Flt.	Max. Power
7.	V_{max} NRT	Up	In	Level Flt.	Idle Power
8.	1.4 V_{SG}	Up	Out	Level Flt.	Gear Down
9.	V_{max} Power	Up	In	Level Flt.	Speed Brakes Out

Note: V_S based on stalling speed at $C_L = 0.75$. It is desirable that the trim change in item 3 be such that a pull force is required to maintain the given trim speed.

Test:
Test: Longitudinal Trim Change CVAC Index No. 1.1F
Applicable Spec. R 1815-B Para. 4.6.3.

Purpose: Determine longitudinal trim change in sideslips.

Requirements: The stick force in pounds required to maintain trim speed in sideslips obtainable with 50 lbs. rudder pedal force or up to maximum attainable sideslip, shall not exceed numerically the value of the stick force gradient in accelerated maneuvers or 10 lbs., whichever is lower. The preferred variation of stick force in sideslip is for increased pull force on stick with increased sideslip.

Procedure: No special test needed. Measure and record stick force during tests for static directional stability.

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Test: High Speed Requirements CVAC Index No. 1.7
Applicable Spec. R 1815-B Para. 4.5.1

Purpose: To demonstrate that no dangerous effects occur when the airplane is flown through the transonic speed range.

Requirements: The following adverse or dangerous effects due to compressibility shall not occur:

1. Severe buffeting or shaking.
2. Excessive longitudinal trim changes or uncontrollable recovery moments.
3. Erratic or violent behavior of airplane or controls.

Procedure: Demonstration of compliance for high speed requirements will be accomplished concurrently with flutter tests.

Test: Static Directional Stability CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 5.2.1

Purpose: To prove that the airplane has positive rudder-fixed static directional stability.

Requirements: The rudder-fixed static stability shall be such that rudder deflection from its trim position at zero bank produces sideslip in the correct direction when performing a steady sideslip; that is, positive (left) rudder deflection from trim shall produce positive (right) sideslip and vice versa. This requirement applies to all configurations and speed ranges listed in Table I, page 17, and in configuration PA with maximum power at all permissible speeds above $V_{S_{PA}}$. For angles of sideslip between plus and minus 15 degrees, the angle of steady sideslip shall be approximately proportional to the rudder deflection from trim. For angles of sideslip greater than 15 degrees, increases in rudder deflection shall produce increases in the angle of steady sideslip up to full rudder deflection.

Procedure:

1. Conduct at altitudes specified in Table I, page 17.
2. Trim airplane for wingslevel flight in all of the configuratioms given in Table I, page 17, at the given trim speeds.

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Test: Dynamic Directional Stability CVAC Index No. 1.2B
Applicable Spec. R 1815-B Para. 5.1.1

Purpose: To determine the damping of the yaw induced lateral-directional oscillation.

Requirements: Damping shall be positive and shall be such that the time to damp to one-half amplitude and the period shall fall within the satisfactory area of the chart given in Figure 1, page 27. There shall be no tendency for undamped small amplitude oscillations to persist.

Deviation: Artificial damping devices may be used to meet the above requirements.

Procedure:

1. Conduct tests at altitudes specified in Table I, page 17.
2. Airplane to be tested in all configurations and trim speeds given in Table 1, page 17.
3. Develop wings level yaw with rudder, applying aileron as necessary to keep wings level.
4. Release rudder abruptly, return aileron to neutral and hold fixed.
5. Record data until lateral-directional oscillations cease.
6. Use manual mode of control.

Test: Spiral Stability CVAC Index No. 1.2C
Applicable Spec. R 1815-B Para. 5.1.2

Purpose: To demonstrate the airplane has satisfactory spiral stability characteristics.

Requirements: Spiral stability is not required but the allowable rate of divergence of the spiral mode with controls free shall not be so great that the spiral motion doubles amplitude in less than 4 seconds. This requirement applys to the conditions of Table I, page 17.

Procedure:

1. Conduct tests in configurations and at the altitudes and trim speeds specified in Table I, page 17.
2. Trim airplane carefully in straight steady level flight for the conditions specified above. Cause a small lateral displacement by slowly applying a small aileron displacement (approximately 2-3 degrees) and record data during subsequent airplane behavior. Perform maneuver with all controls free and also with all controls returned to neutral and held fixed following initial displacement.

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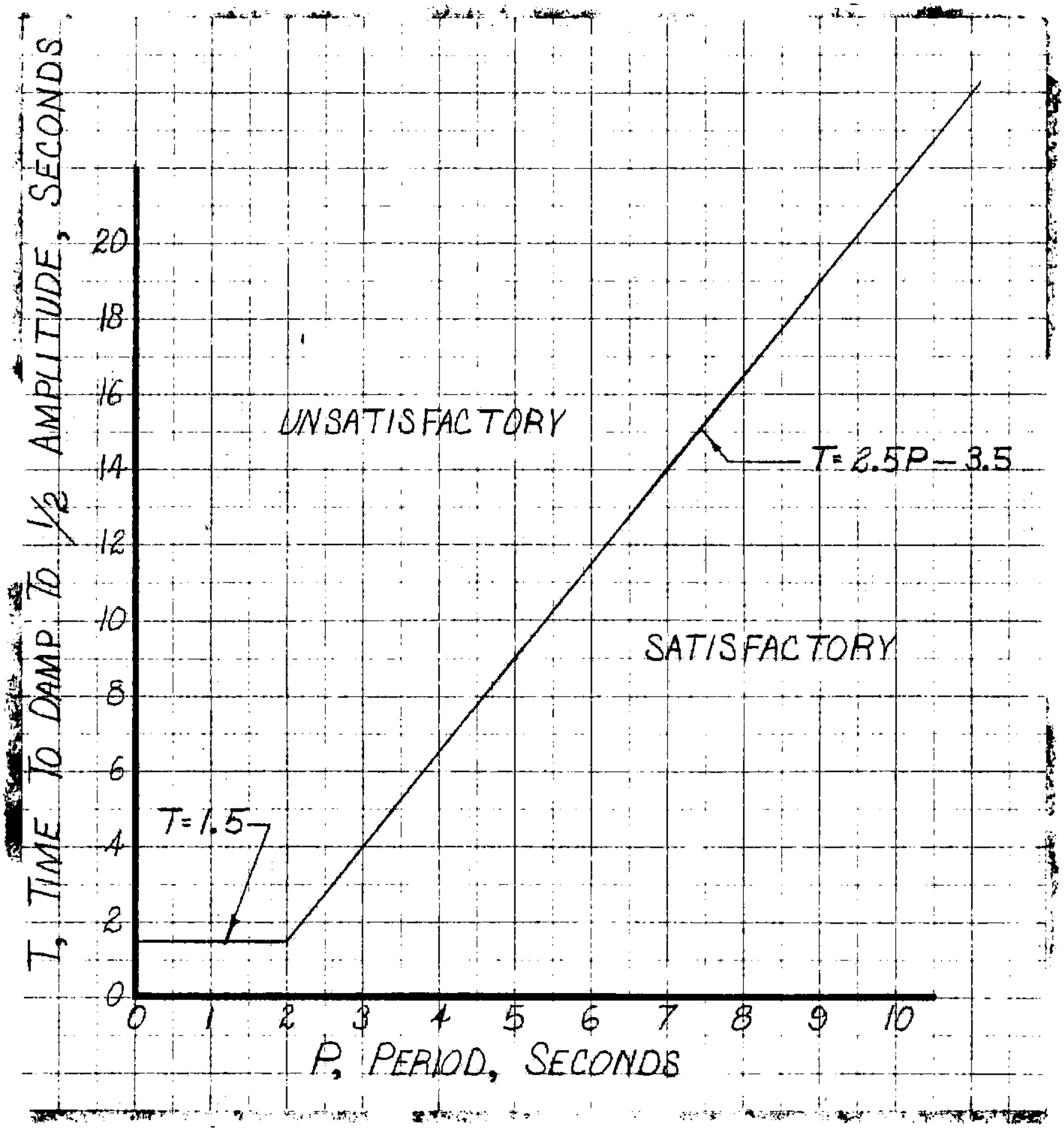

FIGURE 1

REQUIRED DAMPING OF THE LATERAL-DIRECTIONAL OSCILLATION

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Test: Static Directional Stability (Continued) CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 5.2.1

Procedure:

3. Perform a straight steady sideslip, banking the airplane as necessary to maintain straight flight, in increments of three degrees of rudder deflection until limit side load factor is reached.
4. Stabilize at each rudder setting and record data.

Remarks: Direct manual mode of control.

Test: Static Directional Stability CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 5.2.2

Purpose: To determine adverse yaw produced during roll-outs from banked steady turns.

Requirements: Rudder-fixed static directional stability shall be sufficient to restrict the angle of sideslip due to sudden application of aileron deflection to not more than one degree of sideslip per 5% of full aileron deflection, when rolling out of a trimmed steady 45° banked turn at $1.4\ V_{S_{CR}}$ in configuration G with power for straight and level flight at $1.4\ V_{S_{CR}}$ and in configuration PA at $1.4\ V_{S_{CR}}$.

Procedure:

1. Conduct in configuration G and in configuration PA at 10,000 ft.
2. Trim airplane in configuration to be tested at $1.4\ V_{S_{CR}}$ in a steady 45° banked turn.
3. Apply aileron deflections abruptly to roll airplane into opposite bank, holding rudder fixed in trim position, continue roll to 45° bank on opposite side.
4. Resume a steady turn in trimmed flight, then reverse aileron displacement back to original turn again.

NOTE: The limiting factor in conducting these tests is the side load.

Test: Rudder Control Effectiveness CVAC Index No. 1.2B
Applicable Spec. R 1815-B Para. 5.3.1

Purpose: To prove that rudder effectiveness is sufficient to maintain straight level flight in all configurations.

Requirements: Maintain straight level flight for all configurations and trim speeds given in Table I, page 17, and also in configuration WO trimmed at 1.2 V_{S_L}. Maximum rudder pedal forces not to exceed 180 lbs. except for configuration WO where they shall not exceed 100 lbs.

Procedure: No special test required. This will be demonstrated in other tests. Refer to static longitudinal stability.

Note: Configuration WO
Maximum Available Thrust-Gear Down-Trim Speed 1.2 V_{S_L}.

Test: Rudder Control Effectiveness CVAC Index No. 1.2H
Applicable Spec. R 1815-B Para. 5.3.6

Purpose: To show that rudder control is sufficient to overcome adverse yaw due to aileron action.

Requirements: Rudder control shall be sufficient to overcome adverse yaw when airplane is rolled abruptly out of steady 45° banked turns in configuration G with power for straight and level flight at 1.4 $V_{S_{CR}}$ and in configuration at PA at 1.4 $V_{S_{CR}}$.

Procedure:

1. Conduct at 10,000 ft.
2. Trim airplane in configuration to be tested in a steady 45° banked turn at 1.4 $V_{S_{CR}}$.
3. Apply aileron deflection abruptly to roll airplane into opposite bank using necessary rudder to prevent skidding. Continue roll to 45° bank on opposite side or until maximum allowable side load factor is reached.
4. Resume a steady turn in trimmed flight, then reverse aileron displacement back to original turn.

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Test: Directional Trim Control Effectiveness — CVAC Index No. 1.2F
Applicable Spec. R 1815-B Para. 5.6.3.

Purpose: To demonstrate that directional trim is sufficient to reduce rudder pedal force to zero for the given conditions.

Requirements: The directional trimming device shall be capable of reducing the rudder pedal forces to zero and/or trim for straight zero yaw wings level flight in the following configurations and speed ranges:

Configuration	Speed Range
G	$1.2V_{S_G}$ to V_{max} practical
P	$1.2V_{S_C}$ to V_{max} practical
PA	$1.2V_{S_L}$ to 310 MPH

Procedure: No special test is needed. The above requirements will be demonstrated during tests for static longitudinal stability.

Test: Static Lateral Stability — CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 6.2.1

Purpose: To demonstrate that the airplane has positive dihedral effects for the conditions specified.

Requirements: Right stick force and deflection is required to depress right wing when airplane is in right steady sideslip with the airplane trimmed for steady unyawed flight and vice versa.

This is required for all configurations and trim speeds given in Table I, page 17, and for configuration WO.

Procedure: No special tests are required. Data obtained from tests for static directional stability will demonstrate compliance with the requirements listed above.

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Test: Dynamic Lateral Stability — CVAC Index No. 1.2B
Applicable Spec. R 1815-B Para. 6.1.3.

Purpose: To determine the damping of the roll induced lateral-directional oscillations.

Requirements: Same as for dynamic directional stability.

Procedure:

1. Conduct at altitudes specified in Table I, page 17.
2. Airplane to be tested in all configurations and at all trim speeds given in Table I, Page 17.
3. Abruptly deflect ailerons by sharp application of approximately one-eighth to one-quarter full lateral displacement of stick and immediately release control.
4. Record data until lateral-directional oscillations cease.
5. Use manual mode of control.

Test: Dihedral Effect — CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 6.2.3.

Purpose: To determine the aileron deflection required to overcome the dihedral effect in steady sideslip.

Requirements: Not more than 90% of full aileron deflection shall be required to counteract the rolling moment produced by full available rudder pedal deflection in steady sideslips.

Procedure: No special testing is needed. Data will be obtained during tests for static directional stability.

Test: Dihedral Effect — CVAC Index No. 1.2A
Applicable Spec. R 1815-B Para. 6.2.5.

Purpose: To show that side force due to sideslip is in the right direction.

Requirements: Right skidding turns shall accompany right rudder deflection and vice versa.

Procedure: No special tests are needed. During steady sideslips, show that an increase in right bank accompanies an increase in right sideslip.

Test: Aileron Control Effectiveness CVAC Index No. 1.2D
Applicable Spec. R 1815-B Para. 6.3.4.

Purpose: To demonstrate that the ailerons are sufficiently powerful to produce peak roll rates in a rudder-fixed roll.

Requirements: The ailerons shall be sufficiently powerful to produce the following peak roll rates with the rudder held fixed in the position required for steady flight prior to the roll.

Config.	Gear	Power	Speed Range	Min. Acceptable Peak Roll Rate
P	Up	L.F.	$1.1V_{S_G}$ to V_{max} NRT	$\frac{Pb}{2V} = .090$
L	Down	Idle	$1.1V_{S_L}$ to $1.5V_{S_L}$	$\frac{Pb}{2V} = .07$
L	Down	Idle	$1.1V_{S_L}$	$Pb = 10$ ft/sec

Procedure:

1. Conduct at 10,000' PA.
2. Trim airplane at the required speed for the configuration to be tested in 45° banked turn. Speed increments, configuration P - 50 MPH, increments configuration L - 10 MPH.
3. After turn is stabilized, hold rudder fixed and apply full opposite aileron abruptly and hold until airplane is rolled into an opposite 45° bank.
4. Repeat with roll in opposite direction.

Deviation: Peak roll rates at V_{max} NRT to be approved by WADC.

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Test: Aileron Control Effectiveness CVAC Index No. 1.2D
Applicable Spec. R 1815-B Para. 6.3.1 and 6.3.3.

Purpose: To show that rolling velocity produced by abrupt aileron action varies smoothly with aileron deflection and is in the correct direction. (Rudder fixed)

Requirements: The variation of angle of bank with time immediately following an abrupt control deflection shall always be in the correct direction.

The maximum rolling velocity obtained by abrupt aileron deflection, with rudder held fixed in the trim position, shall vary smoothly with the aileron deflection from the trim position and shall be approximately proportional to the aileron. Deflection from the trim position.

Procedure: No special test needed, data obtained from tests for aileron control effectiveness will demonstrate above requirements.

Test: Aileron Control Forces CVAC Index. No. 1.2E
Applicable Spec. R 1815-B Para. 6.4.1.

Purpose: To determine aileron control forces during accelerated maneuvers.

Requirements: Aileron control force shall vary smoothly with stick deflection during rolling maneuvers and steady sideslips. Aileron control shall be free from shake and snatching tendency at any speed for any permissible acceleration.

Procedure: No special test needed, aileron control force data obtained from tests for static directional stability and aileron control effectiveness will demonstrate above requirements.

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Test:	Aileron Control Effectiveness Applicable Spec. R 1815-B Para. 6.3.2	CVAC Index No. 1.2D
Purpose:	To determine the magnitude and time lag of the peak rolling acceleration.	
Requirements:	The peak value of rolling acceleration shall not occur later than $3.75\ \bar{c}/V_t$ seconds or 0.1 seconds whichever is greater after full aileron deflection is reached in a rudder-locked roll in configuration G with level flight power at $1.2\ V_{S_G}$.	
Procedure:	No special test needed; data obtained during tests for aileron control effectiveness will demonstrate above requirements.	

Test:	Lateral Trim Control Effectiveness Applicable Spec. R 1815 - B Para. 6.6.2 and 6.6.3.	CVAC Index No. 1.2F
Purpose:	To determine lateral trimming effectiveness.	
Requirements:	The aileron trim control shall maintain a given setting indefinitely unless changed intentionally. The lateral trim control shall be capable of reducing the aileron control force to zero for the same conditions given for directional trim control effectiveness, page 30.	
Procedure:	No special test needed. Trimming effectiveness for all speeds with wings level will be determined during tests for static longitudinal stability.	

Test: Stalling Characteristics CVAC Index No. 1.4
Applicable Spec. R 1815-B Para. 7.1.1, 7.1.2, 7.1.3 and 7.3.1.

Purpose: To demonstrate that the airplane exhibits no abrupt changes in flight characteristics in the region of maximum lift coefficient.

Requirements:

1. For purposes of stalling characteristics, demonstration only, the stall shall be considered to be identified by full up elevator deflection.
2. If it can be shown that the aircraft exhibits no abrupt changes in flight characteristics in the region of maximum lift coefficient, the complete stall shall be coincident with the minimum speed in straight flight or the maximum acceleration in turning flight.
3. The airplane shall exhibit no excessive downward pitch, excessive roll, excessive loss of altitude, or excessive delay in recovery from the stalled flight condition as defined above.
4. The stalling requirements shall govern for both straight and turning flight for all service loading conditions in the following configurations:

Cruise	CR
Glide	G
Land	L
Power Approach	PA

Procedure:

1. Trim airplane in straight, steady, level flight in the following configurations:

Configuration	Trim Speed	Power Setting	Altitude
CR	M = .85	Power for level flight at M=.85	12,000 & 30,000' PA
G	1.4 V_{S_G}	Idle	20,000' PA
L	1.4 V_{S_L}	Idle	Minimum Safe Altitude
PA	1.4 V_{S_L}	Power for level flight at 1.4 V_{S_L}	Minimum Safe Altitude
P	1.4 $V_{S_{CR}}$	Power for level flight at 1.4 V_{S_L}	Speed Brakes Extended

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Test: Stalling Characteristics (Continued) CVAC Index No. 1.4
Applicable Spec. R 1815-B Para. 7.1.1, 7.1.2, 7.1.3 and 7.3.1.

Procedure:

2. Trim airplane in each of the conditions specified above and gradually reduce airspeed by pulling back on stick until maximum elevator deflection is reached. The rate of decreasing speed should not exceed 1 MPH per second.
3. Trim airplane as for (1) above and perform coordinated turns in which the maximum positive load factor ($n = 5.33$) is reached.
4. Perform in direct manual flight mode.

Note: V_S used for trim will be based on a stall speed at $C_L = 0.75$ and not on the stall speeds as defined in the requirements above.

Test: Power Control System Operation CVAC Index No. 4.6
Applicable Spec. R 1815-B, Para. 8.4.1

Purpose: To demonstrate satisfactory operation of the control system when either the primary or secondary hydraulic system fails.

Requirements: In trimmed level flight in configuration P at 10,000 ft. at all speeds between 1.4 V_{SG} and V_{max}, abrupt failure of either the primary or secondary hydraulic systems shall not result in out of trim conditions exceeding the following limitations:

Elevator: ± 1/2g accelerations - controls free
Rudder: maintain zero sideslip with no more than 100 pounds pedal force.
Ailerons: rate of roll not to exceed 5°/sec with controls free.

Procedure: Compliance with the above requirements will be demonstrated during ground test and subsequent flight test.

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Test: Flight Control System Operation CVAC Index No. 4.6

Purpose: To evaluate the components of the control system.

Requirements: The components of the control system as listed below shall operate in an acceptable manner under all flight conditions.

Procedure: The data required will be obtained concurrently with other tests. No special tests are required for this evaluation which will be obtained from qualitative and quantitative data. The items to be evaluated are listed below.

Rudder Pedals (Directional)

1. Travel
2. Comfort
3. Relation to surface travel

Rudder Pedal Adjustment (Directional)

1. Comfort

Rudder Pedal Force Sensing Elements (Directional)

1. Force vs. Surface Deflection
2. Summing and Differentiation
3. Environment Effects
4. High Speed Limitations of Rudder Deflection

Stretch Spring (Directional)

1. Stretch vs. Pedal Force vs. Rudder Position
2. Environment

Trim Motors (ALL)

1. Rate
2. Sensitivity

Manual Mode Trim

1. Rate
2. Sensitivity
3. Stability

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Test: Flight Control System Operation (Cont'd) CVAC Index No. 4.6-1

"Q" Pot (Directional Longitudinal)

1. Pressure vs. airspeed and/or regulated press
2. Effect of Attitude, Yaw, Load Factor
3. Control Forces vs. pressures and position from trim.

"Q" Pot Pressure Regulation (Directional Longitudinal)

1. Pressure sensed vs. airspeed
2. Regulated Press vs. Sensed Pressure
3. Effect of Attitude, Yaw, Load Factor

Valve Follow up Lock out (ALL)

1. Rate of operation
2. Breakout forces

Servo Valve Actuator (ALL)

1. Rate of operation
2. Lags
3. Stability
4. Lockout operation

Servo Valve Operation (ALL)

1. Valve deflection vs. surface travel rate
2. Stability

Surfaces (ALL)

1. Lag behind servo valve operation

Control Stick (Longitudinal, Lateral)

1. Comfort and room
2. Position relative to surface deflections
3. Play

Aileron Feel Spring (Lateral)

1. Force vs. deflection vs. airspeed

Test: Flight Control System Operation (Cont'd) CVAC Index No. 4.6-1

Friction (ALL)

1. Effect of control force sensitivity
2. Effects of Environment

MEASUREMENT GROUPS

1. General
2. Performance
3. Engine cooling
4. Longitudinal dynamic response
5. Lateral dynamic response
6. Directional dynamic response
7. Control system operation
8. Flutter analysis
9. Structural cooling
10. Angle of attack and yaw calibration
11. Engine operation
12. Low pressure pneumatic system operation

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YF-102 No. 53-7994

Flight Test Program

MEASUREMENT GROUPS

Group No. 1
Type Test: General

	Measurement	Recording	Range
1	Airspeed	Photo	50 - 700 mph
2	Mach Number	Pilot's	.5 - 1.5
3	Pressure Altitude	Photo	0 - 60,000 ft.
4	Attitude 3 Axes	Photo	—
5	Time of Day	Photo	Sweep Second
6	O.A.T.	Photo	-58° +122°F
7	Engine RPM N_2	Photo	0 - 7000 rpm
8	Turbine Discharge Pressure PT_7	Photo	0 - 200 in.Hg
9	Nozzle Position Indication	Photo	On - Off
10	Angle of Attack	Photo	-5 +25°
11	Fuel Remaining	Photo	0 - 1000 gals.
12	Normal Acceleration Pilot's Panel	Photo-Osc. Pilot's	+12g -5g

Group No. 2
Type Test: Performance

	Measurement	Recording	Range
1	Group (1)	----	----
2	Speed Brake Positions	Photo	0 - 60°
3	Fuel Flow to Engine	Photo	0 - 26 gpm
4	Total Fuel Flow	Photo	0 - 52 gpm
5	Engine RPM (N_1)	Photo	0 - 7000 rpm
6	Engine Inlet Total pressure	Photo	0 - 200 in.Hg Abs
7	Engine Inlet Temperature	Photo	+300°F -96°F
8	Turbine Discharge Temperature	Photo	0° - 1830°F
9	Elevon Surface Positions	Photo	+35° -20°
10	Rudder Surface Positions	Photo	±25°
11	Normal Acceleration C.G.	Osc.	±10g
12	Longitudinal Acceleration	Osc.	±2g

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Group No. 3
Type Test: Engine Cooling

	Measurement	Recording	Range
1	Group (1)	----	----
2	Fuel Flow to Engine	Photo	0 - 26 gpm
3	Total Fuel Flow	Photo	0 - 52 gpm
4	Engine RPM (N_1)	Photo	0 - 7000 rpm
5	Air Inlet to Engine Total Pressure	Photo	0 - 200 in.Hg Abs
6	Air Inlet to Engine Total Temperature	Photo	+300°F -96°F
7	Inter Compressor Total Pressure	Photo	0 - 200 in.Hg Abs
8	Inter Compressor Total Temperature	Photo	0 - 1830°F
9	Compressor Discharge Pressure	Photo	0 - 400 in.Hg Abs
10	Compressor Discharge Temperature	Photo	0 - 1830°F
11	Turbine Discharge Temperature	Photo	0 - 1830°F
12	Frame Temperatures	Telemeter	0 - 480°F
13	Fuel Control Air Temperature	Telemeter	0 - 212°F
14	Engine Oil In Temperature	Telemeter	0 - 480°F
15	Tail Pipe Surface Temperature	Telemeter	0 - 1830°F
16	GLA Ignition Temperature		
17	Aft Firewall Fuselage Pressure	Telemeter	0 - 15 psia

Group No. 4
Type Test: Dynamic Response Longitudinal

1	Group (1)		
2	Longitudinal Trim Position	Photo	+12.5° -2°
3	Elevon Surface Position	Photo-Osc.	+34° -19°
4	Speed Brake Positions	Photo	0 - 60°
5	Static Pressure Elevon "Q" Pot	Photo	0 - 60,000 ft.
6	Differential Pressure Elevon "Q" Pot	Photo	50 - 700 mph
7	Programmer Longitudinal Signal Output	Osc.	----
8	Elevon Valve Spool Positions	Osc.	±.125"
9	Longitudinal Stick Position	Osc.	+20° -9°
10	Pitch Rate	Osc.	±60°/sec.
11	Pitch Acceleration	Osc.	± 3 RAD/sec.^{2}
12	Pitch Attitude	Osc.	±90°
13	Angle of Attack	Osc.	+25° -5°
14	Pitch Damper Signal	Osc.	----

Group No. 5
Type Test: Dynamic Response (Lateral)

	Measurement	Recording	Range
1	Group (1)		
2	Lateral Trim Position	Photo	±3°
3	Rudder Surface Position	Osc.	±25°
4	Pitch Attitude	Osc.	±90°
5	Roll Attitude	Osc.	±175°
6	Angle of Attack	Osc.	+25° -5°
7	Angle of Yaw	Osc.	±15°
8	Normal Acceleration at C.G.	Osc.	±10g
9	Yaw Damper Signal	Osc.	----
10	Yaw Acceleration	Osc.	1.5 RAD/sec.^{2}
11	Yaw Rate	Osc.	±25°/sec.
12	Elevon Surface Positions	Photo-Osc.	+34° -19°
13	Elevon Valve Spool Positions	Osc.	±.125"
14	Lateral Stick Position	Osc.	±20°
15	Lateral Stick Force	Osc.	±100 lbs.
16	Roll Rate	Osc.	±300°/sec.
17	Roll Acceleration	Osc.	±3 RAD/sec.^{2}
18	Elevon Actuators Positions	Photo	+35° -20°
19	Programmer Lateral Signal Output	Osc.	---

Group No. 6
Type Test: Dynamic Response (Directional)

1	Group (1)		
2	Rudder "Q" Pot Differential Pressure	Photo	0 - 5000 psi
3	Speed Brake Positions	Photo	0 - 60°
4	Rudder Trim Position	Photo	±5°
5	Rudder Surface Position	Osc.	±25°
6	Angle of Attack Nose Boom	Osc.	+25° -5°
7	Programmer Output (Yaw)	Osc.	
8	Lateral Acceleration	Osc.	±5g
9	Yaw Damper Signal		
10	Rudder Pedal Position	Osc.	±3.75 in.
11	Direction Control Force	Osc.	0 - 300 lbs.
12	Rudder Control Valve Spool Position	Osc.	±.125 in.
13	Rudder Control Valve Servo Actuator Pos.	Osc.	±.120 in.
14	Yaw Acceleration	Osc.	1.5 RAD/sec.^{2}
15	Yaw Rate	Osc.	±25°/sec.
16	Yaw Angle	Osc.	±15°
17	Elevon Surface Positions	Osc.	+34° -19°

(Continued next page)

Group No. 6 (Continued)
Type Test: Dynamic Response (Directional)

	Measurement	Recording	Range
18	Pitch Rate	Osc.	±60°/sec.
19	Roll Rate	Osc.	±300°/sec.
20	Roll Attitude	Osc.	±175°
21	Directional Heading	Osc.	±180°

Group No. 7
Type Test: Control System (Operation)

1	Group (1)		
2	Elevon Surface Positions	Photo	+34° -19°
3	Elevon Surface Positions	Osc.	+34° -19°
4	Rudder Surface Position	Photo	±25°
5	Rudder Surface Position	Osc.	±25°
6	Longitudinal Trim Position	Photo	+12.5° -2°
7	Lateral Trim Position	Photo	±3°
8	Directional Trim Position	Photo	±5°
9	Rudder "Q" Pot Total Sensing Pressure	Photo	50 - 700 mph
10	Rudder "Q" Pot Differential Pressure	Photo	0 - 5000 psi
11	Elevon "Q" Pot Differential Sensing Pres.	Photo	50 - 700 mph
12	Engine RPM N_1	Photo	0 - 7000 rpm
13	Yaw Damper Signal	Osc.	----
14	Rudder Pedal Position	Osc.	±3.25 in.
15	Longitudinal Stick Position	Osc.	+20° -9°
16	Lateral Stick Position	Osc.	±20°
17	Longitudinal Stick Force	Osc.	±150 lbs.
18	Lateral Stick Force	Osc.	±100 lbs.
19	Rudder Pedal Force	Osc.	0 - 300 lbs.
20	Rudder Control Valve Spool Position	Osc.	±.125 in.
21	Elevon Control Valves Spools Positions	Osc.	±.125 in.
22	Rudder Control Valve Servo Actuator Pos.	Osc.	±.120 in.
23	Elevon Control Valve Servo Actuator Pos.	Osc.	±.120 in.
24	Angle of Yaw	Osc-Photo	±15°
25	Angle of Attack	Osc-Photo	+25° -5°
26	Pitch Damper Signal	Osc.	----
27	Hydraulic Oil Temperature	Telemeter	0 - 212°F
28	Rudder pedal force Signal Output	Osc.	----

Group No. 8
Type Test: Flutter Analysis

	Measurement	Recording	Range
1	Group (1)		
2	Elevon Surface Position	Photo-Osc.	+34° -19°
3	Rudder Surface Position	Photo-Osc.	±25°
4	Wing Vibrator Signals	Osc.	
5	Vertical Accelerations Wings & Elevons	Osc.	±5g
6	Lateral Accelerations Fin	Osc.	±5g
7	Wing Vibration	Pilot's	

Group No. 9
Type Test: Structure Cooling

1	Group (1)		
2	Fuel Flow to Engine	Photo	0 - 26 gpm
3	Total Fuel Flow	Photo	0 - 32 gpm
4	Engine RPM (N_1)	Photo	0 - 7000 rpm
5	Turbine Discharge Temperature	Photo	0 - 1830°F
6	Frame Temperatures	Telemeter	0 - 480°F
7	Rear Spar Carry Through Temperatures	Telemeter	0 - 480°F
8	Wing Spar #5	Telemeter	0 - 480°F
9	Drogue Chute Pan Temperature	Telemeter	0 - 480°F

Group No. 10
Type Test: Angle of Attack and Yaw Calibration

1	Group (1)		
2	Angle of Attack Nose Boom Aft	Photo	-5 +25°
3	Angle of Attack (Fuselage) Multiple	Photo-Osc.	-5 +25°
4	Angle of Yaw (Fuselage)	Photo	±15°

Group No. 11
Type Test: Engine Operation

1	Group (1)		
2	Fuel Flow Engine	Photo	0 - 26 gpm
3	Fuel Flow Total	Photo	0 - 52 gpm
4	Engine RPM (N_1)	Photo	0 - 7000 rpm
5	Engine Inlet Total Pressure	Photo	0 - 200 in.Hg Abs
6	Engine Inlet Temperature	Photo	+300° -96°F
7	Turbine Discharge Temperature	Photo	0 - 1830°F

(Continued next page)

Group No. 11 (Continued)
Type Test: Engine Operation

	Measurement	Recording	Range
8	Inter Compressor Temperature	Photo	0 - 1830°F
9	Inter Compressor Total Pressure	Photo	0 - 200 in. Hg
10	Compressor Discharge Temperature	Photo	0 - 1830°F
11	Fuel Pump Inlet Pressure	Telemeter	0 - 50 psi
12	Engine Oil In Temperature	Telemeter	0 - 480°F
13	Engine Oil Pressure	Telemeter	0 - 100 psi

Group No. 12
Type Test: Low Pressure Pneumatic System

1	Group (1)		
2	Cockpit Pressure	Pilot's	
3	Generator and Alternator Temperatures	Telemeter	0 - 480°F
4	Air Pressure to Fuel Tanks	Telemeter	0 - 10 psi
5	Fuel Tank Regulator Air Sensing Temps.	Telemeter	0 - 15 psia
6	Fuel Inlet Temperature R & L Wing	Telemeter	0 - 480°F

YF-102 No. 52-7994 FLIGHT TEST

CVAC Index No.	Test	Configuration	CONFIGURATION			
			Trim Speed	Speed Range	Pressure Altitude	
1.5	Low Speed Taxi Tests		—	0-60	Ground	
1.5	High Speed Taxi Tests		—	0 - T.O.	Ground	
1.12	Shakedown Flight Tests	Clean	—	500 MPH(TAS)	Up to 20,000'	Op
2.1	Airspeed Calibration - Low Speed Range	Clean	—	150-350 MPH	Optional	Op
		Speed Brakes Extended	—	150-350 MPH	Optional	Op
		L.G. Extended		150-310 MPH	Optional	Op
2.1	Airspeed Calibration - High Speed Range	Climb	M=.80	M=.85	Ground-Service Ceiling	
		Dive	M=1.3		Service Ceiling to Min. Pull-out	R
5.0	Angle of Attack and Angle of Yaw Pickup Calibration	All	—	130 MPH to V_{max}	Optional	Op
1.8	Flutter and Vibration Tests	Clean-Level Flt.	M=.70*	To V_{max}	20,000'	Co po to Fu
1.7	Dynamic Response Tests	Clean-Level Flt	—	—	—	—
1.12	Preliminary Flight Characteristics Tests	Clean-Level Flt	—	—	20,000'	Opt
4.16	Fuel Tank Pressurization System Tests	All	—	0 - V_{max}	All	—
4.2	Drogue Chute Deployment Tests	On ground	—	30 &150 MPH	Ground	Lan
4.1	Emerg. Ram Air Hyd. Power Unit Tests	Level Flight	V_{max}	V_{max}-Land	35,000'	Opt
2.9	Fuel Consumption Tests	—	—	—	—	—
3.0	Emerg. Fuel System Operation Tests	Clean	—	—	5,000 35,000	Opt
4.14	Speed Brake Tests	As required	—	All	— —	Opt Lan

IGHT TEST PROGRAM

ssure itude	G.W.	C.G. % MAC	Meas. Req'd. (Instr. Groups)	Est. Flt. Hrs.	Est. Calendar Period From	To	Remarks
und	T.O.	Result.	3+7+9 +11+12	0	10/53	11/53	
und	T.O.	Result.	"	0	10/53	11/53	
to 000'	Optimum	Result.	All	2	10/53	11/53	
ional	Optimum	Result.	2+10				With pacer aircraft
ional	Optimum	Result.	"	1.5	10/53	11/53	
ional	Optimum	Result.	"				
nd- ice ling	T.O.	Result.	"	3	10/53	3/54	With radar and radio sonde weather data
ice ing in. -out	Result.	Result.	"				
onal	Optional	Optional	"	3	10/53	6/54	
00'	Corresponding to 60% Fuel ±10%	Result.	2+8	16.5	10/53	3/54	*Initial speed. Speed then to be increased by M = .10 and test cont.
	—	—	4+5+6+7	30	12/53	6/54	Details of this test not available
00'	Optimum	Result.	4+5+6+7	6	11/53	12/53	
	—	—	12	1	10/53	3/55	
d	Landing	Result.	2	0	10/53	11/53	
00'	Optimum	Result.	2	0.5	10/53	11/53	
	—	—	2+11	2	10/53	3/55	Tests to be conducted concurrently with other tests
0	Optimum	Result.	2+3+11	.5	12/53	1/54	
	Optimum Landing	Result. Result.	2	1	5/54	3/55	

CVAC Index No.	Test	Misc. Items	Trim Speed	Configuration		
				Speed Range	Press. Alt.	G.W.
1.1A	Static Longitudinal Stability	Cruise	M=.85	M=.7 to V_{max} practical &	12,000 30,000	Normal Servi
1.1C	Dynamic Longitudinal Stability					
1.1D	Longitudinal Control Force					
1.1E	Longitudinal Control Effectiveness					
1.1F	Longitudinal Trim Change					
1.2A	Static Directional Stability					
1.2B	Dynamic Directional Stability					
1.2D	Rudder Control Effectiveness					
1.2A	Static Lateral Stability					
1.2B	Dynamic Lateral Stability					
1.1A	Static Longitudinal Stability	Power on Clean	L.F.atNRT M=.93			
1.1C	Dynamic Longitudinal Stability					
1.1D	Longitudinal Control Force					
1.1E	Longitudinal Control Effectiveness					
1.1F	Longitudinal Trim Change					
1.2A	Static Directional Stability					
1.2B	Dynamic Directional Stability					
1.2D	Rudder Control Effectiveness					
1.2A	Static Lateral Stability					
1.2B	Dynamic Lateral Stability					
1.1A	Static Longitudinal Stability	Level Flight 50% NRT	Corresp.	------	12,000	Corres ~ 31 fuel
1.1D	Longitudinal Control Force	Dive	M ≅.95	------	45,000 15,000	T.O.

t Program

G.W.	C.G. % MAC.	Meas. Req'd (Instr. Groups)	Est. Flt. Hrs.	Estimated Calendar Period From	To	Remarks
ormal ervice	Result.	4+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		6+7	.5	3/54	3/55	
		6+7	.5	3/54	3/55	
		6+7	.5	3/54	3/55	
		5+7	.5	3/54	3/55	
		5+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		4+7	.5	3/54	3/55	
		6+7	.5	3/54	3/55	
		6+7	.5	3/54	3/55	
		6+7	.5	3/54	3/55	
		6+7	.5	3/54	3/55	
		6+7	.5	3/54	3/55	
orres to ∼ 3100# fuel		4+7	.5	3/54	3/55	C.G. shift due to change in attitude.
T.O.		4+7	.5	3/54	3/55	Increase speed increments of 50 mph.

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YF-102 No. 52-7994 FLIGHT TEST PROGR

CVAC Index No.	Test	Miscellaneous	Trim Speed	Configuration Speed Range		Press. Alt.	G.W
1.2A	Static Directional Stability	Power-On	1.4 V_{S_C}*	------		12,000	T.O
1.2A	Static Directional Stability	Power-on-approach	1.4 V_{S_C}*	------		12,000	T.
1.2H	Rudder Control Effectiveness	Glide	1.4 V_{S_C}*	------		12,000	T.O
1.2H	Rudder Control Effectiveness	Power-on-approach	1.4 V_{S_C}*	------		12,000	T.O
				Gear	Brakes		
1.1F	Longitudinal Trim Changes	Idle Power	1.4 V_{S_G}	Up	In	12,000	T.O
		50% NRT	1.4 V_{S_G}	Up	In	12,000	T.O
		50% NRT	1.4 V_{S_L}	Down	In	12,000	T.O
		Max. Power	1.2 V_{S_L}	Down	In	12,000	T.
		Level Flight	V_{max} NRT	Up	In	12,000	T.
		Level Flight	1.2 V_{S_L}	Down	In	12,000	T.
		Level Flight	V_{max} NRT	Up	In	12,000	T.
		Level Flight	1.4V_{S_G}	Up	Out	12,000	T.
		Level Flight	V_{max}	Up	Out	12,000	T.

T PROGRAM

s. t.	G.W.	C.G. % MAC	Meas. Req'd (Instr. Groups)	Est. Flt. Hrs.	Est. Calendar Period From	To	Remarks
00	T.O.	Result	6+7	.5	3/54	3/55	Adverse yaw tests.
00	T.)		6+7	.5	3/54	3/55	*Trim in 45° banked steady turns.
00	T.0		6+7	.5	3/54	3/55	
00	T.0		6+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	
00	T.O.		4+7	.5	3/54	3/55	

CVAC Index No.	Test	Misc. Items	Trim Speed	Configuration Speed Range	P
1.1A	Static Longitudinal Stability	Power Approach	172 MPH	All permissible speeds above V_{S_L}	12
1.1C	Dynamic Longitudinal Stability	↓	↓	↓	
1.1D	Longitudinal Control Force	↓	↓	↓	
1.1E	Longitudinal Control Effectiveness	↓	↓	↓	
1.1F	Longitudinal Trim Change	↓	↓	↓	
1.2A	Static Directional Stability	↓	↓	↓	
1.2B	Dynamic Directional Stability	↓	↓	↓	
1.2D	Rudder Control Effectiveness	↓	↓	↓	
1.2A	Static Lateral Stability	↓	↓	↓	
1.2B	Dynamic Lateral Stability	↓	↓	↓	
1.1A	Static Longitudinal Stability	Landing	200 MPH	↓	
1.1C	Dynamic Longitudinal Stability	↓	↓	↓	
1.1D	Longitudinal Control Force	↓	↓	↓	
1.1E	Longitudinal Control Effectiveness	↓	↓	↓	
1.1F	Longitudinal Trim Change	↓	↓	↓	
1.2A	Static Directional Stability	↓	↓	↓	
1.2B	Dynamic Directional Stability	↓	↓	↓	
1.2D	Rudder Control Effectiveness	↓	↓	↓	
1.2A	Static Lateral Stability	↓	↓	↓	
1.2B	Dynamic Lateral Stability	↓	↓	↓	
1.4	Stalling Characteristics	Cruise	$M = .85$	↓	12,000
		Power-On	$1.4\ V_{S_{CR}}$	↓	Min. S
		Glide	$1.4\ V_{S_G}$	↓	20
		Land	$1.4\ V_{S_L}$	↓	Min. S
		Power Approach	$1.4\ V_{S_L}$	↓	Min. S
4.6	Power Control System Operation	Power-On	To be specified later	$1.4V_{S_G}-V_{max}$	12

ogram

Press. Alt.	G.W.	C.G. % MAC.	Meas. Req'd (Instr. Groups)	Est. Flt. Hrs.	Est. Calendar Period From	To	Remarks
ds12,000	Normal Service	Result.	4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			5+7	.5	3/54	3/55	
			5+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			5+7	.5	3/54	3/55	
			5+7	.5	3/54	3/55	
,000&30,000			4+7	.5	3/54	3/55	
n. Safe Alt.			4+7	.5	3/54	3/55	
20,000'			4+7	.5	3/54	3/55	
n. Safe Alt.			4+7	.5	3/54	3/55	
n. Safe Alt.			4+7	.5	3/54	3/55	
12,000			4+5+6+7	1.0	3/54	3/55	

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YF-102 No. 52-7994 Flight Test

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CVAC Index No.	Test	Configuration Misc. Items	Trim Speed	Speed Range	Press. Alt.
1.2D	Aileron Control Effectiveness	Power-on		* 1.1 V_{S_G}-Vmax. NRT	12,000
	Aileron Control Effectiveness	Landing		* 1.1 V_{S_L} - 1.4 V_{S_L}	12,000
1.1A	Static Longitudinal Stability	Dive	All Dive Speeds	All permissible speeds above V_{S_G}	From 30,000
1.1C	Dynamic Longitudinal Stability	↓	↓	↓	↓
1.1D	Longitudinal Control Force				
1.1E	Long. Control Effectiveness				
1.1F	Longitudinal Trim Change				
1.2A	Static Directional Stability				
1.2B	Dynamic Directional Stability				
1.2D	Rudder Control Effectiveness				
1.2A	Static Lateral Stability				
1.2B	Dynamic Lateral Stability				
1.1A	Static Longitudinal Stability	Glide	1.4 V_{s_g}	All permissible speeds above V_{S_G} (50 mph increments) All permissible speeds above V_{S_G} (25 mph increments)	12,000 30,000
1.1C	Dynamic Longitudinal Stability	↓	↓	↓	↓
1.1D	Longitudinal Control Force				
1.1E	Long. Control Effectiveness				
1.1F	Longitudinal Trim Change				
1.2A	Static Directional Stability				
1.2B	Dynamic Directional Stability				
1.2D	Rudder Control Effectiveness				
1.2A	Static Lateral Stability				
1.2B	Dynamic Lateral Stability				

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CONFIDENTIAL
SECURITY INFORMATION

est Program

s. .	G.W.	C.G. % MAC	Meas. Req'd (Instr. Groups)	Est. Flt. Hrs.	Calendar Period From	To	Remarks
00	T.O.	Result.	5+7	.5	3/54	3/55	*Trim in steady turns at 50 mph increments
00	T.O.		5+7	.5	3/54	3/55	10 mph increments
m 00	Normal Service		4+7	1	3/54	3/55	
			4+7	1	3/54	3/55	
			4+7	1	3/54	3/55	
			4+7	1	3/54	3/55	
			4+7	1	3/54	3/55	
			6+7	1	3/54	3/55	
			6+7	1	3/54	3/55	
			6+7	1	3/54	3/55	
			5+7	1	3/54	3/55	
			5+7	1	3/54	3/55	
ve V_{S_G} 00			4+7	.5	3/54	3/55	
ve V_{S_G} 00							
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			4+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			6+7	.5	3/54	3/55	
			5+7	.5	3/54	3/55	
			5+7	.5	3/54	3/55	

1813

FILE COPY

om: R. E. Strayer

oject: Flight Test Status Index

purpose of this report is to establish a flight
ck and balance system for each F and TF-102A Airp
erimental Flight Test at Edwards Air Force Base a
ce Base. All test planning reports, number and t
tests are completed and final data reports publis
mer will be entered, balancing the test request.

tus and report entries will be the responsibilit

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CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

Revised

To: Flight Test Planning Engineers

From: R. E. Strayer

Subject: Flight Test Status Index

The purpose of this report is to establish a flight test report index check and balance system for each F and TF-102A Airplane assigned for Experimental Flight Test at Edwards Air Force Base and Holloman Air Force Base. All test planning reports, number and title, will be listed. As tests are completed and final data reports published, the report numner will be entered, balancing the test request.

Status and report entries will be the responsibility of the Planning Engineer.

R. E. Strayer
Flight Test Group Engineer

RES:JPS:mh

EFT-P-8-583

CONVAIR - San Diego

Airplane S/N 52-7994 Page 1

Model YF-102A ZC-8-101

Program Proto. deleted crashed (tests added 7995)

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
1. Low speed taxi tests			X				EF-8-84, Flight Test Summary Flights 1 - 6
2. High Speed Taxi Tests			X				EF-8-84, Flight Test Summary Flights 1-6
3. Shakedown flight test			X				EF-8-84, Flight Test Summary Flights 1-6
4. a. Airspeed Calibration - Low speed b. Airspeed Calibration - High speed				7995			
5. Angle of attack and angle of yaw pickup calibration				7995			
6. Flutter and vibration tests				7995			*EF-8-84, Flight Test Summary Flights 1-6
7. Preliminary airplane performance				7995			*EF-8-84, Flight Test Summary Flights 1-6
8. Dynamic Response Test							
9. Preliminary Flight Characteristics Tests q							*EF-8-84, Flight Test Summary Flights 1-6
10. Fuel Tank Pressurization System Tests							
11. Emergency ram air hydraulic power unit tests							
12. Drogue Chute Deployment Tests							

* Indicates preliminary and/or qualitative.

EFT-P-8-583

CONVAIR - San Diego

Airplane S/N 52-7994 Page 2

Model YF-102A ZC-8-101

Program Proto. deleted crashed (tests added 7995)

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
13. Fuel consumption tests							
14. Emergency fuel system operation tests.							
15. Speed brake tests							*EF-8-84, Flight Test Summary Flts. 1 - 6.
16. Stability and Control Tests							*EF-8-84 Flight Test Summary Flts. 1 - 6
							*Indicates preliminary and/or qualitative.

EFT-P-8-583

CONVAIR - San Diego

Airplane S/N 52-7995 Page 1

Model YF-102A ZC-8-103

Program Phase I, II, III

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
1. Static Thrust Calib.			X				ZC-8-501, Flt. Oper. Rpt.
2. Brake Operation Evaluation			X				ZC-8-501-1,5,6,15,18,Flt.Oper. Report.
3. Engine Control Operation			X				ZC-8-501-9,18,Flt. Oper.Rpt.
4. Speed Brake Operation			X				ZC-8-501-2, Flt. Oper. Rpt.
5. Fuel Tank Pressurization			X				ZC-8-501-4, Flt. Oper. Rpt.
6. Engine Equipment Operation			X				ZC-8-501-2, thru. 20, inc. Flt. Oper. Rpt. EF-8-258 A/B Roughness
7. Flutter and Vibration			X				ZC-8-501-2,3,4,5,6,7, 9,10,16 17, and 20. Flt. Oper. Rpts. ZU-8-013 Results of Flight Flutter Tests.
8. a. EF-8-560 Noise Level Measurement			X				
b. EF-8-591 Cockpit Noise and Vibration			X				ZC-8-501-19 and 20, Flt. Opr. Report
8. Airspeed Calibration			X				ZC-8-501-3,4,6,8,10,12,18,&20 Flt. Oper. Rpts. EF-8-501-SR-2 Summary of Airspeed Calibration Analysis, ZC-8-540

CONVAIR - San Diego

Airplane S/N 52-7995 Page 2

Model YF-102A ZC-8-103

Program Phase I, II, AND III

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
9. Preliminary Flight Characteristics			X				EF-8-442, Longitudinal Stick Force EF-8-614 Dynamic Stability During Rolls. EF-8-416 Vertical Accel. Act.
a. EF-8-520 Airplane Response to High Roll Rate Maneuvers			X				EF-8-558 Roll Rate vs Aileron Angle. EF-8-618 Flight Results During Aileron Rolls.
10. Angle of Attack and Yaw Calibrations				1780			
11. Preliminary Airplane Performance			X				ZC-8-501-7,9,10,&21 Flt. Oper. Rpt. -EF-8-429 Climb Performanc EF-8-354 Climb Performance ZC-8-534
a. EF-8-475 Wing Tip Structural Load			X				EF-8-522 Wing Tip Load Variat. with Mach Num. & Altitude EF-8-472 Wing Tip Strain Gage Data
12. USAF Phase II			X				USAF Phase II Evaluation Rpt. YF-102 S/N 52-7995
13. Missile Bay Door Operation				1780			

CONVAIR - San Diego

Airplane S/N 52-7995 Page 3

Model YF-102A ZC-8-103

Program Phase I, II, AND III

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
14. Airplane Performance			X				EF-8-402 Take-Off Performance EF-8-245 Take-Off Charact.
a. EF-8-595 Landing Loads and Distances			X				ZC-8-501-2,3,8&12, Flt. Oper. Report
b. EF-8-597 Wing Tip - Cambered Not Reflexed			X				EF-8-274 & 274A Drag Polars EF-8-353 Drag Evaluation
15. Engine and Structural Cooling			X				ZC-8-501-2,3,4,5,6,7,8,9,10, 11,13,14,18,19,&20 Flt. Oper. Reports. EF-8-403 Cooling System Analy. EF-8-334 Results of A/B Blanke Installation Cooling Sys.Oper. in YF-102 Airplane
16. Dynamic Response				1780			
17. Stability and Control-Specification Compliance				1791			
18. Pilot Assist System Development				1780			
19. Engine Operation - Air Starts				1781			
20. Drogue Chute Tests			X				ZC-8-501-10, 18, & 20, Flt. Oper. Rpt. EF-8-276 Speed Brake Effect.
21. Emergency Fuel Control System			X				

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Model YF-102A ZC-8-103

Program Phase I, II, AND III

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
22. Ram Air Turbine Tests				1781			Add. 12B Reinstates this test to S/N 52-7995.
23. Flight Control System Operation			X				ZC-8-501-8,9,12,16,17,19,21, 22,23, Flt. Oper. Rpts. ZC-8-512-5,6,&26, Flt. Oper. Rpt. EF-8-323 Control Stick Travel EF-8-420 Shelly Unit with Modified Cam EF-8-287 High Longitudinal Stick Force EF-8-608 Transient Longitud. Oscillograph EF-617 Longitudinal Stick Force ZC-8-519 Flight Control System
24. Engine and Associated Equipment Phase III			X				Operation of the J-57-P11 installed in YF102 and F-102
a. ZC-8-527 Ground Test Combustion Starter			X				ZC-8-512-15 Flt. Oper. Rpt.

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Model YF-102A ZC-8-103

Program Phase I, II, AND III

PHASE I FLIGHT TESTS

FLIGHT TEST ADDENDUM AND MEMO NUMBERS

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
1. Landing Gear Hydraulic Pressure			X				ZC-501-4,12 Flt. Oper. Rpts.
2. Inlet Duct Pressures				X			Superseded by Addendum 16A
3. Dummy Fuel Tank Vent Pressure					X		
4. Inlet Duct Stresses					X		EF-8-234 Engine Inlet Duct Pressures and Stresses
5. Dynamic Directional Stability			X				ZC-8-501-10,15,16,17 Flt. Oper. Report
6. Airspeed Calibration - Transonic				X			Superseded by Addendum 20
7A. Primary Hydraulic System Oil Temperature - With and Without Cooler			X				ZC-8-501-18 &19, Flt. Oper.Rpt ZC-8-516 Hydraulic Sys. Oil Temperature
8. Pressure on Long Nose			X				ZC-8-512-2 Flt. Oper. Rpt. EF-8-626 Elongated Nose Pressure Survey
a. EF-8-592 Radome Glycol Dispersion			X				ZC-8-512-15,16,&29 Flt. Oper. Rpt. EF-8-637 Radome Glycol Anti-Icing System
b. Yellow Canary Pressures			X				ZC-8-501-18&20 Flt.Oper.Rpt. EF-8-574 Evaluation of After-body Pressure Data

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Model YF-102A ZC-8-103

Program Phase I, II, AND III

PHASE I FLIGHT TESTS

FLIGHT TEST ADDENDUM AND MEMO NUMBERS (Continued)

	Future	Current	Completed	Transforred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
10A. Bleed Air Consumption Effect			X				EF-8-247 Thrust Loss Due to Engine Air Bleed EF-8-501-15 Flt. Oper. Rpt.
11. Drag Effects in Region of Mach 0.9			X				ZC-8-501-21 Flt. Oper. Rpt.
12B.Ram Air Turbine Tests			X				ZC-8-512-10 Flt. Oper. Rpt.
13A.Simulated Landings with Ram Air Turbine				X			Superseded by Addendum 11B
14A.Longitudinal Sensitivity			X				ZC-8-501-21 Flt. Oper. Rpt. EF-8-537 Ground Hysteresis Tst
a.			X				ZC-8-501-24 Flt. Oper. Rpt.
15A.Engine Outer Wall and Nozzle Actuator Temperatures			X				ZC-8-501-12,13,14,15,18 &20, Flt. Oper. Rpts.
16A.Compressor Stall Test			X				ZC-8-501-14,20,21,&22 Flt, Oper. Rpt. EF-8-513 Compressor Stall Test EF-8-215 Occurrence of Comp. Surge
17. Engine Oil Temperatures with and without Air-Oil Cooler			X				ZC-8-501-8 and 22, Flt. Oper. Rpt. ZC-8-512-4,&21, Flt.Oper.Rpt.
18. Aft Section Cooling			X				ZC-8-501-19 Flt.Oper. Rpt.

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Model YF-102A ZC-8-103

Program Phase I, II, AND III

PHASE I FLIGHT TESTS

FLIGHT TEST ADDENDUM AND MEMO NUMBERS (Continued)

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flt. Req'd.	REMARKS, REF. MEMOS & REPORTS
19. Instrument Panel Vibration				1791			
20. Airspeed Calibration			X				ZC-8-501-18 & 20, Flt. Oper. Rpt. EF-8-318 Airspeed Calib. Sharp Nose Aircraft ZC-8-540
a. EF-8-428 Relocation of "Q" Pot. Static Source			X				EF-8-642 Shelly Unit Static Press. Source Evaluation
21. Drag Characteristics			X				ZC-8-501-16,18,19,20 & 21 Flight Test Operations. ZC-8-512-27, Flt.Oper.Rpt. ZC-8-501-SR-3
22. Vibration and Acceleration Characteristics			X				ZC-8-501-22 Fl.t Oper. Rpt.
23. Cable Tension Values			X				ZC-8-512-5 Flt. Oper. Rpt.
24A. Buffet Characteristics and Wing Fence Evaluation			X				ZC-8-501-20,21,22, &23 Flt. Oper. Rpt.
a. EF-8-562 Wing Fence Evaluation			X				ZC-8-512-10,11,13,k4,15,16, &18 Flt. Oper. Rpt.
25A. Adverse Yaw and Directional Characteristics			X				ZC-8-512-4, Flt. Oper. Rpt.

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Model F-102A ZC-8-108

Program Dynamic Response/MG-3 AFCS

FLIGHT TEST ITEM AND MEMO NUMBERS	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
1. Airspeed Calibration - Subsonic Speed Range			X				ZC-8-502-1, Flight Operations Report.
2. Airspeed Calibration - Transonic & Supersonic					X		ZC-8-540 Not accomplished
3. Angle of Attack and Angle of Yaw Pickup Calibration							EF-8-294, Preliminary Calib. of Angle of Attack. Incomplete
4. Dynamic Lateral-Directional Stability Without Damping			X				ZC-8-502-2,-3,-4,-6,-7,-8, Flight Operations Report
5. Dynamic Lateral-Directional Stability with Damping			X				ZC-8-502-2,-3,-4,-6,-7,-8, Flight Operations Report.
6. Dynamic Longitudinal Stability Without Damping			X				ZC-8-502-7,-8,-9, Flight Operations Report.
7. Dynamic Longitudinal Stability with Damping			X				ZC-8-502-7,-8,-9, Flight Operations Report.
8. Dynamic Response to Speed Brake Operation					X		Not accomplished
9. Dynamic Response to Armament Bay Door Operation			X				ZC-8-502-3, Flt. Oper. Report.
FLIGHT TEST ADDENDUM AND MEMO NUMBERS							
1. Control System Ground Test			X				ZC-8-502-4, Flight Operations Report.

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Model YF-102 ZC-8-111

Program Phase III, Control Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
1. Dynamic Lateral-Directional Stability			X				ZC-8-503-1,2,5,6,7,8,9,12, Flt. Opers. Rpt.
a. EF-8-515 - Linear Feed back Pot. Environment Tests			X	1798			
2. Directional Dynamic Response			X				ZC-8-503-1,2,5,7,8,9,12 Flt. Opers. Rpt.
a. EF-8-494 - Yaw Rate Gyro Vibration Survey				1791			
b. EF-8-787 - Time Constant & Rate Gain Tests				1787			
3. Dynamic Longitudinal Stability			X				ZC-8-503-2,3,4,5,6,7,8,9,10,11 12.
a. EF-8-515 - Linear Feedback Pot. Environment Tests			X	1798			
b. EF-8-477 - Control System Pitch Oscillation Program				X			EF-8-388-Vertical Acceleration Activity During 1-2 cps Pitch Oscillations.
c. Determination of Most Practical AFT C.G. Limit			X				EF-8-627-Most AFT C.G.Test Flt
d. Bow Spring Tests			X				EF-8-620-Prelim.Eval. of Pitch Damper Bow String Spring. EF-8-639-Eval. of Pitch Damper Bow String Spring.
4. Longitudinal Dynamic Response			X				ZC-8-503,4,5,6,7,8,9,10,11, 12-Flt. Opers. Rpt.
a. EF-8-487 - Damper Pitch Rate Gyro Shock Mount Eval.				1798			
b. EF-8-489 - Damper Amplifier Environmental Tests				1787			EF-8-628-Prelim.Eval. of Elev. Damper Amplifier Tests.
c. EF-8-725 - Electrical Pitch "G" Limiter				1787			EFT-8-6-Present Status of Electronics Pitch "G" Limiter Tests.
d. EF-8-787 - Time Constant & Rate Gain Tests				1787			
e. EF-8-782 - Non Linear Pitch Rate Feedback Pots.			X	1787			EF-636-Prelim.Eval. of Non-Linear Pitch Rate Feedback Test
f. EF-8-786 - Relocated Surface Feedback Pots. Test			X				
5. Full Authority Elevon Control & Preliminary Pilot Assist Tests			X				ZC-8-503-6,7,8,9,10,11,12, Flt. Opers. Rpt.
a. EF-8-342— Pilot Assist System Flight Tests							EF-8-640=Summary of Turn Coord. and Pilot Assist Data.

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Model YF-102 ZC-8-111

Program Phase III, Control Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
5. a. - Continued							EF-8-625-Prelim.Eval.of Pilot Assist Monitor Sys. & Elevon Limit Switch Tests
b. Elevon Surface Pot Temperatures			X				EF-8-610-Elevon Surface Feed-back Potentiometer Temp.
c. EF-8-705 - Ground Tests, Pilot Assist Mode			X				
6. Turn Coordinator Effectiveness of Yaw Damper System							ZC-8-503-1,2,3,6,7,8,9,10,12-Flt. Opers. Rpt.
a. EF-8-335 - Turn Coordinator Tests			X				EF-8-640-Summary of Turn Coord. & Pilot Assist Data EFT-A-16-Lateral Accel. at the Pilots Compartment During Roll maneuvers with turn coord. operating.
b. EF-8-344 - Turn Coordinator Effectiveness			X				
c. EF-8-781 - Turn Coordinator Tests			X				
d. EF-8-781A- Turn Coordinator Tests			X				Supercedes EF-8-781
e. EF-8-793 - Pneumatic Roll Rate Limiter and Turn Coordinator			X				
f. EF-8-575 - Line Lag, in A/S Compensator of Turn Coordinator			X				EF-8-630-A/S Comp.Press.Survey
7. Static Longitudinal Stability				X			
8. Longitudinal Control Forces			X				ZC-8-503-3,4,5,6,7,8,Flt.Oper. Report.
a. EF-8-452 - Elevator & Aileron Break-out forces					X		
b. EF-8-803 - Pilot Assist Beep Trim Authority Eval.				1787			
9. Static Lateral-Directional Stability				X			
10. Angle of Attack and Angle of Yaw Pick-up Condition			X				ZC-8-513-In Flight Calib. of Angle of Attack Indicator.
a. EF-8-312 - Angle of Attack & Angle of Yaw Calib.			X				
b. EF-8-330 - Angle of Attack & Angle of Yaw Determ.			X				EF-8-532-In Flight Calib. of Angle of Attack Indicator.
c. Calibration Procedure for Angle of Yaw & Attack			X				

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Model YF-102 ZC-8-111

Program Phase III, Control Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
11. Airspeed System Calibration					X		
12. Production OAT Survey			X				EF-8-609-Flt.Data Recorded for OAT Indicator Eval. ZC-8-503-3,4
13. ATM Power Capability			X				ZC-8-517-Eval. of the ATM Sys.
a. EF-8-329 - ATM Performance Test Instrumentation Req.			X				
b. EF-8-427 - ATM Cooling & Load Capability			X				EF-8-604-Status of ATM Cap. & Compartment Cooling for Hughes MG-3-FCS.
14. Throttle Idle-Stop Adjustment Check			X		X		Cancelled by Addendum #5
15. Cooling Capability of ATM System			X				ZC-8-517
a. EF-8-427 - ATM Cooling & Load Capability			X				See Item 13, above.
16. ATM System Stability			X				ZC-8-517
17. Ground Operation of the ATM			X				ZC-8-517
18. Water Tightness Tests				X			
19. Power Control Tests				X			
20. Engine Oil System Tests				X			
21. Oil Cooler Tests				X			

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Model YF-102 ZC-8-111

Program Phase III, Control Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
22. Altitude & Fuel Cell Pressure Tests				X			
23. Fuel Availability Tests				X			
24. Emergency Operation of Landing Gear				X			
25. Normal Operation of Landing Gear				X			
26. Electrical System Tests				X			
ADDENDUMS							
1A. Control System Ground Test			X				1A Cancelled Add #1 in its entirety.
2. Boundary Layer Inlet Duct Recovery			X				
3. Temperature of Engine Outer Wall & Nozzle Actuators			X				
4. Flight Test of Pitch "G" Limiter				1787	X		EF-8-601-Prelim. Eval. Pitch "G" Limiter Opers.
a. EF-8-555 - Pitch "G" Limiter				1787			EF-8-478-Flt. Tests of Westing. Pitch "G" Limit.
5. Throttle Idle Stop Adjustment					X		Cancels Item 14, above.
6. Nose Gear Lift Off Speed					X		Cancelled Due to design change
7. Engine Exhaust Nozzle Manual Override			X				ZC-8-517-A/B Nozzle Manual Override Operation.
8. Roll Rate Limiter Development Tests			X				EFT-A-8-24-Evaluation of Rev. Roll Rate Limiter.

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Model YF-102 ZC-8-111

Program Phase III, Control Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
MISCELLANEOUS							
1. EF-8-332 Tape Recorder			X				Instructions
2. Dynamic Response Characteristics for HAC			X				
3. I.L.S. Tests			X				EF-8-632 - ILS - Tests
4. EF-8-539 Decrease in Control Cable Tension with increase in altitude and decreased data.				1795			

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Model YF-102 ZC-8-112

Program 8-82 Arm. Devel. Rckt. MB-1

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
FLIGHT TEST ITEMS (Originally Programmed)							
1. Armament Bay Door Operation			X				ZC-8-504-1,-2,-3,-4,-6,-7
EF-8-541 Arm. Door Cyl. Rod End Loads			X				EF-8-510
EF-8-505 Door Accel. Meas.			X				EF-8-534, ZC-8-512-5
EF-8-506 Arm. Door Oper. Effect on				1788			
2. Armament Launcher Extension Arm Operation			X				ZC-8-504-2,-4,-6,-7,-8,-9
Add. No. 1 Arm. Launcher Operation			X				EF-8-397, EF-8-707
3. Missile Bay & Fwd. Fuse Air Loads				1788			Part Complete. EF-8-594 & 627
4. Captive Missile Loads					X		Part Complete. ZC-8-504-8-9
5. Missile Separation			X				ZC-8-504-4,-6,-8,-9
Add. 2 & 2A Missile Sep. & Traj. at V_{max} & Hp Max.			X				ZC-8-512-1,-19
6. Missile Trajectory			X				ZC-8-504-4,-6,-9
7. Rocket Separation and Dispersion			X				ZC-8-504-7
Add. 3 Two-inch Rocket Package					X		ZC-8-512-25,-23
8. Airspeed System Calib.				1779			
9. Dynamic Response to Arm. Operation			X				ZC-8-504-1,-2,-3,-4,-6,-7,-8,-9
10. Rocket and Missile Exhaust Gas Effects			X				ZC-8-504-6,-7,-9,-4 ZC-8-512-17, EF-8-599

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Airplane S/N 53-1781 Page 2

Model Y F-102 ZC-8-112

Program 8-82 Arm. Devel. Rckt. MB-1

FLIGHT TEST ITEMS (Originally Programmed)	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
EF-8-437 Fuel Venting Press. Meas.			X				ZC-8-508, ZC-8-504-6,-7,-8 ZC-8-512-1
Add. 12 & 13 Preliminary R.A.T. Tests			X				ZC-8-504-1,-2,-3
ZC-8-103 Preliminary R.A.T. Tests			X				

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Model YF-102 ZC-8-131

Program 8-82 Arm. Devel. Rckt MB-1

FLIGHT TEST ITEM NO. (Reinstated Program)	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
1. TETHERED TEST FIRING			X				EFT-A-8-144 & 123 ZC-8-512-
a. EFT-P-8-129 Ramp Jettison Test of the MB-1 Launcher on S/N 1781			X				EF-313-68 MB-1 Ground Jettison Tests.
2. JETTISON INERT ROCKET			X				ZC-8-512-48 2nd jettison cancelled by EFT-P-8-249
3. SEPARATION & TRAJECTORY					X		EFT-A-8-137, 144 & 159 ZC-8-512-52, 55 & 64 EFT-P-8-708 (Two Rockets Not Fired)
4. AIRPLANE RESPONSE TO ROCKET FIRING			X				EFT-A-8-169, 183
5. BLAST EFFECTS OF FIRING MB-1			X				EFT-A-8-266
6. ENGINE SNUFFING				1797			EFT-P-8-754 (Last Two Rockets)
7. EXHAUST DUST EFFECTS			X				EFT-A-8-212
Adden. #1 MB-1 Motor Skin Temp.			X				EFT-A-8-159

*Note: Flt. Test Items 4, 5, 6 and 7 will be conducted with Flt. Test Item 3.

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Model YF-102A

Program Antenna Pattern/MG-3 System Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
FLIGHT TEST ITEMS AND MEMO NUMBERS							
PHASE I FLIGHT TESTS							
1. Antenna - Functional Characteristics - AN/ARN-18 Glide Path			X				ZC-8-505-3, Flight Oper. Rep.
2. Antenna - Functional Characteristics - AN/ARN-14 Omni			X				ZC-8-505-1,-2, Flight Operations Report.
3. Antenna - Functional Characteristics - AN/ARN-25 UHF/DF			X				ZC-8-505-3, Flight Operations Report.
4. Antenna - Functional Characteristics - AN/ARC-34			X				ZC-8-505-2, Flight Operations Report.
5. Antenna - Functional Characteristics - AN/ARN-12			X				ZC-8-505-2, Flight Operations Report.
FLIGHT TEST ADDENDUM AND MEMO NUMBERS							
1. Data Link Antenna - Isolation Study				1783			
2. Glide Path Antenna - Location Study			X				ZC-8-505-3, Flight Operations Report.

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Model Y F-102A

Program Phase III, Cooling, Electrical, etc.

FLIGHT TEST ITEM AND MEMO NUMBERS	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
1. Angle of Attack & Angle of Yaw Pickup Calib.					X		
2. Airspeed Calibration - Subsonic Speed Range			X				ZC-8-506-9 Flight Operations Report , ZC-8-540 EF-8-463, 5-Foot Nose Boom Airspeed Sys. Calib.
3. Outside Air Temperature Pickup Location Study			X				ZC-8-506-7, Flight Oper.Report EF-8-464, Outside Air Temp.Calib
4. Ground Test of Wing, Tail & Engine Inlet Anti-icing System			X				ZC-8-506-6, Flight Oper.Report
5. Wing, Tail & Engine Inlet Anti-Icing System and Windshield ("NESA") Anti-Icing System. (Climb and Descent Tests).			X				ZC-8-506-6,-7, Flight Oper. Report.
6. Wing, Tail & Engine Inlet Anti-Icing System and Windshield ("NESA") Anti-Icing System. (Level Flight Tests)			X				ZC-8-506-6,-7,-8, Flight Operations Report
7. Radome Anti-Icing			X				ZC-8-506-8,-9, Flight Operations Report
8. Windshield and Canopy Defogging System			X				ZC-8-506-6-7, Flight Oper. Report
9. Heating & Ventilating Equipment Specification Checkout.			X				
10. Cabin Pressure Specification Checkout			X				ZC-8-506-4, Flight Oper.Report
11. Power Plant & Structural Cooling Specification Compliance					X		Not accomplished

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Model Y F-102A

Program Phase III, Cooling, Electrical, etc.

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
FLIGHT TEST ITEM AND MEMO NUMBERS							
12. Dynamic Response					X		Not accomplished.
13. Pilot Assist System							Not Required.
FLIGHT TEST ADDENDUM AND MEMO NUMBERS							
1A. Temperature of Engine Outer Wall and Nozzle Actuators			X				ZC-8-506-2-3, Flight Oper.Rep. EF-8-334,Results of A/B Blanket Installation.
2. Windshield Canopy Defog System Test			X				ZC-8-506-7, Flight Oper. Rep.
3A. Cabin Air Temperatures			X				ZC-8-506-4,-5,-6,-7 Flight Operations Report
4. Antenna Flight Test			X				ZC-8-506-2-3-5, Flight Oper. Report.
5. Performance of Ram Air Turbine Emergency Hydraulic Power Unit			X				ZC-8-506-1,-2,-8, Flight Operations Report.
5A. Performance of Modified Ram Air Turbine Emergency Hydraulic Power System.				X			ZC-8-506-8, Flight Operations Report, Transferred to S/N 52-7995.
6. Fin Ram Air Inlet Pressure & Temperature Measurements.				X			Transferred.

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Model YF-102A ZC-8-118

Program 8-90 Flight Test Techniques Devel.

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
Stage I: Ground Tests							
1.01 Engine Operation			X				ZC-8-511-1
1.02 Engine Equipment Operation			X				
EF-8-553, Starter Ground Tests			X				Data Distributed (No memo)
1.03 Static Thrust Calibration			X				EF-8-499
1.04 Low Speed Taxi and Nose Wheel Steering			X				ZC-8-511-1
1.05 High Speed Taxi and Lift-Off Characteristics			X				ZC-8-511-1
State II: Aircraft System Shakedown							
EF-8-540, Performance Evaluation of Whittaker Fuel Flow Equalizer			X				303-2-012
2.01 Shakedown Flights			X				ZC-8-511-1
EF-8-449, Hydraulic System Temperature Measurements			X				Data Distributed (No Memo)
2.02 Flight Control System Operation			X				EF-8-633
2.03 Emergency Fuel Control			X				Data Distributed (No Memo)

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Airplane S/N 53-1787 Page 2

Model F-102A ZC-8-118

Program 8-90 Flight Test Techniques Devel.

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS Stage II: Aircraft Systems Shakedown (Contd.)							
2.04 Engine and Structural Cooling			X				EFT-A-8-121
EF-8-476, Structural Temperature Measurements AFT Fuselage Section			X				303-2-004
EF-8-486, Measurements Pressure Across Eng. Shroud, Sta. 620			X				303-2-005
EF-8-519, Revision of Aft Fuselage Temperature Instruments			X				303-2-011
EF-8-557, Measurements of Oil Cooler Airflow & Associated Structural Pressure and Temperature				1791			To A/C 1791
EF-8-568, Investigation of Nozzle Act. Cyl. Temps. with Flap Type A/B				1791			To A/C 1791
EF-8-593, Tests with Engine Diffuser Heat Shield Removed				1791			To A/C 1791
2.05 Brake Parachute Operation			X				EF-8-602, EF-8-623
2.06 Automatic Flight System			X				EF-8-633
EF-8-716, Trim Servo Evaluation			X				ZC-8-512-28
EF-8-716-1 Trim Servo Evaluation			X				ZC-8-512-28
EF-8-794, Turn Coordinator Tests			X				ZC-8-512-8, ZC-8-512-28
EFT-P-8-71, Install Programmer & Parameter Control Box			X				EFT-A-8-33
EFT-8-714 Rudder Control Sys. Hi-Gain Control Valve			X				EFT-A-8-40
(Not Assigned), H.E.P. Valve Evaluation				1798			
2.07 Control Sensitivity			X				303-2-003
EF-8-535, Elevon Control Force Link Tests				1791			
EF-8-586, Differential Linkage Evaluation Tests				1791			
Stage III: Airspeed Calibration							
EFT-P-8-74, Alternate Airspeed System Calibration			X				ZC-8-512-46, Memo in prep. EFT-A-8-91 ZC-8-537
3.01 Airspeed Calibration (Pacer Method)			X				Memo: EF-8-465 EFT-A-8-242 ZC-8-512-68-86

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Model F-102A ZC-8-118

Program 8-90 Flight Test Techniques Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
Stage III: Airspeed Calibration (Continued)							
3.02 Airspeed Calibration (Fly-by Method)			X				Report: ZC-8-511-1
3.03 Chin Nose Boom Study - EFT-P-8-605				F-106A			Cancelled
EFT-P-8-511			X				EFT-A-8-242, 231 ZC-8-537 EFT-A-8-321, ZC-8-512-88-89 ZC-8-537
EFT-P-8-686					X		
Stage IV: Airplane Performance							
4.01 Take off			X				EFT-A-8-17
4.02 Climb			X				EFT-A-8-17
4.03 Level Flights			X				EFT-A-8-17
4.04 Dive				1791			
4.05 Landing			X				EFT-A-8-17
Stage V: Preliminary Flight Characteristics							
5.01 General Handling Characteristics			X				ZC-8-511-1
5.02 Dynamic Longitudinal Stability			X				ZC-8-512-20
EF-8-780, Manual Mode Pitch Damper Tests			X				Data Distributed (No Memo)

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Model F-102A ZC-8-118

Program 8-90 Flight Test Techniques Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS (Continued)							
Stage V: Preliminary Flight Characteristics (Continued)							
5.03 Static Longitudinal Stability			X				EF-8-552, EF-8-583
5.04 Maneuvering Flight Characteristics			X				EFT-A-8-52
EF-8-832, Determination of Buffet Boundaries			X				EFT-A-8-36
EFT-P-8-28 Transonic Buffet Tuft Survey			X				Data Distributed (No memo)
EFT-P-8-172 Transonic Buffet Survey			X				Data Distributed (No memo)

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Model F-102A ZC-8-117

Program 8-90 Flight Test Techniques Devel.

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
Stage V: Preliminary Flight Characteristics (Continued)							
5.05 Dynamic Directional Stability			X				ZC-8-512-5, ZC-8-512-20, ZC-8-512-25
EFT-P-8-35, Evaluation of Windshield L. E. (Bulge) Fairing			X				Data Distributed (No Memo)
5.06 Static Directional Stability (Sideslip)			X				ZC-8-512-27, EF-8A-1
5.07 Lateral Control (Aileron Rolls)			X				ZC-8-512-21, ZC-8-512-25
EF-8-488A, Phase I Stability & Control Tests			X				Revised Procedures for Tests 5.02 through 5.07
PHASE I FLIGHT TESTS PHASE II FLIGHT TESTS PHASE III FLIGHT TESTS			X				By USAF
Stage VI: Systems Development							
6.01 Boundary Layer Bleed Air				8-10			
6.02 Braking System				8-10			
6.03 Low Press Pneumatic System				8-10			

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Model F-102A ZC-8-117

Program 8-90 Flight Test Techniques Develop.

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
PHASE III FLIGHT TESTS							
STAGE VI: Systems Development (Continued)							
6.04 Ram Air Turbine Tests							Procedure Revised by EF-8-814
EF-8-814, Ram Air Turbine Tests			X				Data Distributed (No Memo)
EFT-P-8-72, Ram Air Turbine Tests			X				EFT-A-8-73 ZC-8-532
6.05 Pilot Assist System			X				Test Proc. Rev. by Memo Listed
EF-8-722, Pilot Assist Tests			X				EFT-A-8-38
EF-8-722-1, Pilot Assist Tests (Invest, Lat. Instability)			X				EFT-A-8-136
EFT-P-8-37, Electronic Pitch g-Limiter Tests			X				EFT-A-8-78
EFT-P-8-48, Damper Amplifier Drift Tests			X				EFT-A-8-94
EFT-P-8-52, Pilot Assist Beep Trim Development Tests			X				EFT-A-8-136
EFT-P-8-57, Westinghouse g-Limiter Evaluation			X				EFT-A-8-67
EFT-P-8-66, Pilot Assist Up-Elevator Function Testing			X				EFT-A-8-136
EFT-P-8-79, Low Force Lockout Flight Test Evaluation				1798			Ref. EFT-P-8-155
EFT-P-8-128, Pitch and Roll Attitude Gain Optimization				1798			
EFT-P-8-106, Revisions to Pilot Assist System Config.			X				Work Completed.
EFT-P-8-144, Evaluation of Statham Angular Accelerometer			X				Data Distributed (No Memo)
EFT-P-8-155, Low Force Lockout Flight Tests				1798			
EFT-P-8-176, Evaluation of Lagged Attitude-Plus-Rate Method for Obtaining Pitch Attitude			X				EFT-A-8-136
EFT-P-8-185, Install. M-H Hig. 4 gyro & M-H w/Doelcam gyro			X				Service Life Eval.
EFT-P-8-206, Evaluation of Lateral Accel. Protection Device			X				EFT-A-8-73
EFT-P-8-211, Evaluation of Kearfott Vertical Gyro				1798			
EFT-P-8-241, Tests with Pitch g-Limiter tied in Operationally with Pilot Assist System			X				DP-8-15

EFT-P-8-583

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Model F-102A ZC-8-117

Program 8-90 Flight Test Techniques Devel.

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE III FLIGHT TESTS							
Stage VII: Dynamic Response							
EF-8-520, Airplane Response to High Roll Rate Maneuvers			X				EFT-A-8-31
EF-8-572, Add. Aileron Roll and Stability Tests			X				EFT-A-8-31
EF-8-572A, Add. Aileron Roll and Stability Tests			X				EFT-A-8-31
EF-8-778, Phase III Roll Testing (Roll Coupling)			X				EFT-A-8-31
EF-8-778-1, Phase III Roll Testing (Roll Coupling)			X				EFT-A-8-31
EF-8-778-2, Phase III Roll Testing (Roll Coupling)			X				EFT-A-8-66
EFT-P-8-44, Add. Roll Coupling Tests with Sideslip F.B.			X				EFT-A-8-66
EFT-P-8-112, Add. Roll Coupling Tests to Determine Max. Roll Rate Limits			X				EFT-A-8-66
Stage VIII: Stability and Control							USAF Spec. R1815-B Compliance
8.01 Dynamic Longitudinal Stability				1791			
8.02 Static Longitudinal Stability				1791			
8.03 Elevator Control Effectiveness				1791			
8.04 Maximum C_L Landing				1791			
8.05 Take-Off				1791			
8.06 Maximum Load Factors				1791			
8.07 High Speed Dive Requirements				1791			
8.08 High Speed Flight Requirements				1791			
8.09 Longitudinal Trim Changes				1791			
8.10 Maneuvering Flight Characteristics				1791			
8.11 Dynamic Lateral-Directional Stability				1791			
8.12 Static Directional Stability				1791			
8.13 Adverse Yaw				1791			
8.14 Rudder Control Effectiveness				1791			
8.15 Aileron Characteristics				1791			
8.16 Stall				1791			
8.17 Trimming Devices				1791			
				1791			

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Airplane S/N 53-1787 Page 7

Model F-102A ZC-8-118

Program 8-90 Flight Test Techniques Devel.

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
ADDENDUM ITEMS							
No. 1, High Speed Vibration (Buzz) Investigation			X				ZC-8-542
EF-8-521, Air Loads and Deflection of Nose Wheel Door			X				
EF-8-554, A/C Modification for High Speed Vibration Program			X				
EF-8-567, Flight Procedure for "Buzz" Investigation			X				
EF-8-569, Add. A/C Mod. for High Speed Buzz Program			X				
EF-8-569-1, Add. A/C Mod. for High Speed Buzz Program			X				
EF-8-569-2, Add. A/C Mod. for High Speed Buzz Program			X				
EF-8-569-3, Add. A/C Mod. for High Speed Buzz Program			X				
EF-8-569-4, Add. A/C Mod. for High Speed Buzz Program					X		
EF-8-568-5, Add. A/C Mod. for High Speed Buzz Program			X				
EF-8-701, Vibration Survey			X				
No. 2, Duct Rumble Investigation				1791			ZC-8-542
No. 3, Engine Compressor Stall Testing				1380			
Ejector Tail Pipe Evaluation, ZC-8-177			X				EFT-8-199, -265
No. 4, Wing Fence Evaluation, EFT-P-8-571			X				ZC-8-512-87, EFT-A-8-328
No. 5, Wing Vent Press. Evaluation, EFT-P-8-596			X				EFT-A-8-202, ZC-8-512-69
No. 6, Alt. O.A.T. Evaluation, EFT-P-8-530			X				EFT-A-220, 237-ZC-8-572-72-75
No. 7, Flight Test Technique Development			X				
Elevon Position Measurement, EFT-P-8-622			X				
Determination of Aerodynamic Stability Derivatives, EFTP-8-619				1398			
No. 8, Angle of Attack, Topp. Instr., Evel. EFT-P-8-569			X				EFT-A-8-355
No. 9, Comparison of Skin Thermocouples, EFT-P-8-675			X				EFT-A-8-334
No. 10, Doelcam, HIG-4 Gyro Life Test, EFT-P-8-543, 185			X				

EFT-P-8-583

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Model F-102 ZC-8-119

Program 8-90 Armament

FLIGHT TEST ITEM & MEMO NUMBERS	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts Req'd.	REMARKS, REF. MEMOS & REPORTS
1. HARDWARE CHECK			X				EF-8-638 Fast Acting Armament Gear Operation F-102A S/N 53-1799. EF-8-707 Armament Gear Load
a. EF-8-565 Instrumentation Required to Lift Armament Restrictions.			X				EF-8-707 Armament Gear Load
b. EF-8-502 Armament Hardware Ground Test			X				ZC-8-512-8 Flt. Oper. Rpt.
c. EF-8-511 Measurement of Rocket Door Aero-Elastic Stability			X				EF-8-585 Rod End Loads EF-8-504 Missile Bay Rod End Loads ZC-8-512-11 Flt. Oper. Rpt.
2. MISSILE SEPARATION & TRAJECTORY			X				ZC-8-512-54, 13, - EFT-A-8-330
a. EF-8-578 F-102 & TF-102 Missile Requirements			X				
3. ROCKET SEPARATION			X				ZC-8-512-23, Flt. Operation Report, ZC-8-512-4, 10, 12, 24

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Model F-102

Program 8-90 Armament

FLIGHT TEST ITEM AND MEMO NUMBERS	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
4. Rocket Trajectory and Dispersion			X				ZC-8-512-25 Flt. Oper. Rpt.
a. EF-8-791 HAC Two-Inch Rocket Program			X				ZC-8-512-23, -40, -45, -48, -12 EFT-A-27
b. EF-8-802 Measurement of Blast Pan & Flipper Door Temp.					X		Rocket Program Changed From 2.75" to 2.00"
5. Armament Bay Airloads			X				EF-313-2, ZC-8-512-11
6. Airplane Response			X				EF-8A-5 Dynamic Response on F-102A S/N 53-1788
a. EF-8-531 Mainlanding Gear Door Link Loads			X				EF-8-643 Main Landing Gear Door Links Loads Invest.
b. EF-8-704 Angle of Attack Measurements			X				Insufficient & inconclusive Data. EF-8-A-3 F-102A 8-10 Angle of Attack Calibration
c. EFT-P-8-22 Angle of Attack Calibration			X				ZC-8-512-57, -75
							EFT-A-8-127 and EFT-A-8-219
7. Exhaust Dust Effects (Missile Exhaust) (Rocket Exhaust)			X X				ZC-8-512-17 Flt. Oper.Rpt.

EFT-P-8-583

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Airplane S/N 53-1788
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Model F-102
ZC-8-119
Program 8-90 Armament

FLIGHT TEST ITEM AND MEMO NUMBERS	Future	Current	Completed	Transforred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
ADDENDUM ITEMS							
1. Visual Evaluation of Windshield Rain Clearance				X			Transferred to another aircraft due to lack of rainfall in Holloman area.
2. Drag Chute Deployment				1787			
3. HAC Captive Missile Test							Superseded by Addendum 3A
3A. HAC Captive Missile Test			X				ZC-8-512-24
4. Evaluation of 8-90 Emergency Alternator During Engine Start			X				EFT-A-8-23, ZC-8-512-36
							Letter: From W. A. Schmidt (Elect. Group) to W. W. Fox/ Strayer - dtd 17 Oct. 1955
5. Missile Trajectory Program				1793			ZC-8-512-54
a. EFT-P-8-367 Exposure of Falcon Missiles Equipped with GAR-1B Nirdomes to Missile Exhaust Gases			X				Nirdomes turned over to Hughes Aircraft Company EFT-A-8-330
6. 2.75" Rocket Jump Angle			X				ZC-8-512-90, -13, -23

EFT-P-8-583

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Model F-102A ZC-8-120

Program 8-10 Extend, Ramp Phase I, III, and IV

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
Test Item 1.00 Ground Tests							
1.01 Engine Operation			X				EFT-A-8-110
1.02 Engine Equipment Operation							
EF-8-809 Determination of g at which AC frequency can be maintained					X		EFT-P-8-658
EFT-P-8-3 Flight Test of Sundstrand Constant Speed Drive Oil System at Red. Engine Oil Tank Press.					X		EFT-P-8-658
EFT-P-8-5 Anti-Surge Bleed Valve Operational Limit Test on J-57-P-23 Engines					X		EFT-P-8-658
EFT-P-8-96 Cancellation of Sundstrand Drive System Tests from ZC-8-120			X				
EFT-P-8-97 Sundstrand Drive Recycling Tests					X		EFT-P-8-659
T-P-8-214 Stratos Unit Tests			X				EFT-A-8-188
1.03 Static Thrust Calibration			X				
1.04 Low Speed Taxi and Nose Wheel Steering							
EFT-P-8-43 Main Landing Gear Flexibility			X				
1.05 High Speed Taxi and Lift-Off Characteristics			X				
Test Item 2.00: Aircraft Systems Functional Tests							
EFT-P-8-140A MLG Sidebrace Loads			X				EFT-A-8-113,113A,ZC-8-512-55-66
EFT-P-8-134A Prod. Config. Sideslip Feedback Device			X				
EFT-P-8-163 MLG Fore & Aft Resonant Frequency							
EFT-P-8-228A Landing Gear Operating Time					X		EFT-P-8-658
EFT-P-8-231 Landing Gear Position & Warning Circuit			X				
EFT-P-8-232 Natural Vibration Freq. of Ldg. Gear			X				EFT-A-8-117
EFT-P-8-747 Wheel Well Ambient Pressure Studies			X				EFT-A-350,ZC-8-512-95

EFT-P-8-583

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Airplane S/N 53-1791 Page 2

Model F-102A ZC-8-120

Program 8-10 Extend, Ramp Phase I, III, and IV

		Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS								
2.01	Functional Test Flight			X				
EFT-P-8-45	Buzz. Elimination of, on S/N 53-1791			X				
EFT-P-8-61	Addendum 1 to EFT-P-8-45			X				
EFT-P-8-94	Addendum II to EFT-P-8-45			X				
2.02	Flight Control System Operation			X				
EFT-P-8-8	Spec. for Manual Mode Pitch Damper Checkout					X		
EFT-P-8-27	Flight Test Of Interim Trim Servo System					X		
EFT-P-8-36	Cancellation of EFT-P-8-8			X				
EFT-P-8-55A	Yaw Rate Gyro Vibration Survey					X		
2.03	Emergency Fuel Control			X				
2.04	Engine and Structural Cooling			X				
EF-8-796	Shelley Unit Compartment Static Press.			X				
EF-8-800	Tests with J-57 Diffuser Heat Shield Removed					X		
EF-8-825	Air-Oil Cooler Air Flow Optimization					X		
EFT-P-8-187	Deletion of Air-Oil Cooler Air Flow Optim. Tests			X				
EFT-P-8-87	Flap Type A/B, Temps.					X		
2.05	Brake Parachute Operation			X				
EFT-P-8-54A	Drag Chute System Deployment Time			X				
EFT-P-8-54B	" " " " "			X				
2.06	Pilot Assist System			X				
EF-8-805	Spec. for Immediate Pilot Assist and Turn Coordinator Checkout			X				

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Model F-102A ZC-8-120

Program 8-10 Extend, Ramp Phase I, III, and IV

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
2.07 Control Sensitivity			X				
EF-8-776 Determination of Hydraulic Gage Needle Fluctuation as Related to Airplane Control Stick and Rudder Movement.			X				EFT-A-8-9
Test Item 3.00 Airspeed Calibration			X				
EF-8-723 Airspeed Calibration Points					X		EFT-A-8-8
EFT-P-8-65 Determination of the Time Lag Constant of the F-102A Static Pressure System Via Ground Test on S/N 53-1791			X				
3.01 Airspeed Calibration (Pacer Method)			X				EFT-A-8-8
3.02 Airspeed Calibration (Fly-by Method)			X				EFT-A-8-8
Test Item 4.00: Flight Flutter Tests			X				
EF-8-807 In-Flight Vibration							
EF-8-785 Elimination of Low Speed Vibration					X		EF-8A-2
EFT-P-8-4 Cancellation of EF-8-785			X				
EFT-P-8-46 External Fuel Tank Flight Tests					X		Cancelled by EFT-P-8-658
EFT-P-8-188 Revised F-102A Flight Flutter Program			X				
4.01 Determination and Investigation of Critical Airplane Vibration Modes.			X				

NOTE: Only the portion covered by EFT-P-8-46 cancelled. External tank program covered by ZC-8-173.

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Model F-102A ZC-8-120

Program 8-10 Extend, Ramp Phase I, III, and IV

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
Test Item 5.00: Airplane Performance							
5.01 Take-off			X				
5.02 Climb			X				
5.03 Level Flight			X				
5.04 Dive						4	
5.05 Landing			X				
Test Item 6.00: Preliminary Flight Characteristics			X				
EF-8-806 Maneuvers to Check Stability and Control Instrumentation			X				
EF-8-819 Differential Linkage Evaluation Tests			X				
6.01 General Handling Characteristics			X				
6.02 Dynamic Longitudinal Stability						1	
6.03 Static Longitudinal Stability			X				
6.04 Maneuvering Flight Characteristics						2	
6.05 Dynamic Directional Stability						1	
6.06 Static Directional Stability (Sideslip)						1	

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Airplane S/N 53-1791 Page 5

Model F-102A ZC-8-120

Program 8-10 Extend, Ramp Phase I, III, and IV

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
6.07 Roll Coupling Tests							EFT-A-8-103
EF-8-788 Installation of Telemetering Equipment on S/N 53-1791 to permit Flight Testing on Roll Coupling Problem			X				
PHASE IV FLIGHT TESTS (U.S.A.F.)							
PHASE III FLIGHT TESTS							
Test Item 7.00: Stability and Control			X				
EF-8-790 Air Loads and Deflections of Nose Wheel Door			X				EFT-A-8-59
EF-8-795 Determination of Dive Brake Effectiveness			X				" " "-29
EF-8-795A Determination of Dive Brake Effectiveness			X				" " " "
EFT-P-8-2 Determination of Dive Brake Temp.			X				
EFT-P-8-53 Controls Testing, Transfer of from S/N 53-1791to 1798.			X				
EFT-P-8-117 Rudder Effectiveness Flight Tests with Increased Area (Production) Speed Brakes Installed					X	2	
EFT-P-8-431 Air Loads & Deflections of the Nose Wheel Door					X		
EFT-P-8-677A Pitch Damper Tests (Completed)			X				EFTA-218, 245 264
EFT-P-8-694A In-flight Calibration of the RWT vs Nose Boom Angle of Attack Vane. (Current)							ZC-8-512-74-77-80 EFT-A-8-219-ZC-8-512-75
7.01 Dynamic Longitudinal Stability				1798			
7.02 Static Longitudinal Stability				1798			
7.03 Elevator Control Effectiveness				1798			
7.04 Maximum C_L Landing				1798			
EFT-P-8-638 High Speed Lateral Directional Oscillations			X				
EFT-P-8-661 Ground Tests to Determine Small Displacement Characteristics, of the Aileron Control System			X				EFT-A-8-210; ZC-8-512-7[illegible]

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Airplane S/N 53-1791 Page 6
Model F-102A ZC-8-120
Program 8-10 Extend, Ramp Phase I, III, and IV

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE III FLIGHT TESTS							
Test Item 7.00: Stability and Control - Continued							
7.05 Take-off				1798			
7.06 Maximum Load Factors				1798			
7.07 High Speed Requirements				1798			
7.08 High Speed Flight Requirments				1798			
7.09 Longitudinal Trim Changes				1798			
7.10 Maneuvering Flight Characteristics				1798			
7.11 Dynamic Lateral - Directional Stability				1798			
7.12 Static Directional Stability				1798			
7.13 Adverse Yaw				1798			
7.14 Rudder Control Forces and Effectiveness				1798			
7.15 Aileron Characteristics				1798			
7.16 Stall Characteristics				1798			
7.17 Trimming Devices				1798			
Test Item 8.00: Noise Level (Cockpit)			X				
Test Item 9.00: Carbon Monoxide (Cockpit)					X		
Test Item 10.00: Spin Test					X		
EFT-P-8-536 Spin Test Instrumentation							
Test Item 11.00: Cockpit and Inclosure Watertightness					X	1	
Test Item 12.00: Cabin Pressurization					X		
Addendum 1.00: Control System Ground Test				1798			
EFT-P-8-53 Controls Testing Transfer of, from S/N 53-1791 to S/N 53-1798			X				
ZC-8-173 External Fuel Tank Tests			X				ZC-8-512-99

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Airplane S/N 53-1791 Page 1

Model F-102A

Program ZC-8-286

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
ARDC Advanced Instrument Display		X					ZC-8-286

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Model F-102A

Program Armament Phase III

FLIGHT TEST ITEM NUMBER	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
1.0 Armament Hardware			X				EFT-A-8-391 ZC-8-512-82
1.01 Structural Qualification of Armament Displacement Hardware			X				EFT-P-8-710 EFT-A-8-274
EFT-P-8-869 Missile Launcher Loads			X				ZC-8-512-82
1.02 24 Rocket Door Configurations			X			*	EFT-P-8-710, EFT-A-8-391
1.03 High Pressure Pneumatic System			X			2	ZC-8-512-117
2.0 Armament Firing			X				EFT-A-8-414
2.01 Missile Separation Verification			X				EFT-A-8-406
EFT-P-8-831 Missile Firings			X				
2.02 Verify 8-82 and 8-90 Ballistic Data							
Demonstrate performance of A/C as a launching platform. EFT-P-8-1064			X				EFT-A-8-383
2.03 Missile and Rocket Blast Effects			X			**	EFT-A-8-383
EFT-P-8-706 Contamination in Cockpit and Electronic Bay During Missile Firing			X				
EFT-P-8-712 Preliminary Check to Cabin Alt with Pressurization System Inoperative			X				EFT-A-8-273
EFT-P-8-841 Effect of 2.75" Rocket Exhaust on GAR-1, -2			X				
2.04 Armament Bay Airloads			X				EFT-A-8-363

* Concurrent with 1.01

** Concurrent with 2.01 and 2.02.

A

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Airplane S/N 53-1793 Page 2

Model F-102A ZC-8-242

Program Armament Phase III

FLIGHT TEST ITEM NUMBER	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
2.05 STUDY TEMPERATURE RISE OF EXTENDED HARDWARE			X				
3.01 IN-FLIGHT CALIBRATION OF THE RELATIVE WIND TRANSDUCER VS TRUE ANGLE OF ATTACK				1791			EFT-P-8-694 EFT-P-8-755
EFT-P-8-1066 Side Strike Camera Buffet Investigation		●	X				
EFT-P-8-1092 Doelcam Gyro Package Test					X		

1. Concurrent with 2.02

EFT-P-3-582

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Model F-102A ZC-8-100

Program Armament Rocket

FLIGHT TEST ITEMS	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
A. GROUP I FLIGHT TESTS - WITHOUT XMG-14 F.C.S.							
Test Item 1.00 Armament Hardware			X				
1.01 Armament Load Check			X				
1.02 Rocket Flyaround Check			X				EFI-P-[illegible]-491 sets [illegible] parameters.
1.03 Rocket Jettison Check			X				
EFT-P-[illegible]42 (Revised) Missile Bay Door Flutter Investigation			X				
Test Item 2.00 MB-1 Firing				[illegible]			
2.01 Separation and Trajectory							
2.02 Blast Effects							
2.03 A/C Response to Firing							
2.04 Engine Response to Firing							
2.05 Exhaust Dust Effects							
B. GROUP II FLIGHT TESTS - WITH XMG-14 F.C.S.							
Test Item 3.00 XMG-14 System					X		[illegible] Miss-evaluation
3.01 XMG-14 Checkout					X		

EFT-P-[illegible]-583

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Airplane S/N 53-1797 Page 1

Model F-102A ZC-8-166

Program Armament Rocket

FLIGHT TEST ITEMS	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
A. GROUP I FLIGHT TESTS - WITHOUT XMG-14 F.C.S.							
Test Item 1.00 Armament Hardware			X				
1.01 Armament [illegible] Check			X				
1.02 Rocket Flyground Check			X				EFT-P-[illegible]-491 sets up flight parameters.
1.03 Rocket Jettison Check			X				
EFT-P-[illegible] (Revised) Missile Bay Door Flutter Investigation			X				
Test Item 2.00 Mb-1 Firing				1-00			
2.01 Separation and Trajectory							
2.02 Blast Effects							
2.03 A/C Response to Firing							
2.04 Engine Response to Firing							
2.05 Exhaust Dust Effects							
B. GROUP II FLIGHT TESTS - WITH XMG-14 F.C.S.							
Test Item 3.00 XMG-14 System					X		EFT-P-[illegible] Miss-evaluation
3.01 XMG-14 Checkout					X		

EFT-P-8-583

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Airplane S/N 53-1797 Page 2

Model F-102A ZC-8-166

Program Armament Rocket

FLIGHT TEST ITEMS	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
B. GROUP II FLIGHT TESTS - WITH XMG-14 F.C.S. - (continued)							
Test Item 4.00 MB-1 Firing				1806			
4.01 MB-1 Firing at Target				1806			
ADD. I - ZC-8-166							
Test Item 1.1 Landing Gear Operating Time			X				
Test Item 1.2 Main Landing Gear Wing Fairing Link Loads			X				
Memo EFT-P-8-548 - Ram Air Turbine Cylinder Rod Loads				1806			EFT-P-8-548 (Rev.) transferred to 1806.
Memo EFT-P-8-527 - Determination of In-Flight Loads on the MLG Wing Fairing Link Loads during Emergency Pneumatic Extension on the F-102A, S/N 53-1797.				1798			EFT-P-8-719
Memo EFT-P-8-736 - Determination Contamination in the Cockpit and electronics Bay during MB-1 Rocket Firing				1806			
Memo EFT-P-8-754 - Rocket Launching at Maximum Practical Altitudes			X				Engine Response was normal during two launchings. EFT-A-8-301

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Model F-102A

Program [illegible] Controls

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item 1:							
AN/ARN-14 Omnidirectional Antenna			X				
Test Item 2:							
AN/ARC-34 UHF Communications			X				
Test Item 3:							
Data Link Communications			X				
Test Item 4:							
APX-27 Air to Air IFF			X				
Test Item 5:							
AN/APX-6 Air to Ground IFF			X				
Test Item 6:							
AN/ARN-12 Marker Beacon			X				
Addendum 1:							
AN/APX-6 Antenna System Study (F-102F)			X				
Addendum 2:							
AN/ARN-14 Omnidirectional Antenna System			X				
AN/ARN-18 Glide Path Antenna System			X				

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Airplane S/N 53-1798 Page 1

Model F-102A ZC-8-167

Program 8-10 Controls

CONTROLS TESTS

PHASE I FLIGHT TESTS

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item 1.00: Ground Tests							
1.01 Hysteresis and Breakout Forces			X				10035
1.02 Frequency Response			X				10035
EFT-P-6-294 Manual Mode Open Loop Freq. Resp.			X				
1.03 Elevon System Operation			X				10035
Test Item 2.00: Control System Flight Evaluation							
2.01 Pilot Assist System			X				ZC-8-539
EFT-P-[illegible] Check-Out of P.A. Up-Elevator			X				
EFT-P-[illegible] Check-Out of P.A. Anti-Engage			X				
EFT-P-6-235 Evaluation of Stick Motion in P.A. Mode					X		Testing deemed unnecessary.
2.02 Turn Coordinator			X				ZC-8-539
2.03 Manual Mode Yaw Damper			X				ZC-8-539
2.04 Manual Mode Pitch Damper			X				ZC-8-539

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Airplane S/N 53-1798 Page 2

Model F-102A ZC-8-167

Program 8-10 Controls

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE III FLIGHT TESTS							
Test Item 3.00: System Development							
3.01 Pilot Assist Development					X		To use MAC System
EFT-P-8-116 Increased Surface F.F. Gain							
3.02 Turn Coordinator Development			X				ZC-8-539
3.03 Pitch Damper Development			X				ZC-8-539
EFT-P-8-040 - Summers Rate Gyro Fwd. Location					X		
3.04 Yaw Damper Development			X				ZC-8-539
3.05 Trim Servo Development			X				
EFT-P-8-444A Trim Servo Test			X				
3.06 Components Development			X				ZC-8-539, 531
EFT-P-8-166 Fin Pitot-Static Investigation			X				
EFT-P-8-311 Rate Gyro Environmental Test			X				
EFT-P-8-424 Western Pitch "g" Limiter			X				
EFT-P-8-442 Shelley Unit Tests			X				
EFT-P-8-495 Electronic Pitch "g" Limiter			X				
EFT-P-8-496 H.S.A. Valve Tests			X				
EFT-P-8-582 Aileron Ratio Changer			X				
EFT-P-8-350 AC Rate Gyro Environmental Test				1390			
EFT-P-8-399A Shelley Unit/N.L. Static Source			X				
EFT-P-8-044 Electronic Pitch "g" Limiter - Functional			X				
EFT-P-8-306 Electronic Pitch "g" Limiter - Functional			X				
EFT-P-8-792 Fin Total Pressure Pick-up			X				

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Airplane S/N 53-1798 Page 3

Model F-102A ZC-8-167

Program 8-10 Controls

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE III FLIGHT TESTS - (Continued)							
Test Item 4.00: Stability and Control							
4.01 Dynamic Longitudinal Stability			X				
4.02 Static Longitudinal Stability			X				FFT-A-8-304A
EFT-P-8-696 - Aft CG Tests			X				
4.03 Elevator Control Effectiveness			X				Concurrent
4.04 Maximum Load Factors			X				S/N 53-1794
4.05 Maximum C_L Landing			X				Concurrent
4.06 Take-Off			X				Concurrent
4.07 High Speed Flight			X				
4.08 Longitudinal Trim Change			X				
4.09 Maneuvering Flight Characteristics			X				
4.10 Dynamic Lateral-Directional Stability				1390			
FFT-P-8-629 - Lateral-Directional Oscillations			X				
FFT-P-8-653 - Lateral-Direct. Oscillations - Corrections			X				

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Model F-102A ZC-8-167

Program 8-10 Controls

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE III FLIGHT TESTS - (Continued)							
4.11 Static Directional Stability				1390		2	
4.12 Adverse Yaw				1390		1	
4.13 Rudder Control Force & Effectiveness				1390			
4.14 Aileron Characteristics			X			2	
4.15 Stall Characteristics				1401		1	
4.16 Trimming Devices							Concurrent
Addendum 1 Production Airspeed Calibration			X				EFT-A-217
EFT-P-8-219 Alternate Static-Source Investigation			X				
EFT-P-8-600 Low Speed Calibration			X				
Addendum 2 P-41 Engine Fuel Control Evaluation			X				Concurrent
Addendum 3 O.A.T. Calibration			X				EFT-A-220
Addendum 4 Ram Air Turbine Check-Out			X				ZC-8-532
EFT-P-8-728 - Rat Cylinder Loads			X				
Addendum 5 Drag Chute Deployment Time							Concurrent
Addendum 6 Reduced Rudder Steering Forces							
Addendum 7 Landing Gear Loads Tests			X				

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Airplane S/N 53-1799 Page 1

Model F-102A ZC-8-143

Program 8-10 Weapon System Demonstration

FLIGHT TEST ITEM	Future	Current	Completed	Transforred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
1. PREPARATION & EQUIPMENT CHECKOUT-EAFB							ZC-8-128 WSD Program Prop. EFT-P-8-56 4138-445 CVAC Support Req. by HAC EF-8-817 CVAC-HAC Plan.Conf.
a. EFT-P-8-11 Fire Control System Cooling			X				EFT-A-8-34
b. EFT-P-8-150 Main Landing Gear Side Brace Load Test			X				EFT-A-8-69
c. EFT-P-8-153 Experimental determination of launcher restrictor size			X				
d. EFT-I-8-51 Installation & checkout of test instrumentation EF-8-10-9-1			X				
e. ZC-8-137 Calibration, harmonization & ground checkout of Fire Control System			X				
f. EFT-I-8-43 Installation of strike camera, alignment & operational			X				
g. Missile rail trunnion mislocation tests & correction			X				Missile gear preload check memo 16 Nov. '55 frm R.Costl
h. Shakedown flight A/C, FCS, and T. M. Missiles							HAC Testing
i. Simulated attacks to check complete system operation with missiles			X				" "
j. Air speed calibration & air data computer check (EFT-P-8-278)			X				ZC-8-512-60, 54, 73
k. Miss evaluation - simulated attacks co-altitude			X				HAC Testing
l. EFT-P-8-263 Snap-up performance evaluation			X				EFT-P-8-264, Fig. 1 pg 10 ZC-8-512-57, page 7
m. EFT-P-8-258 Photo panel installation			X				

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Model F-102A

Program 8-10 Weapon System Demonstration

FLIGHT TEST ITEM	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
2. Weapon System Demonstration, HAFB							ZC-512-80
a. Ground System Checkout with missiles			X				EFT-A-8-148 (Memo on ships power troubles)
b. Co-altitude system verification - single missile firings			X				HAC Testing
c. Co-altitude salvo firing - instrumental missiles			X				" "
d. Co-altitude salvo firing (5) tactical missiles			X				" "
e. Snap-up performance flight			X				ZC-8-512-69
f. Snap-up system verification and miss-evaluation			X				" "
g. Snap-up salvo firing			X				" "
ADDENDUM 1							
Production Missile Range Gate Servo Evaluation			X				Data is at HAC
ADDENDUM 2							
W.S.D. with High Speed Target					X		Target not available
ADDENDUM 3							
Missile and Missile Bay Temperature Evaluation					X		Aircraft bailed to HAC

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Model F-102A

Program Armament Rocket

FLIGHT TEST ITEMS	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
A. GROUP 1.00 FLIGHT TESTS - INERT ROCKET FIRINGS							
1.01 Rocket Ejection Tests		X					Six inert rockets have been launched.
B. GROUP 2.00 FLIGHT TESTS -							
2.01 Separation & Trajectory		X				14	
2.02 Blast Effects	X					*	
2.03 A/C Response to Firing	X					*	
2.04 Engine Response to Firing	X					*	
2.05 Exhaust Dust Effects	X					*	
C. GROUP 3.00 FLIGHT TESTS - ROCKET FIRING WITH XMG-14							
3.01 XMG-14 Checkout					X		
3.02 XMG-14 Evaluation					X		
D. Group 3.1 Snap up	X					14	
EFT-P-9-428 Missile-Bay Door Flutter Investigation			X			6	Completed 5/1/56
EFT-P-9-548 - Ram Air Turbine Cylinder Rod Loads				1798			EFT-P-753

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Airplane S/N 53-1808 Page 1

Model 8-10

Program ZC-3-286

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
ARDC Advanced Instrument Display			X				

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Model F-102A ZC-8-171

Program 8-10 Big Tail (Gloved)

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
Test Item No. 1.00: Flutter							
1.01 Flight Flutter Investigation			X				ZU-8-034
EFT-P-8-195 Flutter Investigations			X				
EFT-P-8-250 Flutter Investigations			X				
Test Item No. 2.00: Stability & Control							
2.01 Static Directional Stability			X				EFT-A-8-81, ZC-8-512-45
2.02 Dynamic Lateral-Directional Stability			X				EFT-A-8-59, ZC-8-512-43
2.03 Roll Coupling			X				ZU-8-055, ZC-8-512-56
EFT-P-8-217 Beta Feedback Tests					X		B Feedback will not be installed
EFT-P-8-233 Additional Stability & Control Test Points			X				
EFT-P-8-266 Lateral Response to Rolling Maneuvers			X				EFT-A-8-81
EFT-P-8-421 Additional Roll Tests			X				EFT-A-8-118
ADDENDUM 1 Landing Light Evaluation			X				ZC-8-512-91, 92
Additional Testing							
EFT-P-8-349 Sundstrand const. speed drive				1380			
EFT-P-8-353 Lat. Control System 20# @5°			X				
EFT-P-8-364 Linear Thrust Fuel Control				1791			
EFT-P-8-356 MLG operating time & fairing link loads				1791			
EFT-P-8-361 Elevon camera fairing			X				Flight Status Report, Flt
EFT-P-8-361 Lateral control system, addition of 2.5 inches free play					X		

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Airplane S/N 53-1813 Page 1

Model F-102A ZC-8-197

Program Side Stick

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd	REMARKS, REF. MEMOS & REPORTS
1.0 Ground Tests			X				
2.0 Flight Tests							
2.1 Taxi Tests			X				
2.2 Elevator Control Forces			X				
2.3 Elevator Control Gradient			X				
2.4 Longitudinal Trimming Devices			X				
2.5 Aileron Control Force			X				
2.6 Aileron Control Effectiveness			X				
2.7 Lateral Trimming Devices			X				

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Airplane S/N 54-1351 Page 1

Model TF-102A ZC-8-140 Rev. 5

Program Phase III

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
Test Item 1.00: Ground Tests							
1.01 Engine Operation			X				ZC-8-512-37, 49
1.02 Engine Equipment Operation			X				Qualitative Tests Only EFT-A-8-213
1.03 Static Thrust Calibration			X				
1.04 Low Speed Taxi Tests			X				" " "
1.05 High Speed Taxi Tests and Lift-Off Characteristics			X				" " "
Test Item 2.00: Airplane System Functional Tests							
2.01 Functional Flight Test			X				
EFT-P-8-114 Ram Air Press. & Temp. Test at Reg. 148							EFT-A-8-189
EFT-P-8-271 Irreverisible Power Control							
EFT-P-8-426 Emergency Landing Gear Extension Time							EFT-A-8-141, ZC-8-512-61 ZC-8-512-62
2.02 Flight Control System Operation			X				
EFT-P-8-402 Improvement of Turn Coordinator and Damper System Operation			X				ZC-8-512-58, ZC-8-512-60

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Model TF-102A

Program Phase III

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item 2.00: Airplane System Functional Tests - (Continued)							
2.03 Emergency Fuel Control			X				ZC-8-512-36
2.04 Power Plant Operation			X				ZC-8-512-49, 64 EFT-A-8-11
2.05 Engine & Structural Cooling			X				Qualitative Only ZC-8-512-36, ZC-8-512-60
2.06 Brake Parachute Operation			X				
2.07 Control Sensitivity Check			X				ZC-8-512-36, ZC-8-512-60
Test Item 3.00: Calibrations							
3.01 Airspeed Calibration			X				
EFT-P-8-181 Airspeed Calibration			X				ZC-8-512-60, 40 EFT-A-8-203 54-55
Test Item 4.00: Airplane Performance							
4.01 Climb			X				ZC-8-512-55 EFT-A-8-140 ZC-8-512-62

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Airplane S/N 54-1351 Page 3

Model TF-102A

Program Phase III

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item 4.00 Airplane Performance - (Continued)							
4.02 Level Flight							
EFT-P-8-328 Performance Tests - Level Flight Performance			X				ZC-8-512-41, 43, 44, 45, 48, 50, 51, 53,
Test Item 5.00 Preliminary Flight Characteristics							EFT-A-8-164; ZC-44, 67 EFT-A-8-1[illegible]
5.01 Damper System Check			X				EFT-A-8-166, ZC-8-512-60
5.02 Roll Coupling			X				EFT-A-8-260
EFT-P-8-459 Positive Quadrant Roll Coupling Tests			X				
5.03 Buffet Investigation							
EFT-P-8-113 Investigation of Buffet		X	X				ZC-8-512-417, ZC-8-512-63,
EFT-P-8-113A " " "			X				
EFT-P-8-113B " " "			X				
EFT-P-8-113C " " "			X				
EFT-P-8-113D " " "					X		
EFT-P-8-113E " " "					X		
EFT-P-8-113F " " "			X				EFT-A-8-98
EFT-P-8-113G " " "			X				EFT-A-8-~~107~~ EFT-A-8-173 -178
EFT-P-8-113H " " "			X				
EFT-P-8-113I " " "					X		
EFT-P-8-113J Optimization Vortex Generators			X				
EFT-P-8-113K " " "			X				

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Airplane S/N 54-1351 Page 4

Model TF-102A

Program Phase I, II, III

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE III FLIGHT TESTS							
Test Item 6.00 All Weather Equipment							
6.01 Windshield Anti-Icing System EFT-P-8-939B & 893			X				EFT-A-8-420A
6.02 Rain Clearing System EFT-P-969			X				EFT-A-8-420A, ZC-8-512-109
6.03 Radome Anti-Icing		X					EFT-A-8-449, EFT-A-8-342
Test Item 7.00 Hydraulic System							
7.01 Hydraulic System Flight Tests		X					EFT-A-8-424
EFT-P-8-845 Instrument Panel Vibration Test			X				EFT-A-8-415
ZC-8-140 Addendum 2 1 November 1956 (added)	X						
Angle of Attack Calibration			X				

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Airplane S/N 54-1353 Page 1

Model TF-102A ZC-8-180, Rev. 1

Program Roll Coupling

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
PHASE I FLIGHT TESTS							
Test Item No. 1.00: Preliminary Evaluation							
1.01 Qualitative Evaluation of Buffet			X				Qualitative data only ZC-8-512-63, EFT-A-8-417, EFT-A-8-178
Test Item No. 2.00: Quantitative Evaluation							
2.01 Cut Down Canopy			X				ZC-8-512-63, EFT-A-8-417
2.02 Cut Down Canopy plus Minor Modifications			X				ZC-8-512-63, EFT-A-8-417
Landing Gear Extension Time Tests, EFT-P-8-426			X				EFT-A-8-141, ZC-8-517-61
Test Item No. 3.00 - Stability & Control							
3.01 Static & Dynamic Directional Stability			X				
3.02 Roll Coupling EFT-P-931 P-948 P-924			X				EFT-A-8-282
EFT-P-8-855 Irreversible Power Control			X				
EFT-P-8-874 Engine Trim with inlet Duct Screens Installed on Aircraft 1380 & 1252							AVO to Controls 6 RP Trans to 54-1380
Performance Max. speed dive - ZC-8-512-91							

EFT-P-8-583

CONVAIR - San Diego

Airplane S/N 54-1354 Page 1
Model TF-102A (8-12 #4) Prod. #46
Program ZC-8-141

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Items							
1.01 Strike Camera Boresighting			X				
1.02 R.W.T. Boresighting per SF-0972				1351			
2.01 Armament Hardware Checkout							ZC-8-512-83, 79
(1) Rockets			X				
(2) Missiles			X				
3.01 Separation and Trajectory							
(1) Rockets		X					EFT-P-8-952, EFT-A-8-399, ZC-8-512-111
(2) Missiles			X				EFT-P-8-834
3.02 A/C Response to Armament Operation			X				
3.03 Engine Response to Rocket and Missile Exhaust			X				EFT-P-8-773, ZC-8-512-82
Phase III Development Tests							
4.01 Inlet Duct Anti-Icing System-Spec Compliance			X			2	Ref. A.F. Spec 40395-C
Aircraft Weapon Control System Tests							Addendum 1 to ZC-8-141
5.00 FCS Vibration Environment					X		EFT-P-8-1009
6.00 AWCS Response to Firing	X					10	

EFT-P-8-583

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Airplane S/N 54-1361 Page 1

Model TF-102A 8-12

Program Structural Demonstration

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Flight Test Item 1.00 Structural Demonstration							
1.01 Wing Symmetrical Loads Demonstration			X				
1.02 Bank to Bank Rolls			X				
1.03 Level Flight Rolls 360°			X				
1.04 Rudder Kicks		X				32	Completion Expected 8/9/57
1.05 Zero Load Factor Rolls			X				
1.06 360° Rolls		X					
Flight Test Item 2.00 Cabin Pressurization	X						
Flight Test Item 3.00 Heating and Ventilating							
3.01 Ground Cooling of Electronic Bays	X						
3.02 In Flight Heating and Ventilating	X						
Flight Test Item 4.00 External Tank Tests				4051			
Air Force Phase IV	X						

EFT-P-8-583

CONVAIR - San Diego

Airplane S/N 54-1362 Page 1

Model TF-102A 8-12

Program Stability & Control, Spin Testing

		Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd:	REMARKS, REF. MEMOS & REPORTS
Flight Test Item 1.00	**Stability and Control**	X						
1.01	Directional Stability and Yaw Damper	X						
	Optimisation	X						
1.02	Dynamic Longitudinal Stability and Pitch	X						
	Damper Optimization	X						
1.03	Static Longitudinal	X						
1.04	Turn Coordinator Optimisation	X						
1.05	Elevator Control Forces and Effectiveness	X						
1.06	Maximum C_L Landing	X						
1.07	Take-off	X						
1.08	Rudder Control Forces and Effectiveness	X						
1.09	Adverse Yaw	X						
1.10	Trimming Devices	X						
1.11	Longitudinal Trim Changes	X						
1.12	Aileron Characteristics	X						
1.13	Maneuvering Flight	X						
1.14	Stall Characteristics	X						
1.15	High Speed Flight Requirement	X						
Flight Test Item 2.00	**Spin Tests**			X				
2.01	Special Equipment Checkout			X				
2.02	Preliminary Stall and Spin Investigation			X				
2.03	Specification Demonstration			X				
Flight Test Item 3.00	**Aircraft Calibration**					X		
3.01	Production Airspeed Calibration and Alternate Source Location					X		

CONVAIR - San Diego

Airplane S/N 54-1362 Page 2

Model TF-102A 8-12

Program Stability & Control, Spin Testing

		Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Flight Test Item 4.00	Specification Compliance & Development					X		
4.01	Cockpit Noise Level					X		
4.02	Carbon Monoxide Detection (Cockpit)					X		
4.03	Cockpit Watertightness					X		
Flight Test Item 5.00	Developmental Testing (To be Determined)					X		

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Airplane S/N 56-4051 Page 1

Model TF-102A

Program RFT-F-3-3038

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
External Tank Evaluation Tests			X				

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Airplane S/N 56-1138 Page 1

Model F-102A

Program ZC-8-288

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Flight Evaluation of ARN-21 and ARN-31 "TACAN"			X				
Speed Brake Structural Demonstration Test			X				

CONVAIR - San Diego

Airplane S/N 54-1317 Page 1

Model 8-10

Program ZC-8-375 Revision A

		Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Flight Test Item 2.00	Angle of Attack Calibration							
3.00	Stability and Control							
3.01	Dynamic and Static Directional Stability							
4.00	Structural Demonstration General Instructions							
4.01	Pull-Up - Push Overs							
4.02	Roll Maneuvers							
4.03	Rudder Kicks							

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Airplane S/N 54-1380 Page 1
Model F-102A ZC-8-240
Program Power Plant Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item 1.00 Ground Tests							
1.01 Engine Operation							EFT-P-8-874
1.02 Engine Equipment Operation							
1.03 Static Thrust Calibration							
1.04 Low Speed Taxi Tests							
Test Item 2.00 Aircraft Systems Functional Tests							
EFT-P-8-699 Determination of Div. Brake Effectiveness					X		EFT-P-8-735
EFT-P-8-730 Large Area Speed Brake Closing Times				1390			
2.01 Functional Test Flights			X				
2.02 Emergency Fuel Control System			X				(EFT-A-8-340, -366, ZC-8-512, (-94, -100 ZC-8-512-106
2.03 Engine and Structural Cooling			X				
2.04 Removal of Engine Diffuser Heat Shield			X				
Test Item 3.00 Fuel System							
EFT-P-8-799 Ground Fuel Heating and Engine Performance with Heated Fuel							Cancelled by AVO
3.01 Flow Joiners (Hydra Electric, others)			X				EFT-A-8-392
3.02 Twin Boost Pump			X				EFT-A-8-411
3.03 Modified Engine Fuel Control			X				EFT-A-8-270
EFT-P-8-576 Linear Thrust Fuel Control					X		EFT-P-8-735
3.04 Fuel Tank Negative "g" Vent Valve			X				
Test Item 4.00 Anti-Surge Bleed Valve Optimization			X				EFT-A-8-194-243-298 ZC-8-512-69-77
EFT-P-8-566 Anti-Surge Bleed Valve Optimization					X		EFT-P-8-735
Test Item 5.00 Afterburner							
EFT-P- -,76 A/B Shroud Surface Temperatures			X				EFT-A-8-229,
5.01 J-57-P-43 Engine Afterburner Flame Out			X				EFT-A-8-341, 277, ZC-8-,72-81,
5.02 A/B Igniter Valve Continuous Fuel Flow			X				-93, -101

CONVAIR - San Diego

Airplane S/N 54-1380 Page 2

Model F-102A

Program Power Plant Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item 6.00 Exhaust Nozzle							
6.01 Nozzle Actuating Temperatures			X				EFT-A-8-253
Test Item 7.00 Sundstrand Unit							
7.01 Negative "g" Check			X				ZC-8-512-83, -85,EFT-A-8-333 EFT-A-8-346
7.02 Development Changes					X		
7.03 Checkout Revised Engine Oil Breather Valve Setting			X				
7.04 Independent Oil System					X		
Test Item 8.00 General Airframe Improvements			X				(EFT-A-8-193, ZC-8-512-68-84
EFT-P-8-613A Landing Gear Loads Test							(EFT-A-8-362)
EFT-P-8-2024 Emergency Landing Gear Extension Tests							ZC-8-512-97 EFT-A-8-419
8.01 N_1 Bleed Ducts			X				
8.02 Plastic Ducts			X				
8.03 Relocated Oil Breather Outlet			X				
8.04 External Fuel Tanks - Development Tests					X		
Test Item 9.00 Combustion Starter Changes			X				ZC-8-512-97, EFT-A-8-397
Test Item 10.00 Fire Detector			X				
10.01 Evaluation of Existing System			X				EFT-A-8-367, ZC-8-512-100
10.02 Evaluation of Walter Kidde Detector					X		

CONVAIR - San Diego

Airplane S/N 54-1380 Page 3

Model F-102A ZC-8-240

Program Power Plant Development

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item 11.00 Cabin Pressurization and Ventilating System							
EFT-P-8-567A Cabin Pressure Surge Testing				X			ZC-8-512-95
11.01 Specification Conformance (Dive and Climb)				X			
11.02 Carbon Monoxide Testing			X				EFT-A-8-380,ZC-8-512-104
11.03 Hamilton Standard System			X				
Test Item 12.00 Engine and Afterburner Re-light Envelope			X				EFT-A-8-235 [illegible] & 341 ZC-8-512-72-81-8, 93,101, 106
Test Item 13.00 Engine Development Changes					X		
Test Item 14.00 Fuselage Skin Drain Adequacy			X				

EFT-P-8-583

CONVAIR - San Diego

Airplane S/N 54-1388 Page 1
Model F-102A ZC-8-183 Revision I
Program Phase III Systems Tests

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Group I							
1.00 Engine Inlet Scoop Anti-Icing							
1.01 Engine Inlet Scoop Anti-Icing, Dry Air Tests			X				Flight Test Complete, Data in Work, EFT-A-8-402
EFT-P-8-579 Ground Checkout of T/M and Engine Inlet Scoop Anti-Ice Instrumentation			X				ZC-8-512-78
EFT-P-8-611 Ground Check of Air Flow Distribution to the Engine Inlet Scoop Anti-Ice System Computation Procedure for Duct Anti-Icing Test			X				EFT-A-8-175
Group II							
2.00 Low Pressure Pneumatics							
2.01 Air Conditioning and Ventilation			X				
EFT-P-8-656 Hamilton Standard Light Wt Refrigeration Package					X		
EFT-P-8-819A3 Response of Cabin Temp Control			X				ZC-8-512-115
EFT-P-8-878 MG-3 Operation During Heat & Vent Tests Computation Procedure for Heat & Vent Tests			X				EFT-A-8-291
EFT-P-8-895 Aft Electronic Area Cooling Evaluation		X					In flight complete, ground to be done, EFT-A-8-400
EFT-P-8-910 Stratos Refrigeration Unit Installed					X		
EFT-P-8-1031A Stratos Environmental Tests			X				
EFT-P-8-2079 Hamilton Standard Air Conditioning Unit and Canopy Defog Systems		X					
2.02 Hot Day Ground Support Tests			X				
EFT-P-8-796 Hot Day Ground Support Test			X				Testing Complete, Data in Work EFT-A-8-356, EFT-A-8-255, ZC-8-512-87
Group III							
3.00 Anti-Icing & Rain Clearing							
3.01 Windshield & Canopy Defogging & Anti-Icing			X				
EFT-P-8-974 Windshield Light Baffle Installation			X				
EFT-P-8-1056 W/S Defog-Hot Air and Extended Electrical coating cone					X		
3.02 Windshield Rain Clearing			X				
			X				EFT-A-8-427

CONVAIR - San Diego

Airplane S/N 54-1388 Page 2

Model F-102A ZC-8-183 Revision 1

Program Phase III Systems Tests

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Group IV							
4.00 MG-10 Vibration Environment	X						
4.01 Vibration Survey							
EFT-P-8-827 Instrument Panel Vibration			X				
EFT-P-8-970 Configuration Requirements for MG-10 Vibration Environment Tests	X						

EFT-P-8-583

CONVAIR - San Diego

Airplane S/N 54-1390 Page 1

Model F-102A

Program ZC-8-195

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item No. 1.00 Flutter							
1.01 Flight Flutter			X				EFT-A-8-254 ZC-8-512-78(-92)
EFT-P-8-926 Flight Flutter, Low Altitude - High Dynamic Pressure Test Points			X				EFT-A-8-313
Test Item No. 2.00 Stability & Control							
2.01 Directional Stability & Yaw Damper Optimization			X				ZC-8-512-87
2.02 Turn Coordinator Optimization			X				
2.03 Adverse Yaw			X				
2.04 Rudder Control Force & Effectiveness			X				
EFT-P-8-737 Effect of Large Speed Brakes Uppon Rudder Effectiveness			X				
2.05 Aileron Characteristics			X				
EFT-P-8-741 and -741A Aileron Roll Tests, With and Without Stability Augmentation			X				EFT-A-8-353 to AF
EFT-P-8-946 Aileron Roll Tests Without Stability Augmentation - Add. Test Points			X				
EFT-P-8-811 Investigation of Roll off During High Speed Pull Outs			X				EFT-A-8-278
EFT-P-8-953 Determination of Hinge Moments and Investigation of Pull Outs			X				ZC-8-512-92-95-96
EFT-P-8-860 Evaluation of Shelley Unit and Determination of Trim Requirements			X				ZC-8-512-104

CONVAIR - San Diego

Airplane S/N 54-1390 Page 2

Model F-102A

Program ZC-8-195

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Test Item 3.00 Performance							
3.01 Level Flight Performance					X		
3.02 Take-off & Landing Performance			X				
3.03 Speed Brake Effectiveness			X				
Miscellaneous EFT-P-8-810 Investigation of Air Oil Cooler Duct Failure			X				EFT-A-8-283
EFT-P-8-937 Evaluation of Skewed Main Landing Gear			X				EFT-A-8-361 (ZC-8-512-96, -97, -100, -98)
EFT-P-8-790A Large Speed Brake Closing Time			X				EFT Log & status report (flt 9) 9-15-56
EFT-P-8-938A Drag Parachute Deployment Time With Large Speed Brakes			X				EFT-A-8-358 & EFT-A-8-377 ZC-8-512-101-104, -102
EFT-P-8-892 Drag Parachute Riser Service Test			X				ZC-8-512-102
Test Item 4.00 Tactical Improvements	X						
Prod. 354 Config. Wring Out		X					ZC-8-195 Supplement 1
Lateral Oscillations in Pilot Assist		X					[redacted]

CONVAIR - San Diego

Airplane S/N 54-1398 Page 1

Model F-102A

Program ZC-8-243

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
Asymmetrical Maneuvers							
Rolls							
Bank to Bank		X				28	
180°		X					
360°		X					
Rudder-Slips		X					
Stability Derivatives					X		
High Drag							
Ram Air Turbine (EFT-P-3060)			X			1	
Rolling Pull Out (7° Aileron With External Fuel Tanks			X			2	
Longitudinal Control Stability (With tanks) (EFT-P-8-3060)					X		
Landing Gear							
Structural Demonstration (Airloads) (EFT-P-8-3060)			X			8	

EFT-P-8-583

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Airplane S/N 54-1401 Page 1

Model F-102A

Program Stall, Spin and Hydraulic Systems Tests

	Future	Current	Completed	Transferred To:	Cancelled	Est. Flts. Req'd.	REMARKS, REF. MEMOS & REPORTS
1.00 Stalls & Spins							
1.01 Special Test Equipment Checkout (Revision 1)			X				
1.02 Preliminary Investigation			X				
1.03 Specification Demonstration, Spins							
1.04 Specification Demonstration, Stalls							
2.00 Hydraulic System Tests							
2.01 Functional Demonstration							
2.02 Partial Failure Simulation, Primary System							
2.03 Partial Failure Simulation, Secondary System				1399			
ZC-8-256 Addendum 1							
1.00 [illegible] Development Tests	X					25	
2.00 In-Flight Electrical Loads Tests	X					2	

T-9 FLUTTER TESTS
1954-1955-1956

2743

Fefferman
Due in S.D. Mon 11/19/57

[redacted]

HEADQUARTERS
AIR MATERIEL COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

1697

UNCLASSIFIED

Classification F102 P.O File No T-9
UNCLASSIFIED OCT 3 '56
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by:

SUBJECT: Contract AF 33(600)-[illegible]942 F-102A Flutter Status
Dept 130-1 Date
R. G. Avritt

THRU: Air Force Plant Representative

TO: Convair
Division of General Dynamics Corporation
P. O. Box 1950
3165 Pacific Highway
San Diego 12, California

REF: (a) Convair Report ZC-8-195, dated 2 July 1956
(b) AFPR, Convair, San Diego, TWX SARSP-9-55-E, dated 5 September 1956

TO SPEED DELIVERY OF YOUR REPLY

	MODEL PROJECT OFFICE	INFO	COMMENT	ACTION	REPLY	INITIALS AND DATE
	Adamson					
	Bokemeyer					
	Benson					
	Carlson					
1 – R	KULL					10/24
	Ferguson					
4 – R	Fox					
	Mohr/Wells					
2 – R	~~Ottem~~					
	Strayer					
3 – R	~~White~~					
	Worden					
C	Dublin				X	

REMARKS:

1. Reference (a) proposes, among other items, a flutter program for the F-102A production large tail. This program is acceptable to this Command except that no flights are scheduled for low altitudes where the critical transonic region will be traversed at high dynamic pressures.

2. Therefore, it is requested that Convair substantiate the complete flutter safety of the production large tail by conducting flight flutter tests at 5000-8000 foot altitude at speeds between M=0.9 and the aircraft's limit speed at that altitude.

3. Reference (b) notified this Command of the lifting of flight restrictions on F-102A, S/N 53-1813, above the limit of M=1.2 or 500 KIAS, whichever is less, which had previously been approved. Since it is a function of this Command to approve flight limitations, it is requested that Convair either submit flight data to show that the aircraft is free from flutter up to the restriction mentioned in reference (b), or prove by ground vibration test data that the production large tail flight test

Fefferman in Dayton
on this subject

UNCLASSIFIED

CONVAIR SECURITY LOG
DEPT ___ LOG NO 5567 ___ COPY 11
56 SARS 18520
SEP 1 3 1956

[redacted]

AMC Form 9, 25 Aug 52

"MORE AIR FORCE PER DOLLAR"

OCT 1 7 1956

CONFIDENTIAL

AMC Ltr MC[illegible]([illegible]/1-12A)JEH/ewt, dtd [illegible] 1952 to Convair, San Diego, Calif, Subj: Contract AF 33(600)-5942, F-102A Flight Flutter Studies

data are applicable to the gloved tail aircraft. This data should include the information requested in paragraph 2 above, and should be submitted as expeditiously as possible in order to obtain approval and/or recommendations pertaining to the flutter safety of the aircraft.

FOR THE COMMANDER:

WM [illegible] Major USAF
Chief [illegible]
Fighter Branch

[illegible]

Copy # 1
①
W. W. Shaf/B. ~~Kraft~~ JK.

Flt test now trying to set a program to do this together. Awk

CONVAIR
A Division of General Dynamics Corporation
San Diego

DF-8-208

F 102 PO File No T-9

26 September 1956

RECEIVED
OCT - 1 1956
F-102
PROJECT
OFFICE

FURTHER FLIGHT FLUTTER TEST REQUIREMENTS ON PRODUCTION BIG TAIL F-102A AIRCRAFT

INTRODUCTION

Flight flutter testing has been accomplished on Flight Test Operations 1-8 on S/N 54-1390 (see Reference 1) which lifts all speed restrictions on production big tail aircraft to the design limit dive speed of the aircraft. This testing was accomplished in the region between 20,000 and 35,000 feet and no flight flutter testing was done at lower altitudes. Personnel of the Dynamics Branch of WADC have indicated that the entire altitude-speed envelope should be flown during flight flutter testing.

Thus, in anticipation of a WADC requirement that flight flutter test speeds be flown at lower altitudes, it is proposed that a program be set up accomplishing this testing at the earliest possible date.

TEST PROGRAM

In order that this program may be flown with the least possible effort, it is proposed that F-102A S/N 54-1390 be used. This airplane already has the proper instrumentation installed and could be used as a continuation of the flight flutter program at the higher altitudes.

Since flight flutter test points have been flown at the maximum "q" condition, it remains that the effects of density only be checked. It is therefore proposed that an aileron pulse in each direction, left and right, be made at the following points:

1. V_c = 655 knots @ 15,000 ft.
2. V_c = 655 knots @ 10,000 ft.
3. V_c = 655 knots @ 7,500 ft.

If a flight over the Salton Sea can be arranged, a fourth point should be flown:

4. V_c = 655 knots @ 5,000 ft.

It is anticipated that the above speed-altitude points are all that will be required; however, if any adverse aerodynamic damping is encountered, it may be necessary to take smaller speed increments.

INSTRUMENTATION

The instrumentation to be used is the same already installed in S/N 54-1390 and called out in Reference (2). The responses of the velocity pickups are to be telemetered to the ground station, recorded on magnetic tape and an oscillograph, and monitored on a Sanborn recorder.

Prepared by RC Peller
R. C. Peller

Checked by R L Fefferman
R. L. Fefferman

Approved by M Dublin
M. Dublin

cc: W. W. Fox/C. Kull
P. Prophett/W. Withee
T. Maloy/H. Schuette
J. Beckman/W. Walker
M. Edelstein
R. L. Fefferman/R. C. Peller
M. Dublin
Dyn. Files

REFERENCES

1. Convair Report ZU-8-034, Add. A, "Results of Flight Flutter Testing of Large Tail F-102A Aircraft"

2. Convair Memorandum DF-8-189, "F-102 Flight Flutter Test Program on Large Tail Aircraft," dated 27 April 1956

Copy # 1

[redacted]

Strayer

Kellogg AWK

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SEP 11 1956

F-102
PROJECT
OFFICE

C O N V A I R
A Division of General Dynamics Corporation
San Diego

DF-8-197 7 September 1956
Addendum III

F 102 PO File No T-9

OUTLINE OF F-102A S/N 53-1791 FLIGHT FLUTTER TESTS FOR CONFIGURATION WITH EXTERNAL WING TANKS

The purpose of this memorandum is to revise and replace all former programs as given in the main body of this memorandum and Addendum I and II to it. The tests shall be carried out as indicated in Table A-III. In all cases speeds are not to exceed 400 knots indicated or .95 M_{true} whichever is the lower.

Prepared by A. D. Mac Lellan
A. D. Mac Lellan

Checked by R L Fefferman
R. L. Fefferman

Approved by M Dublin
M. Dublin

cc: W.W. Fox
P. Prophett/H. Cheney
T. Maloy
D. Hull
R. L. Johnson/V. Allwardt
M. Brady/R. Sassaman
R. L. Fefferman/R. C. Peller
W. D. Daniels
M. Dublin/L. Payne
F. Haas/Dyn. Files

DF-8-197
Addendum III

TABLE A-III

SCHEDULE OF FLIGHT FLUTTER TESTS ON F-102A AIRPLANE WITH EXTERNAL FUEL TANKS INSTALLED

Data Point	Speed*	Altitude (Ft)	External Configuration	Test Objective
1	.75M	26,000	Full	Sweep Frequency Records
2	.85M	26,000	Full	Decay Records
3	.95M	26,000	Full	Decay Records
4	.75M	26,000	Half-full; fwd. c.g.	Sweep Frequency Records
5	.85M	26,000	Half-full; fwd. c.g.	Decay Records
6	.95M	26,000	Half-full; fwd. c.g.	Decay Records
7	.75M	26,000	Half-full; aft. c.g.	Sweep Frequency Records
8	.85M	26,000	Half-full; aft. c.g.	Decay Records
9	.95M	26,000	Half-full; aft. c.g.	Decay Records
10	400 kn	15,000	Half-full; aft. c.g.	Decay Records
11	400 kn	15,000	Full	Decay Records
12	400 kn	15,000	Half-full; fwd. c.g.	Decay Records
13	400 kn	7,500	Half-full; fwd. c.g.	Decay Records
14	400 kn	7,500	Full	Decay Records
15	400 kn	7,500	Half-full; aft. c.g.	Decay Records
16	400 kn	h**	Half-full; aft. c.g.	Decay Records
17	400 kn	h**	Full	Decay Records
18	400 kn	h**	Half-full; fwd. c.g.	Decay Records

* M indicates true Mach number
kn Means indicated air speed in knots
** Minimum density altitude at which airplane may be safely flown to 400 knots indicated.

CONVAIR - SD

MODEL 8 PROJECT OFFICE MEMO T-9 28 August 1956

TO: H. R. Francon - H. R. Kennedy - D. R. Stewart Page 1 of 1

SUBJECT: Flight Flutter Tests - Completion of
F-102A Large Tail Airplanes

REFERENCE: (a) Model 8 Project Office Memo dated 24 July 56, subj:
"Flight Limitations, Airplanes S/N 54-1388, 1390, 1398."

(b) Model 8 Project Office Memo dated 25 July 56, subj:
"Flight Limitations, Airplane S/N 54-1401."

(c) Model 8 Project Office Memo dated 24 July 56, subj:
"Flight Limitations, Airplanes S/N 55-3357 thru 55-3379."

(d) Model 8 Project Office Memo dated 24 July 56, subj:
"Flight Limitations, Airplanes S/N 55-3380 thru 55-3426."

References (a) thru (d) restrict those airplanes (excluding 54-1388, small tail) to a maximum speed of Mach = 1.25 or 600 knots I.A.S., whichever occurs first, pending completion of flight flutter tests on the large tail configuration.

Results of flight flutter tests on F-102A aircraft, S/N 54-1390 production large tail airplane, show adequate flutter stability at speeds equal to or exceeding the limit dive speed of the airplane.

By issue of this memo, the flutter limitation for large tail airplanes is recinded.

for W. W. FOX
Senior Project Engineer

WWF:CJK:jh

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CONVAIR
A Division of General Dynamics Corporation
San Diego 12, California

Letter of transmittal
to be declassified when
documents are detached.

In reply refer to:
WWF:JAB:jj
Ltr. #6-4779
Log #3-7580

F 102 PO File No 7-9

18 July 1956

Subject: Contract AF33(600)-5942
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model F-102A Airplanes
Flight Flutter, Stability and Control and
Tactical Improvement Tests - Submittal of

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

Attention: MCPHFC (RDZSFG)

Via: Air Force Plant Representative
San Antonio Air Materiel Area
Convair, A Division of General Dynamics Corporation
San Diego 12, California

Reference: (a) Supplemental Agreement No. 88 to Subject Contract

Enclosure: (A) Five (5) copies Confidential Report ZC-8-195, dated 2 July 1956 - Flight Flutter, Stability and Control and Tactical Improvement Tests F-102 S/N 54-1390

(B) Thirteen (13) copies Convair Confidential DD Form 250 (modified) P.S. No. 649289, dated 18 July 1956

1. Enclosure (A) is submitted in compliance with requirements of Reference (a).

CONVAIR
A Division of General Dynamics Corporation
San Diego 12, California

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

Ltr. #6-4779
Log #3-7580
18 July 1956

2. It is requested that Contractor be notified of acceptance of the enclosed data by returning an executed copy of Enclosure (B).

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

W. W. Fox
Senior Project Engineer

cc: AFPR (4) W/4 Encl. (A),(B)
LADFO
USAF (13) W/13 Encl. (B)
Dayton Office (2)
Contracts
J. J. Alkazin (3)
L. Moore (2)
Engr. Files
Model 8 P/O (3)
Engr. Flight Test Plan.
Acct/Bill W/Encl. (B)
Spec. Files (2)

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documents are detached.

~~Strange~~ PJO
~~Jattelin~~
~~White~~
~~Fay~~ [illegible]

RECEIVED
JUN 13 1956
F-102 PROJECT OFFICE

C O N V A I R
A Division of General Dynamics Corporation
San Diego

DF-8-167
Addendum II

7 June 1956

OUTLINE OF F-102A S/N 53-1791 FLIGHT FLUTTER TESTS FOR CONFIGURATION WITH EXTERNAL WING TANKS

F 102 PO File No F-9

Addendum I of this memorandum presents the outline of flight flutter tests to be performed on F-102A S/N 53-1791 for the configuration with external wing tanks. Since issuance of this memorandum, Reference (1) has been received in which paragraph 3 e. requires that flight flutter testing be accomplished at the minimum altitude at which M = 0.95 can be obtained in level flight. These tests are to be performed in the three external tank fuel conditions and in two equal speed increments beginning at a speed corresponding to a compressible dynamic pressure of 950 lbs/ft^2 and terminating at M = 0.95. These tests are to be run subsequent to flight flutter testing at 15,000 ft.

In order that an adequate margin of safety be allowed, it is proposed that these tests be accomplished at an altitude of 7500 ft. which corresponds to the minimum altitude at which the flight loads are to be determined. Table I - II of this addendum presents the airspeeds, Mach numbers, and dynamic pressures for the test points to be run. This table is to be considered a continuation of Table I - I of Addendum I.

Prepared by R C Peller
R. C. Peller

Checked by R L Fefferman
R. L. Fefferman

Approved by M Dublin
M. Dublin

cc: ✓W. W. Fox
P. Prophett/H. Cheney
T. Maloy
D. Hull
R. L. Johnson/V. Allwardt
M. Brady/R. Sassman
R. L. Fefferman/R. C. Peller
W. D. Daniels
M. Dublin
Dyn. Files

Reference:

1. Air Force letter MCPHFC(RDZSFG) 5-1-3a, WJM/jem, dtd. Mar. 13, 1956, Convair Log No. 6A-4112.

CONVAIR
A Division of General Dynamics Corporation
San Diego

DF-8-167
Addendum II

7 June 1956

TABLE I - II

SCHEDULE OF FLIGHT FLUTTER TEST WITH EXTERNAL WING TANKS

Run No.	M	h	V_t	q_c	Remarks
10	0.84	7500	541	950	Full Tanks Sweep records
11	0.895	7500	576	1094	Decay records
12	0.95	7500	611	1262	Decay records
13	0.84	7500	541	950	Half Full Tanks - Fwd C. G. Sweep records
14	0.895	7500	576	1094	Decay records
15	0.95	7500	611	1262	Decay records
16	0.84	7500	541	950	Half Full Tanks - Aft C.G. Sweep records
17	0.895	7500	576	1094	Decay records
18	0.95	7500	611	1262	Decay records

Legend:

M = True Mach number

h = Pressure altitude (feet)

V_t = True air speed (knots, standard day)

q_c = Compressible dynamic pressure (lbs/ft^2, standard day)

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A Division of General Dynamics Corporation
San Diego 12, California

Letter of transmittal
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F 102 PO File No 7-9

In reply refer to:
[illegible]

6 June 1956

Subject: Contract AF33(600)-5942, -23903, [illegible]
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model F-102A and TF-102A Airplanes
Flight Flutter Tests - Submittal of

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

Attention: MCPEC (MCPEFG)

Via: Air Force Plant Representative
San Antonio Air Materiel Area
Convair, A Division of General Dynamics Corporation
San Diego 12, California

Reference: (a) Convair Specification ZU-8-025

Enclosures: (A) Three (3) copies Convair Confidential Report ZU-8-034, Results of F-102A Airplane Flight Flutter Tests, dated 10 May 1956

(B) Thirteen (13) copies Convair Confidential DD Form 250 (modified) P.S. #639443, dated 4 June 1956

1. Enclosure (A) is submitted for acceptance and approval in compliance with requirements of Reference (a).

2. It is requested that Contract [illegible] enclosed data by returning an executed co [illegible]

C O N V A I R
A Division of General Dynamics Corporation
(San Diego)

[illegible]. W. Fox
Senior Project Engineer

cc: AFPR (3) W/3 Encl. (A), (B)
LADFO
USAF (13) W/13 Encl. (B)
Dayton Ofc. (2)
Contracts
J. J. Alkazin (5)
Spec. Files (2)
Engr. Files
Model 8 P/O (3)
L. Moore (2)
E. S. Brown - Convair F.W. (2)
Acct/Bill W/Encl. (B)
[illegible]. Dublin - Dynamics

~~CONFIDENTIAL~~

PO #3598

[illegible]

CONFIDENTIAL

F 102 PO File No T-9

CONFIDENTIAL

To J. W. Fox

For Action as Indicated ☐

For Reply ☐

Comment & Return to

Contract Administration ☐

For Information ☑

☐

Copy to

File:

Contract Administrator

Copy for AFDFR

CONFID IAL
NACA - Langley

1-4626

Langley Field, Va.
March 26, 1956
Rec'd AFDFR 3 Apr. 56

MEMORANDUM For Associate Director

Subject: Telephone conversation with Mr. A. D. Mac Lellan of Convair, San Diego, concerning flight test of fifth rocket model testing F-102 wing flutter (cambered leading edges and reflex tips)

Reference:
(a) Langley Let. to AFDFR Jan. 4, 1956 FLF JAS BRO'K ncr Enc.
(b) Memorandum dated January 11, 1956 regarding rocket model flutter tests of fifth set of 1/9-scale wings (cambered leading edges) of the Convair F-102 airplane - telephone conversations with Mr. A. D. Mac Lellan of Convair, by Burke R. O'Kelly

1. On March 22, 1956, I called Mr. Mac Lellan to inform him that the subject model had been flight tested and that its performance was similar to that of the previous model (ref. (a)), that is, the wings suddenly failed as the model reached maximum speed (about M = 1.5). The wing vibrometers showed an intermittent low amplitude frequency of 85.5 cps (first bending 85 cps) beginning at sustainer firing. No other regular wing or control oscillations took place. There was no buildup to failure and there was no indication of flutter or control buzz. The wing that was repaired (ref. (b)) remained intact for a longer period than the wing which did not require repair.

2. We discussed the possibility of age weakening the bonding agent in the wing structure. Mr. Mac Lellan said that he knew that the bond lost strength with age but not how much nor how soon.

3. I asked if any more tests were contemplated because of the sudden failures encountered. He said that they were satisfied with the tests on the full-strength wings in that no flutter was encountered, but that one more test on wings of reduced stiffness was being considered. He also said that they are tentatively considering two vertical fin tests of a different configuration but approval has not yet been granted by WADC.

4. The name of the airplane was not mentioned and the conversation was held in such a way as not to jeopardize security.

RDZSFG,

Burke R. O'Kelly
Aeronautical Research Scientist

JAS
PKH
DGS
WMG
BRO'K.bes

(Typed 4/2/56)

DATE: 3 apr. '56
INFO. CYS. TO:
WCLS
Convair

TO:

JEAN G. GOPPERT, Major, USAF
Gov. Field Representative
At the NACA Laboratory
Langley Air Force Base, Va.

ADDRESS ANS
THRU THIS O

APR 10 1956

CONFIDENTIAL

L

CONVAIR
A Division of General Dynamics Corporation
San Diego

CONFIDENTIAL

Copy 2

DF-8-189

F-102 PO File C 7-9

14 March 1956

F-102 FLIGHT FLUTTER TEST PROGRAM ON LARGE TAIL AIRCRAFT

The purpose of this memorandum is to outline a flight flutter test and a ground vibration test program for an F-102A airplane with a large tail.

Introduction

The solution of the "roll-coupling" problem has been found to be enlargement of the vertical fin on the F-102A. This large tail was flight tested on S/N 53-1813 and increasing the tail size was accomplished by the addition of a "glove" to the production small tail. The regular production large tail and the retrofited large tails will be identical. No fuselage "beef-up" is contemplated because of the large tail although some frame strengthening is contemplated for other reasons. An elevation view showing the tail sizes is shown in Figure (1).

Because of the new tail and the limited speed to which S/N 53-1813 was flown (M = 1.20 at 35,000 ft., see Reference 1), it will be necessary to flight flutter test an airplane with the large tail. This flight flutter testing should be accomplished on the first airplane to be fitted with the large tail and on which the "buzz" fix has been incorporated. This is to obviate the necessity of placing a speed limitation of M = 1.20 on airplanes receiving the big tail prior to flight flutter testing.

Flight Flutter Program

In order that the flight flutter test program may be carried out in the least possible time a minimal program will be investigated. A table of the speed and altitude points to be tested is shown in Table (1).

Excitation will be provided by a rapid pulse of the aileron control at each speed point. The records are then to be analyzed by Dynamics Group personnel prior to release of the airplane to the next higher speed point.

Instrumentation

The instrumentation to be provided will consist of two accelerometers to be located on the fin and two velocity pickups to be located on the rudder as shown in Figure (2). The two velocity pickups are to be hooked up together to read rudder rotation. Outputs of these pickups are to be telemetered to the ground station and recorded on magnetic tape. These outputs are then to be recorded on an oscillograph.

Ground Vibration Tests

In addition to the above mentioned flight flutter tests, it will be necessary that the fin modes of vibration be obtained by means of ground vibration test.

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An electronic shaker is to be attached to the lower end of the rudder and a slow frequency sweep made from 5 cps to 50 cps and back to 5 cps, while recording the outputs of the ship's fin and rudder instrumentation on an oscillograph. After examination of the resulting oscillogram for resonances, the fin is to be shaken at each resonant point and the mode shape determined by a roving pickup. The pickup locations at which the roving pickup is to be placed are shown in Figure (3).

Prepared by
R. C. Peller

Checked by
R. L. Fefferman

Approved by
M. Dublin

cc: T. Maloy/Shuette
W.W. Fox/ R. Strayer
P. Prophett/ H. Cheney
G. Putness/ V. Allwardt
F. Cook/R. L. Johnson
R. L. Fefferman/R. C. Peller
M. Dublin
F. Haas/Dyn. Files

Reference:

(1) CVAC Memorandum DF-8-185, "Results of Flight Flutter Tests of F-102A S/N 53-1813 With Gloved Fin", dtd. 20 January 1956.

TABLE 1

SPEED-ALTITUDE POINTS REQUIRED FOR FLUTTER INVESTIGATION ON F-102A WITH PRODUCTION LARGE TAIL

Flight	Mach No.	Altitude
1	0.90	35000
	1.05	35000
	1.15	35000
2	1.20	35000
	1.25	35000
3	1.30	35000
	1.35	35000
4	1.40	35000
	1.45	35000
5	1.50	35000
6	1.30	25000
7	1.40	25000
8 *	1.50	25000

* Flight #8 is the maximum "q" condition.

FIGURE 1

COMPARISON OF VARIOUS F-102A FIN CONFIGURATIONS

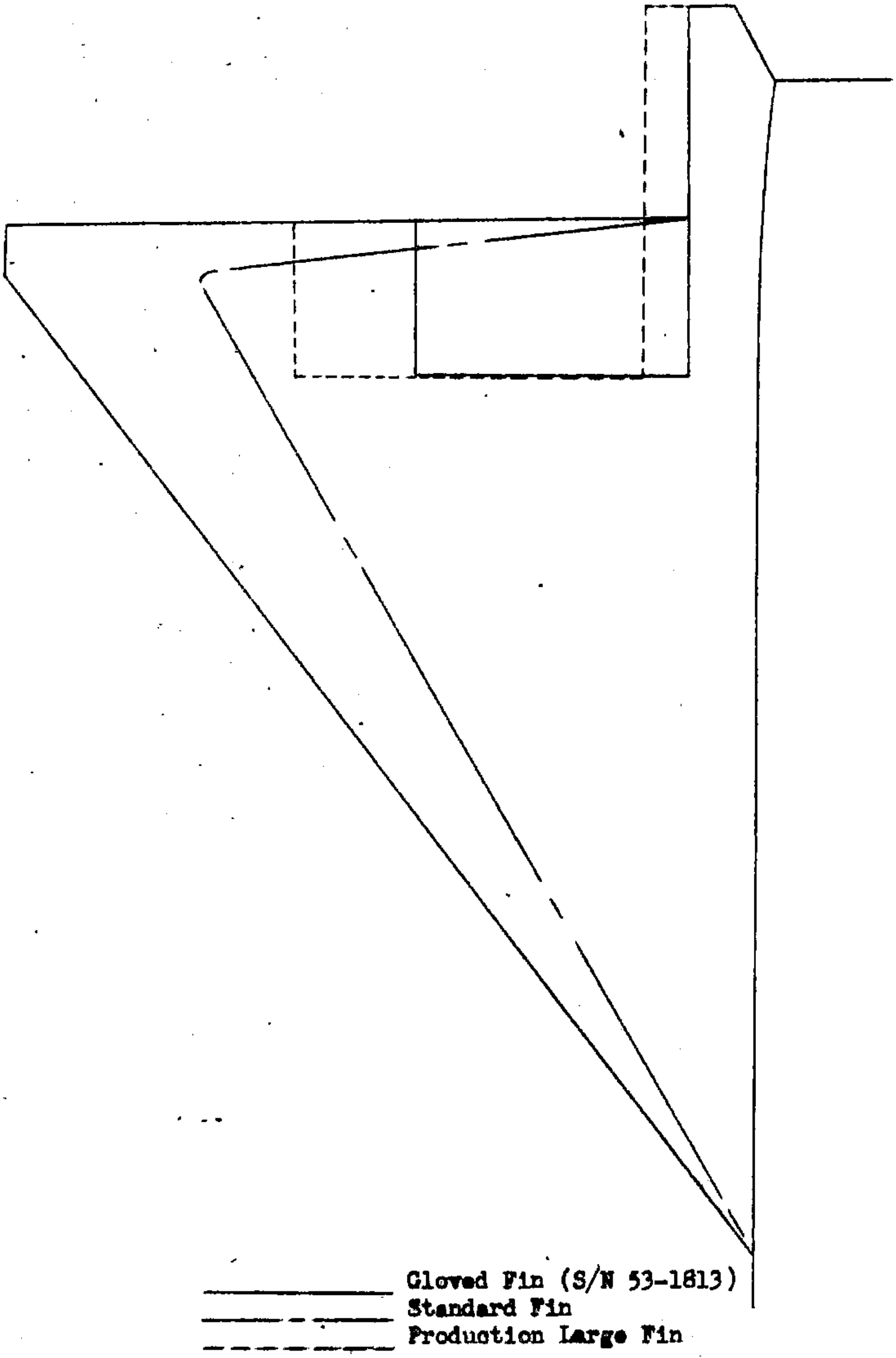
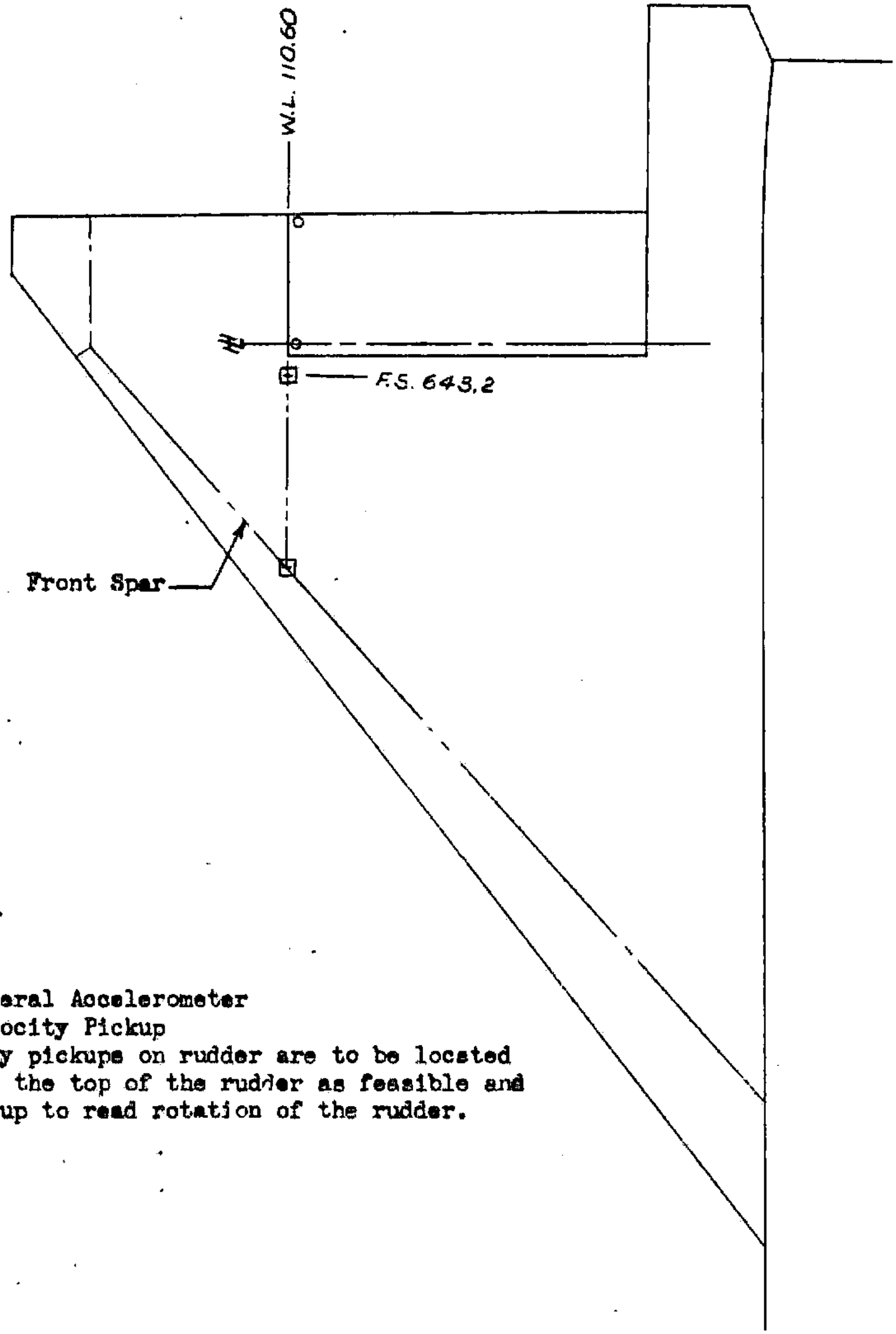


FIGURE 2

LARGE TAIL FLIGHT FLUTTER INSTRUMENTATION



Notes:

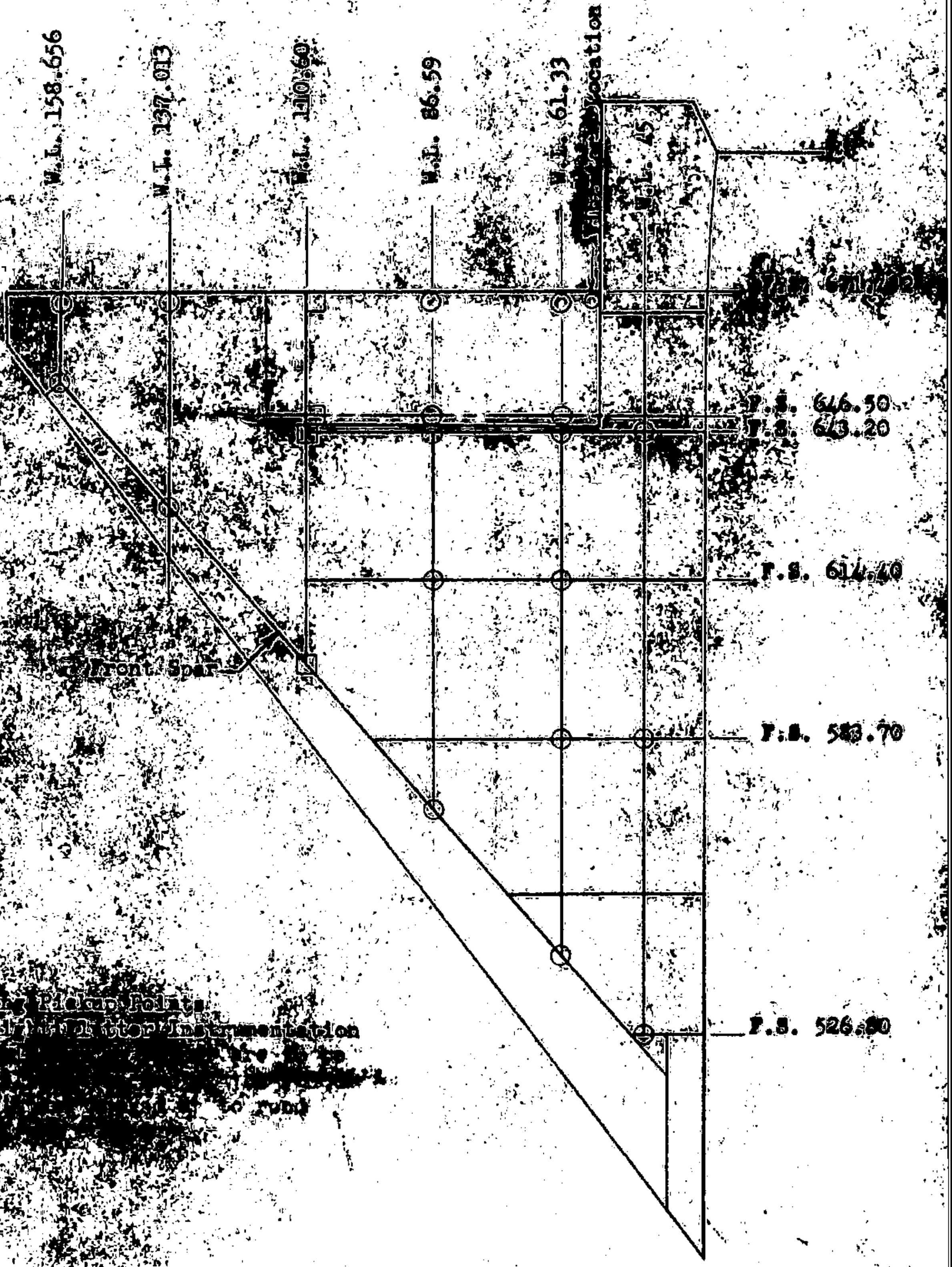
□ Lateral Accelerometer
○ Velocity Pickup

Velocity pickups on rudder are to be located as near the top of the rudder as feasible and hooked up to read rotation of the rudder.

CONVAIR
A Division of General Dynamics Corporation
SAN DIEGO

FIGURE 3

GROUND VIBRATION TEST PICKUP LOCATIONS



Notes:

○ Roving Pickup Points
⊡ Inflight Flutter Instrumentation
[illegible]

1010

1691

ACTION

REPLY

INITIALS AND DATE

HEADQUARTERS
AIR MATERIEL COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

OCT 5 1956

TO SPEED DELIVERY OF YOUR REPLY, PLEASE ADDRESS COMMUNICATION TO ATTENTION OF
MCPHFC(RDZSFG/19-1A)
JEH/ewt

SUBJECT: Contract AF 33(600)-5942
Model F-102A Airplanes
Flight Flutter, Stability and Control
and Tactical Improvement Tests

THRU: Air Force Plant Representative

F 102 PO File No T-9

TO: Convair
Division of General Dynamics Corporation
P. O. Box 1950
3165 Pacific Highway
San Diego 12, California

RECEIVED
OCT 19 1956
F-102
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REF: (a) Convair Letter 6-4779, dated 18 July 1956, w/I
(b) AMC Letter MCPHFC(RDZSFG/1-12A), dated 3 October 1956

1. Convair CONFIDENTIAL Report ZC-8-195, submitted by reference (a), has been reviewed by this Command and the following recommendations are submitted:

a. Under "Procedure" paragraph 5 of Test 2.01, Directional Stability and Yaw Damper Optimization, it is stated that only a qualitative evaluation will be made to demonstrate that right rudder will result in right skidding turns when wings are level and an increase in right bank shall accompany an increase in right sideslip (requirement 4c, page 9). Seeing that the instrumentation is available, these characteristics should be obtained quantitatively.

b. For data requirements of Test 2.03, Adverse Yaw, it states that aileron angle versus sideslip for both rudder fixed and rudder coordinated rolls will be plotted. It is not stated what sideslip values will be plotted. They should be peak values. Time histories of these maneuvers should also be included showing aileron, rudder and sideslip angles and aileron and rudder control forces. Roll-rate versus angle of bank plots are satisfactory.

c. Test 3.03 should include stick and rudder force required to hold initial trim conditions if any trim change occurs with speed brake deflection. This test should show compliance with paragraph 4.6.4 of USAF Specification 1815B.

1691

AMC ltr MCPHFC(MDZSFC/19-1A)JEE/ewt, dtd OCT 5 195[illegible] to Convair, San Diego, Calif, Subj: Contract AF 33(600)-5942, Model F-102A A/P, Flight Flutter, Stability and Control and Tactical Improvement Tests

2. Other requirements relative to flutter testing of the aircraft have been established by reference (b).

3. Upon incorporation of the items listed above in the subject flight test program, Engineering approval will be granted.

FOR THE COMMANDER:

Wm. G.

[illegible] Section
[illegible] Branch

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

F 102 PO File No 7-9

In reply refer to:
WWF:AML:sr
Letter #6-8127

14 December 1956

SUBJECT: Contract AF33(600)-5942 (-23903) (-29264) (-31174)
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model F-102A Airplanes
Flight Flutter Tests

TO: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

ATTENTION: MCPHFC (RDZSFG)

VIA: Air Force Plant Representative
San Antonio Air Materiel Area
Convair, A Division of General Dynamics Corporation
San Diego 12, California

REFERENCE: (a) AF Letter MCPHFC(RDZSFG/1-12A)JEH/ewt, dated 3 October 1956, (Confidential).

(b) Convair Memo DF-8-8, "Revised Flight Flutter Test Program on Production Big Tail F-102A Aircraft", dated 16 October 1956, (Confidential).

(c) Convair Report ZU-8-031, Add. I, "Results of Ground Vibration Tests on F-102A's S/N 53-1813 and S/N 55-3357 Large Tail Aircraft," dated 29 August 1956.

1. In accord with the request in Paragraph 2 of Reference (a), the flight flutter testing of large vertical tail F-102A Airplanes has been extended to lower altitudes in accord with Reference (b). These tests are completed and the data is being reduced for presentation in report form.

2. As requested in Paragraph 3 of Reference (a), ground vibration tests of airplanes S/N 53-1813 and 55-3357, gloved tail and production tail airplanes, respectively, have been conducted and their results are presented in Reference (c) which has been transmitted to the Air Force.

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

N. W. Fox
Senior Project Engineer

Distribution:

AMC (11)
AFPR (3)
LADFO (1)

cc: Dayton Office (2)
Contracts
J. J. Alkazin (3)
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Model 8 P/O (3)
Spec. Files (2)
M. Dublin/Y. Weisman
R. Fefferman/A. MacLellan

C O N V A I R
A[illegible]ision of General Dynamics Cor[illegible]tion
(San Diego)

In reply refer to:
WWF:AML:sr
Letter #6-8126

F 102 PO File No 7-9

14 December 1956

SUBJECT: Contract AF33(600)-30169 (-33808)
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model F-106A/B Airplanes
Flutter Model Design and Airplane Flutter Analysis

TO: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

ATTENTION: MCPHFC (RDZSFG)

VIA: Air Force Plant Representative
San Antonio Air Materiel Area
Convair, A Division of General Dynamics Corporation
San Diego 12, California

REFERENCE: (a) AF Letter MCPHFC(RDZSFG/1-12B)JEH/ewt, dated 23 October 1956.
(b) Convair Report ZU-8-053 dated 6 February 1956 (Confidential).

1. In accord with the requests in Paragraph 1 of Reference (a), Items a, b, and c have been accomplished. These items pertain to design features of F-106 Fin-Rudder Flutter Models. Items d and e, which pertain to the flutter analysis of the airplane fin-rudder, are in process of being accomplished and will be submitted as a revision to Reference (b). In reply to the question of Item f, no correction was included in the original F-106A Fin-Rudder Flutter Analysis for the fin sweepback. This will be noted in the above revision to Reference (b).

2. The transonic flutter model test noted in Paragraph 2 of Reference (a) is now scheduled for 20 February 1957.

C O N V A I R
A Division of General Dynamics Corporation
(San Diego)

W. W. Fox
Senior Project Engineer

Distribution:

AMC (11)
AFPR (3)
LADFO (1)

cc: Dayton Office (2)
Contracts
J. J. Alkazin (3)
R. L. Benson
L. Moore
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Eng. Files (2)
Model 8 P/O (3)
Spec. Files (2)
M. Dublin/Y. Weisman
R. Fefferman/A. MacLellan

CONVAIR - S.D.
EAFB

DATE: 16 February 1956

F 102 PO File No 7-9/1

TO: W. W. Fox

SUBJECT: Flight #20 Flutter and Roll Investigation at 12,000 Feet and Increased Area Vertical Tail.

During this flight, aileron pulses were used for excitation for continuation of the flutter program which was extended to 1.2M.

1.2M is the maximum to be investigated for this duct configuration because of buzz intensity.

Full deflection aileron rolls were accomplished at several mach numbers through a range of .6M to .95M at 12,000 feet. There were no in-flight indications of any undesirable characteristics. Telemetering indicated approximately 1 "G" lateral load. Roll rates were extremely high; in some instances approximately 50° in excess of requirements.

It seems reasonable and desirable to investigate roll performance using a maximum of about 5° of aileron. The 5° maximum is suggested because it would insure appreciable maximum lateral load reductions; we would retain more than adequate roll rates under most conditions and would, I believe, retain satisfactory aileron control for our extreme conditions. In addition, it could cause a quick and complete alleviation of our double-headed stick clearance problem; with a compromise change, it could cause some alleviation of clearance and cause some improvement relative to lateral control oversensitivity.

For an interim, and perhaps permanent solution to the lateral load situation, without fuselage beef-up, the aileron stops could simply be changed to approximately 5° max. then, if the present plan to alleviate lateral sensitivity works, the final solution for tail retrofit is clear.

For long range planning, and in consideration of the F-102B, a sophisticated fail-safe maximum aileron angle scheduling device should be developed. For the F-102A, the ultimate required may be a 2° change scheduled with landing gear position.

If the proposals for the retrofit case are unsatisfactory, I suggest we consider some limitation similar to "maximum aileron deflection below 20,000 feet and above 450 knots indicated air speed shall not exceed half deflection."

Could an early meeting on this subject be arranged?

R. L. Johnson
Chief Engineering Test Pilot
(303)

RLJ:MdP

DISTRIBUTION: -

R. L. Bayless
J. H. Famme
A. Burstein
M. Dublin
M. Masaki/Mancuso
Y. Weisman
W. J. Darracq
G. A. Lemke
P. M. Prophett
T. L. Maloy

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C O N V A I R
A Division of General Dynamics Corporation
(San Diego)

F 102 PO File No 7-9

In reply refer to:
WWF:LVA:co
Log #3-5168
Ltr. #6-712

THIS LETTER MAY BE DECLASSIFIED
WHEN ENCLOSURES ARE DETACHED.

3 February 1956

Subject: Contract AF33(600)-5942, 23903
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model F-102A Airplanes
Theoretical Flutter Investigation of the F-102 A Airplane

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

Attention: MCPHFC (WCSF-6)

Via: Air Force Plant Representative

Reference: AAF Specification 1817 "Flutter Divergence and Reversal in Aircraft: Prevention of"

Enclosure: (A) Convair Report ZU-8-032 "Theoretical Flutter Investigation of the F-102A Airplane"

1. Enclosure (A) is herewith submitted in accordance with the provisions of Contract No. AF33(600)-5942 and -23903.

C O N V A I R
A Division of General Dynamics Corporation
(San Diego)

W. W. Fox
Senior Project Engineer

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CONVAIR
A Division of General Dynamics Corporation
(San Diego)

F 102 PO File No T-9

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In reply refer to:
WWF:AEM:ce
Log #3-4425
Ltr. #6-6970

25 January 1956

Subject: Contract AF33(600)-30169
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model F-102B Airplanes
F-102B Airplane Revised Flutter Model Program

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

Attention: MCPHFC (WCSF-6)

Via: Air Force Plant Representative

Reference: (a) Conference at Dayton, Ohio, between W. Mykytow of WADC and R. L. Fefferman of Convair-SD on 13 October 1955

Enclosure: (A) Convair Secret Memo ZF-SB-108 dated 21 November 1955, "A Revised Program of Flutter Model Testing for the F-102B Airplane."

1. As requested in Reference (a) conference, Enclosure (A) is forwarded for approval and suggested modifications.

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

for W. W. Fox
Senior Project Engineer

Distribution:
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FEB -9 '56

NACA - Langley

PROJECT OFFICE 102 PO File No T-9A

Langley Field, Va.
January 11, 1956

(Rec'd AFDFR 24 Jan. '56)

MEMORANDUM For Associate Director

Subject: Rocket model flutter tests of fifth set of 1/9-scale wings (cambered leading edges) of the Convair F-102 airplane - Telephone conversations with Mr. A. D. Mac Lellan of Convair.

Reference: Langley Let. to AFDFR Jan. 4, 1956, Enc.

1. The fifth rocket flutter model to test Convair F-102 wings was being prepared for flight tests on January 6, 1956, when it was discovered that the upper surface skin of the right wing was loose. The landing gear door was removed and it could be seen that the skin doubler had pulled away from the spars and ribs. The area involved is about a square foot, from a line located 17 inches from the leading edge point to a line located 32 inches from the leading edge point. These lines extend spanwise from just outboard of the root chord to the second fence (about 12 inches).

2. I called Mr. A. D. Mac Lellan on the afternoon of January 6, to inform him of the skin looseness. He said that he and his associates would decide on a course of action and call me back. I told him I would determine whether or not the wing could be removed undamaged from the airframe.

3. I told him that the data letter covering results of the fourth test left Langley Field on January 4, 1956.

4. Mr. A. D. Mac Lellan returned my call on January 10, 1956, and said that Convair preferred having the wing shipped to them for repair.

5. I told him that I had conferred with NACA shop personnel concerning removal of the wing from the airframe and that they expected to have to use heat to soften the bonding paraplex resin to allow removal of the wing. Mr. Mac Lellan felt that such application of heat would probably destroy or weaken the Epon 6 bond in the remainder of the wing and he would prefer not using heat for this reason. I told him that we would decide on some other course of action.

6. I told him that we would ship the wing as soon as possible and that I would call him as to the shipping date and mode of transportation.

7. The name of the aircraft, F-102 was not mentioned during the conversation which was held in such a manner as not to jeopardize security.

Burke R. O'Kelly
Aeronautical Research Scientist

Jas

FRH

DGS

WNG

BRO'K.bss

CONFIDENTIAL Model 8

Copy No.______

C O N V A I R
A Division of General Dynamics Corporation
(San Diego)

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WHEN DOCUMENTS ARE DETACHED

In reply refer to:
AKF:RLF:mgn
6-6303
Log. No. 3-4152

24 November 1955

Subject: Contract AF33(600)-5942
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Models YF-102 and F-102 Airplanes
Flutter Model Vibration Mode Data and Test Program Changes

F 102 PO File No T-9

To: Air Force Development Field Representative
Langley Field, Virginia

Attention: D. G. Stone / B. R. O'Kelly, NACA PARD

Via: Air Force Plant Representative

Reference: (a) Telephone conversation on 19 October 1955 between B. R. O'Kelly of NACA PARD and A. D. MacLellan of Convair, San Diego

(b) Conference at Dayton, Ohio, on 13 October 1955 between W. Mykytow of WADC, E. Kats of WADC, and R. L. Pefferman of Convair, San Diego

Enclosure: (A) Node Line Plots of Resonant Modes of a Cantilevered Reduced Stiffness 1/9 Scale YF-102 Wing and Elevon

(B) Node Line Plots of Resonant Modes of aCantilevered 1/9 Scale YF-102 Wing Elevon Having Cambered Leading Edge and Reflexed Tips.

(C) Convair Confidential Report ZU-8-022, "Design and Construction of F-102A Fin Rocket Flutter Model", dated 25 May 1955.

1. As requested in Reference (a), Enclosures (A) through (C) contain the node line patterns for the cantilever modes of the reduced stiffness 1/9 scale model of the YF-102 wing and elevon, the cambered leading edge 1/9 scale model of the YF-102A wing and the reduced stiffness 1/5 scale model of the F-102A fin.

2. To confirm the information given in Reference (a), the cambered leading edge model is to be test flown to Mach number 1.6 without Thermoclad insulation instead of to Mach number 2.1 with Thermoclad insulation as originally planned. Reference (b) is the authority for this revision in the model test program.

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

for C. A. Mohr
W. W. Fox
Senior Project Engineer

Distribution:
AFDFR #1-3, w/3 copies of encls. (A) and (B)
and 1 copy of encl. (C)
AFFR #4-6, w/3 copies of encls. (A) and (B)
LADPO (1) w/o encl.

LETTER OF TRANSMITTAL MAY BE DECLASSIFIED
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cc: Dayton Office (2)
Contracts
J. J. Alkasin (2)
R. L. Benson
L. Moore (2)
W. F. Brown (2)
M. Dublin/Fefferman (2)
Engr. Files (2)
Model 8 P/O (3)

cc: Mac Lellan
W. W. Fox
~~[illegible]~~

October 28, 1955

F 102 P O File No T9
630

To: Air Force Development Field Representative - NACA, Langley Aeronautical Laboratory, Langley Air Force Base, Va.

Subject: Results of vibration studies of the fourth set of the 1/9-scale flutter wing models of the Convair F-102 airplane

Reference:
(a) Langley Let. to AFDFR Jan. 5, 1955 Enc.
(b) Memo. dated February 16, 1955 regarding conference at Pilotless Aircraft Research Division concerning degree of similitude between third 1/9-scale Convair F-102 flutter models and the prototype, by Mr. Burke R. O'Kelly

1. It would be appreciated if your office would forward the information of this letter to Convair at 3302 Pacific Highway, San Diego 12, California, marked for the attention of Mr. A. D. Mac Lellan and to WADC marked for the attention of Messrs. Henry Katz and W. J. Mykytow.

2. The fourth set of model wings was constructed similar to the wings previously investigated except that their stiffness has been reduced by 15 percent. The thermoclad coating has not been applied (as it was for the third set of wings) since the flight test will be to only M = 1.6. The vibration studies were made with the complete model mounted over the exciter in the same manner as described in reference (a).

3. The results of the vibration research are shown in figure 1. As in the vibration investigation of the third set of wings (with thermoclad coating, reference (b)) there seems to be but little agreement between the present wings and the first set of wings which were chosen as a standard of comparison at the conference of reference (b). Because these wings are of reduced stiffness, some differences are to be expected. Although the first mode frequency is high (92 cps) in the present case, the location of the node line is in agreement with previous wings. The

NOV 4 1955

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third and fourth mode nodal lines are similar in location to the seventh mode of the standard wing, the frequency of the third mode being lower and of the fourth mode being higher. A rotational frequency of 117 cps was recorded for the elevon in this study which is similar to corresponding frequencies of the first and third sets of wings. The expected elevon frequency (206 cps) could not be excited.

4. In making the enclosed data available to Convair and WADC, it should be pointed out that they are preliminary and as such are subject to further check. Any discrepancies noted therein should be brought to the immediate attention of the Langley Laboratory.

H. J. E. Reid
Director

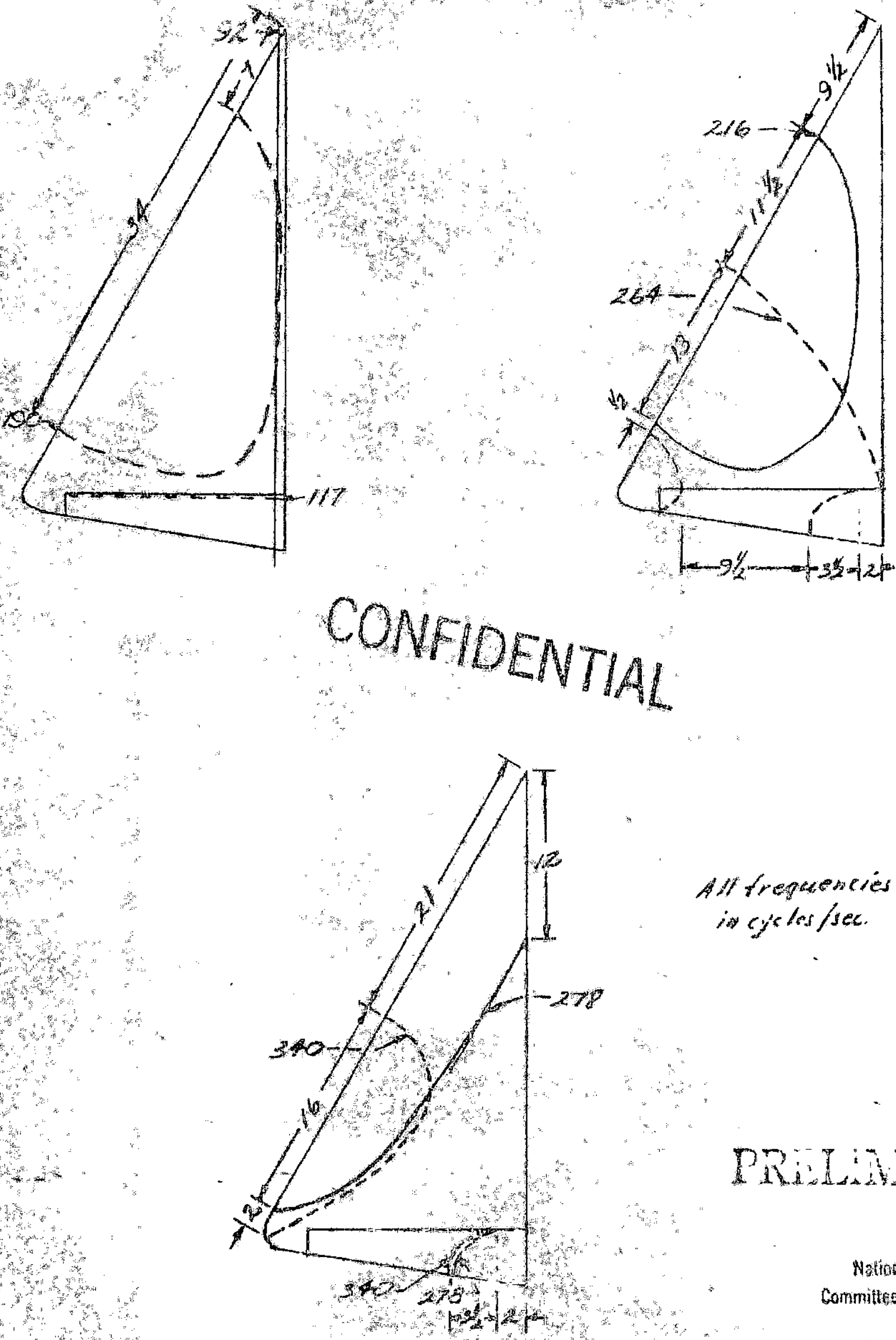
Enclosures:
1. ~~5 copies~~ figure 1
1 copy

DATE: 1 Nov 55
INFO. C.S. TO:
NACA
✓ Convair
AD...
THIRD ...

TO: ARDC, RDZSFG,

Jean F. Goppert
... F. G. GOPPERT, Major, USAF
... Representative
...atory
Langley Air Force Base, Va.

NOV 4 1955



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All frequencies in cycles/sec.

PRELIMINARY

National Advisory Committee for Aeronautics

Figure 1 - Natural frequencies and node lines for reduced stiffness wings.

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS
LANGLEY AERONAUTICAL LABORATORY
LANGLEY FIELD, VA.

NACA HEADQUARTERS
1512 H STREET, NORTHWEST
WASHINGTON 25, D. C.

AMES AERONAUTICAL LABORATORY
MOFFETT FIELD, CALIF.

LEWIS FLIGHT PROPULSION LABORATORY
21000 BROOKPARK ROAD, CLEVELAND 11, OHIO

TELEPHONE HAMPTON 8341

To [illegible] for Action as Indicated ☐
For Reply ☐
Comment & Return to ☑
Contract Administration
For Information ☐

Copy to

File: 633
Contract Administrator

cc: Pifferman

October 25, 1955

Rec'd AFDFR 26 Oct 55

1-3129

To: Air Force Development Field Representative - NACA, Langley Aeronautical Laboratory, Langley Air Force Base, Va.

Subject: Preliminary results obtained from the rocket model investigating flutter of the 1/5-scale fin of the F-102A airplane

Reference: (a) Langley Let. to AFDFR June 8, 1954 Enc.
(b) Langley Let. to AFDFR July 15, 1955 Enc.

F 102 P O File No T-9

1. It would be appreciated if your office would make the enclosed data available to Convair at 3302 Pacific Highway, San Diego 12, California marked for the attention of Mr. A. D. MacLellan and to WADC marked for the attention of Mr. W. J. Mykytow.

2. The enclosed data were obtained from a rocket model flutter investigation of a 1/5-scale F-102A vertical fin dynamically similar to the full-scale fin at 20,000-foot altitude. The rocket model, as shown in figure 1, booster and instrumentation were similar to those used in previous F-102 flutter tests. (See reference (a).) One additional telemeter channel was used to transmit model longitudinal acceleration. No temperature measurements were made. The test fin was constructed in a similar manner to the wings described in reference (a). The rudder was frictionally damped to simulate the irreversibility of the boost system in the full-scale airplane. A dummy fin having the same geometrical shape as the test fin was provided to make the model symmetrical. Both fins were covered with a coating of the thermoclad to protect them from aerodynamic heating. The frequencies of vibration and associated node lines are given in reference (b).

3. Figure 2 shows a time history of the model normal acceleration and rudder deflection. Plots of Mach number, velocity, and air density as functions of flight time are shown in figure 3. A portion of the telemeter record displaying all the channels of information is reproduced in figure 4. At M = 1.83, as the model approached maximum speed, the fin vibrometer showed a frequency of 94 cps upon which a frequency of 390 cps became superimposed after about 7 cycles. The 94 cps vibration changed to 73 cps after about 9 cycles. The lower frequency damped out fairly rapidly (M = 1.87) but the 390 cps frequency persisted (at a continually reducing amplitude) through maximum speed (M = 1.89) and

NOV 1 1955

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until the model had coasted down to M = 1.72. The frequency of 94 cps is near a natural frequency of the fin (92 cps) as is the 73 cps frequency (65 cps first bending). There was no natural frequency as high as 390 cps recorded for the model fin although a frequency of 332 cps was recorded. At the time of the initial vibration, the rudder was disturbed as may be seen in figures 2 and 4. No other oscillations were experienced which resembled flutter. However, rough air was prevalent at the time of the flight and caused other low frequency vibrations and low amplitude pitch oscillations of the model.

4. In summarizing, it may be said that results from a test of a 1/5-scale F-102A airplane fin dynamically similar to the full-scale fin at 20,000 feet show oscillations on the fin as the model approached maximum speed (M = 1.89). The initial oscillation of 94 cps (near a natural frequency of the fin) had a frequency of 390 cps to become superimposed on it. The 94 cps frequency then changed to 73 cps (near first bending frequency of the fin) with the 390 cps frequency still superimposed on it. No other oscillations were experienced except low frequency, low amplitude pitch oscillations of the model itself due to the roughness of the air prevalent at the time of the flight.

5. In utilizing the enclosed data it should be kept in mind that they are preliminary and as such are subject to further check. Any discrepancies noted therein should be brought to the immediate attention of the Langley Laboratory.

H. J. E. Reid
Director

Enclosures:
1. 5 copies each figs. 1-4

DATE: 26 October 1955 TO: Convair
[illegible] CYS. TO:
RDZSFG
WCLSY

GOPPERT, Major, USAF
[illegible]ative
[illegible] Base, Va

NOV 1 1955

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NACA - Langley

Langley Field, Va.
October 20, 1955

MEMORANDUM For Associate Director

Subject: Telephone conversation with Mr. A. D. Mac Lellan of Convair, San Diego, California, concerning rocket model flutter tests of 1/9-scale Convair F-102 wings.

1. On October 18, 1955, Mr. Mac Lellan called me to inform me that Mr. Pefferman of Convair and Mr. Mykytow of WADC had decided to test the cambered leading edge-reflex tip wings to M = 1.6 instead of M = 2.1. They asked that the thermoclad coating be removed from the wings. A letter from Convair will follow making the request official.

2. I told Mr. Mac Lellan that the data letter covering results from the 1/5-scale F-102A fin test would be transmitted in a few days. I asked him for drawings on the cambered leading edge wing as well as vibratory information on the fin, the reduced stiffness wing and the cambered leading edge wing. He said that this information would be sent along with the confirmation letter. He also asked for a test date on the weaker wings which I gave him as about mid-November.

3. The name of the airplane was not mentioned during the conversation which was held in such a way as not to jeopardize security.

Burke R. O'Kelly
Aeronautical Research Scientist

JAS

FRH

DOS

WMO wmh

BRO'K.bes

DATE: 21 Oct 55
INFO. CYS. TO:
WCLSY
✓ Convair

ADDRESS ANY REPLY
THRU THIS OFFICE

TO: ARDC-RDZSFG,

JGG

JOHN G. GOPPERT, Major, USAF
Air Dev. Field Representative
At the NACA Laboratory
Langley Air Force Base, Va.

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~~W. W. Hoyt~~ #1
~~White~~
~~Harbor~~
~~Strange~~
CO

RECEIVED
OCT 20 '55
F-102
PROJECT
OFFICE

13 October 1955

To: M. Dublin

From: R. L. Fefferman

Subject: Trip to WADC Aircraft Lab, Dayton, Ohio to Discuss F-102A and F-102B Flutter Programs

Persons contacted were W. Mykytow and H. Katz.

The following memorandums were hand transmitted to them:

DF-8-167	Add I	31 May 1955
DF-8-176		30 August 1955 (with data in excess of M = 1.2 pencilled in).
DF-8-178		5 October 1955 (with verbal comments about aircraft changes before flights can be made).
M-8-52		

F 102 P O File No T-9

Mr. Katz requested that counter number intervals on flight test reports be indicated. The case in question was ZC-8-5 12-26 for which the interval was later verified as two seconds.

Flight Flutter Tests

Results of flight flutter testing of S/N-53-1791 out to M = 1.4 were discussed. Permission was received to fly out to top speed at the high altitude without requiring shaker runs. However, WADC requests that a run in between M = 1.4 and the top speed be included and that at both speed runs, the output of the vibration pickups be recorded.

In the next summary of flight flutter tests, WADC requests further representative records and a complete plot of dynamic pressure versus altitude in the transonic range of all F-102 planes to date.

F-102A Rocket Flutter Models

Mr. Katz is preparing a reply to our letter of 5 August 1955 in which we summarized the model tests to that date, and suggested the further use of spare models. Mr. Katz's letter will state that we fire Test No. 5 with 15% reduced stiffness and fire Test No. 6 to M = 1.6 without the use of thermoclad coating. Pending the successful outcome of these two flights, the F-102A model program will be finished and permission is granted the non-destructive tests of a pair of wings as requested by Convair Structures Group. This spare model, as well as all others, shall be completed and sent to NACA.

F-102B Ground Vibration Test

Mr. Katz will address the necessary changes to the J P O in their letter of 13 August 1955. Paragraph 1a (1) will be altered to read:

> Wing tanks 2 and 3 full, fuselage tank empty, aft tank full, no external stores.

Paragraph 1a (2) will be eliminated.

Paragraph 1a (3) will read:

> Wing tanks 1, 2, and 3 full, fuselage tank full, aft wing tanks empty, 50% full external fuel tanks.

Other requests of the subject letter were agreed to by Convair representatives.

F-102A Ground Vibration Tests with External Tank

WADC will try to have representatives present, but in any case wishes previous notification of testing date. WADC requests that future ground vibration test reports contain pertinent mass parameters of movable surfaces.

F-102A Analytical Flutter Investigation

Progress of this study was outlined and I pointed out that our results indicated instability in the supersonic region where the airplane had already flown successfully. WADC recommended that we wrap up this study in a final report in which we should compare observed results with our calculations.

F-102B Analytical Flutter Investigation

WADC recommended that we perform a minimal study if we provided an adequate model program. To this end ZU-8-044 "Outline of F-102B Aerodostic Investigations" should be modified reflecting this attitude. Mr. Mykytow requested that we make a fin-rudder analysis immediately.

F-102B Flutter Model Study

The proposed Convair program outlined in DF-8B-104 "A

Practical Design and Test Program for a 1/8 Scale F-102B Flutter Model" dated 12 August 1955 was unacceptable to WADC. Convair representative pointed out that limited data from flight flutter surveys indicated that higher altitude conditions produced lower damping and that our program was designed to model test the entire airplane at highest Mach number and at an altitude in excess of the minimum at which this speed could be attained. Notwithstanding, WADC is most apprehensive about the flutter characteristics in the transonic regime at highest g. Means of attaining WADC desires were discussed. Convair representatives elected to take matter under advisement and propose a new program.

R. L. Fefferman
Design Specialist

cc: W. W. Fox
T. Maloy/M. Edelstein
W. Mykytow
Dynamic Files

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Corporation

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OCT 11 '55
F-102
PROJECT
OFFICE

DF-8-178

5 October 1955

STATUS OF F-102A FLIGHT FLUTTER TESTS

F 102 P O File No T-9

Introduction

Data obtained on Flight 30 of F-102A S/N 53-1791 have proved adequate for completion of the 35,000 foot phase of the flight flutter tests. The test outline presented in Reference 1 calls for testing at 35,000 feet to a Mach no. of 1.5; however, extrapolations of data accumulated at various speed points up to $M_n = 1.4$ indicates the presence of adequate damping at Mach numbers considerably in excess of 1.5 at 35,000 feet. *

The completion of the modified program outlined in Table I requires that several flights be made at test altitudes of 25,000 and 15,000 feet.

Discussion

Lifting of the flutter restriction on the test airplane for Mach numbers up to 1.5 above 35,000 feet is considered to lift the flutter restriction below 35,000 feet indicated airspeeds equal to or less than that achieved at $M_n = 1.5$ and 35,000 feet (i.e. 545 Knots but not greater than $M_n = 1.5$)*

Completion of flight flutter testing at 25,000 and 15,000 feet will of necessity require that flights be made during those times of day when atmospheric turbulence at test altitudes will be at a minimum.

Recommendations

The raised limitations on the test airplane offer considerable latitude for conducting other essential tests not related to flutter; consequently it is suggested that a program of non-flutter testing be established which will be tolerant of interruption when conditions are appropriate for conducting flight flutter testing (i.e. smooth air at test altitude).

* Flutter restrictions for other F-102A airplanes are defined in Memorandum M-8-91

Prepared by: H A Mitchell
H. A. Mitchell

Checked by: R L Fefferman
R. L. Fefferman

Approved by: M Dublin
M. Dublin

K. Adrian
G. Putness/R. L. Johnson/V. Allwardt

cc: W. W. Fox
P. Prophett/H. Chevey
T. Maloy
M. Dublin/ Dyn Files

Reference

1. Convair Memorandum DF-8-167 "Outline of F-102A S/N 53-1791 Flight Flutter Tests for Configuration Without External Wing Tanks" revised 18 May 1955

2. Convair Memorandum DA-8-331 "Dive and Pull-Out Performance — F-102A S/N 53-1794 Flight Air Loads Flight Test Program"

TABLE I

REVISED F-102A FLIGHT FLUTTER PROGRAM

Run No	M	h	V_t	q_c	Remarks
1*	1.50	35,000	995	1195	No Record
2	1.10	25,000	761	896	Sweep Records
3	1.20	25,000	831	1110	Decay Records
4	1.30	25,000	900	1340	Decay Records
5	1.40	25,000	969	1600	Decay Records
6*	1.50	25,000	1038	1880	No Records
7	.90	15,000	649	830	Sweep Records
8	1.10	15,000	793	1350	Decay Records

Legend:
M = true Mach number
h = Pressure altitude (feet)
V_t = True air speed (MPH, standard day)
q_c = Compressible dynamic pressure (lbs/ft^2, standard day)

Note: * See Reference 2

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Contract Administrator

NACA - Langley

Langley Field, Va.
August 10, 1955

Rec'd APDFR 17 Aug. 55

MEMORANDUM For Associate Director

Subject: Telephone conversation with Mr. A. D. Mac Lellan of Convair, San Diego, concerning the rocket model flutter test of the 1/5-scale F-102A vertical fin

F 102 P O File No T-9

1. On August 9, 1955, Mr. A. D. Mac Lellan of Convair, San Diego, called me to obtain a progress report on the rocket flutter model testing of the 1/5-scale F-102A vertical fin. I told him that the model was being instrumented at this time and firing would be about August 15 to 25, 1955. He wanted to be notified when a more definite testing date is known, but he did not say definitely that he would witness the test.

2. I asked him when the next models would be shipped and he said in about a week when their vibrational tests are completed. In one set, the wings are of reduced stiffness to be tested to M = 1.6 and in the other set the wings have cambered leading edges and reflex tips to be tested to M = 2.0. The last mentioned wings will have a Thermoclad coating which will be thinner than the coating applied to previous models.

3. The name of the airplane F-102, was not mentioned and the conversation was held in such a way that security would not be jeopardised.

Bob
Burke R. O'Kelly
Aeronautical Research Scientist

JAS
PRH
DOS
WWO
BRO'K.bms

DATE: 17 aug. 55
INFO. CYS. TO:
WCLS
Convair

ADDRESS ANY REPLY
THRU THIS OFFICE

TO: WRIGHT AIR DEVELOPMENT CENTER, WCSF

JEAN G. RUPPERT, Major, USAF
Air Dev. Field Representative
At the NACA Laboratory
Langley Air Force Base, Va.

AUG 24 [illegible]

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For Information

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M Dublin
Colliezzi
Brown
EG
AD

mmFox
White

HEADQUARTERS
AIR MATERIEL COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

AUG 8 1955

TO SPEED DELIVERY OF YOUR REPLY, PLEASE ADDRESS COMMUNICATION TO ATTENTION OF

MCPHFG(RDZ-1SFG/1.12)
WFB/eb

What are we doing on this [illegible]
proceeding with flutter model

SUBJECT: (Uncl) Contracts AF 33(600)-5942, -23903, 29264
F-102 Flutter Status

THRU: TO:
Air Force Plant Representative
Convair
Division of General Dynamics Corporation
P. O. Box 1950
3165 Pacific Highway

F 102 PO File No T-9

Ref: (a) Convair ltr 6-1797, dtd 7 Apr 55, w/incl.
(b) Convair ltr 6-1963, dtd 15 Apr 55, w/incl.
(c) Meeting 13 May 55 with personnel of WADC Aircraft Lab and Mr. N. Dublin, Convair, San Diego.

1. References (a) and (b) have been reviewed by this Command and the following comments are made:

a. This Command concurs that the results of the F-102A flutter investigations will also apply to the YF-102A.

b. Based upon the data of the referenced reports, the YF-102A aircraft appears to be free from flutter to M=1.32 or 544 KIAS, whichever is less. It is understood that the YF-102A aircraft will be restricted to these speeds, at least until flight flutter tests on the F-102A airplane are completed.

c. This Command does not concur with the contractor that the F-102B aircraft is sufficiently similar to the YF-102A to preclude flutter testing of a F-102B model.

2. During reference (c) meeting the relative merits of testing a composite F-102B model as opposed to tests of individual components on several models were discussed. Further investigation

AUG 1 8
AUG 1 8 1955

55MCP-18778

AMC Form 9, 28 Aug 52

"MORE AIR FORCE PER DOLLAR"

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of the subject was felt necessary. It is requested that plans for a F-102B flutter model be submitted to this Command for review.

FOR THE COMMANDER:

CARL F. DAMBERG
Colonel, USAF
Chief, Aircraft Division

AIR [illegible] [illegible] Gen. [illegible]
[illegible] 1954
Y Jr, Colonel, USAF

Capt.

Model 8
Copy 29

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

Log No. 3-2945
In reply refer to:
WHF:ADM:bh
6-6235

5 August 1955

Subject: Contract AF33(600)-5942
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model YF-102 Airplane
Flutter Model Program Resume'

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

F 102 PO File No T-19

Attention: MCPHFC (WCSP-6)

Via: Air Force Plant Representative

Reference: (a) Convair Confidential Memo M-8-43, "Trip Report to WADC on Model 8 Aeroelastic Investigations", dated 24 May 1955 - copy to W. Mykytow (Aircraft Lab. - Dynamics Branch)

1. In order to formulate the remainder of the free flight flutter model program it is desirable to appraise results and current work to date. Table I presents a summary of the model tests completed, planned and suggested.

2. Flight testing of the cambered leading edge wing model for test 6 noted in Table I will complete testing of YF-102 wing models of the original structural configuration. Tests of the reduced stiffness wing models for flight tests 5 and 7 will complete all testing of YF-102 wing models. Test 5 is made to Mach 1.6 as requested by the Air Force in Reference (a). Test 7 is suggested to investigate the speed range up to Mach 2.1 with the minimum thermoclad coating considered safe, based on the results of other flights of coated models.

3. As indicated in the table, the reduced frequency F-102A fin model will be tested to Mach 2.1. If this test is successful, it is anticipated that no further testing will be required and the fin model designed to directly simulate the airplane fin will be a spare.

4. Since two spare pairs of YF-102 wing models will remain, the Contractor's Structures Group has become interested in possible acquisition

of one pair. This pair of wings would be used for experimental determination of the structural characteristics of a delta wing structure. New theoretical work is planned which would be greatly assisted by structural tests of such close tolerance models. The determination of structural deformations and skin stress distribution for various loadings would be prime objectives of such investigations.

5. It is requested that comments or approval of Table I and the suggested use of a spare pair of wing models as noted in paragraph 4 above be given as soon as possible.

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

for C. O. Mohr
W. W. Fox
Senior Project Engineer

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Model 8 P/O	29-31	

TABLE I

F-102 Flutter Model Test Program Summary

Order of Test	Model Type	Test Mach No.	Completion Date	Test Date
1.	1/9 Scale Wing - Orig. Config. (Uncoated)	1.35	12-8-53	2-25-54
2	Same as 1.	1.93	2-17-54	5-6-54 (Structural Failure)
3	1/9 Scale Wing - Orig. Config. (Thermoclad coated)	1.97	11-3-54	5-15-55
4	1/5 Scale Fin - 15% Reduced Freq. (Thermoclad coated)	2.1	5-26-55	8-8-55 (Estimated)
5	1/9 Scale Wing - 15% Reduced Stiffness (Uncoated)	1.6	8-19-55 (Estimated)	10-19-55 (Estimated)
6	1/9 Scale Wing - Cambered L.E. (Thermoclad coated)	2.1	8-10-55	11-14-55 (Estimated)
7 (Tentative)	1/9 Scale Wing - 15% Reduced Stiffness (Uncoated)	2.1	12-16-55 (Estimated)	3-14-56 (Estimated)
Not Scheduled	1/9 Scale Wing - Orig. Config.		Spare	
Not Scheduled	1/9 Scale Wing - Orig. Config.		Spare	
Not Scheduled	1/5 Scale Fin - Direct Simulation		Spare	

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

In reply refer to:
WWF:AML:bh
6-2514

12 May 1955

Subject: Contracts AF33(600)-5942 and -23903
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model F-102A Airplane
Fin Flutter Model and Flight Flutter Testing
of Airplane with External Fuel Tanks

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

102 PO File No T-9

Attention: MCPHFC (WCSF-6)

Via: Air Force Plant Representative

Reference: (a) AMC letter MCPHFC (WCSFG/19) WFB/eh dtd 8 Feb 1955
(b) Convair Conf. Memo DF-S-167 "Outline of F-102A S/N 53-1791 Flight Flutter Tests" dtd 25 April 1955

1. In reply to paragraph 2.a of Reference (a), it has been mutually agreed upon by Air Force and Convair representatives that one fin-rudder model shall incorporate reduced fin root attachment fixity and reduced outer skin thickness. This reduction will give, as near as possible, a fifteen percent reduction in frequencies. The completed fin-rudder model which had been built as a direct simulation of the full scale article will be held as a spare.

2. With respect to flutter testing for the airplane with external fuel tanks as noted in paragraph 2.b of Reference (a), it is the contractor's intention to submit a detailed outline of flight flutter tests similar to Reference (b). This outline will be released in the near future, prior to availability of external tanks, pylons and related equipment, citing specific test procedures for the airplane with full external tanks and with half-full tanks having a forward c.g. in a dive and an aft c.g. in a climb, as recommended.

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

W. W. Fox
W. W. Fox
Senior Project Engineer

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~~W Fox~~
Wagner
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TO SPEED DELIVERY OF YOUR REPLY, PLEASE ADDRESS COMMUNICATION TO ATTENTION OF
MCPHFC(WCSFG/19-2)
WFB/eh

SUBJECT: Contract AF33(600)-5942
1/9 Scale Wing Model with Thermoclad Coating - Dynamic Similitude of

THRU: Air Force Plant Representative

F 102 PO File No T-9

TO: Convair
Division of General Dynamics Corporation
P. O. Box 1950
3165 Pacific Highway
San Diego 12, California

Ref: Convair ltr 6-1299, dtd 14 Mar 55, w/incls.

1. The referenced data has been reviewed and it has been concluded that the thermoclad covered model (model no. 3) has dynamic properties sufficiently close to those of the actual airplane to permit firing of the model. The data on thermoclad effectiveness gained from this model would also be applicable to future themoclad-covered models that will be tested to substantiate safety from flutter of the airplane.

2. This letter confirms telecon authority to proceed with the test between the W.A.D.C. and Convair and W.A.D.C. and P.A.R.D., NACA, Langley.

FOR THE COMMANDER:

Bruce Calhoun Col USAF
for CARL F. DAMBERG
Colonel, USAF
Chief, Aircraft Division

IND.
AFPR Office, SAAMA, Convair - A Div. of Gen. Dynamics, San Diego 12, Calif. 21 Apr 55
To: Convair - A Div. of Gen. Dynamics, San Diego
Attn: Contr. Adm.
Parts Sales
GFP Group

Forwarded
for JAMES F. McCARTHY, Jr., Colonel, USAF
AF Plant Representative

Hq AMC Form 9, 25 Aug 52

"MORE AIR FORCE PER DOLLAR"

Models & P/O
Log No. 3-1533
Copy #23

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

In reply refer to:
WWF:RLF:bh
6-1299

14 March 1955

Subject: Contract AF33(600)-5942
Convair, A Division of General Dynamics Corporation
San Diego 12, California
Model YF-102 Airplane
1/9 Scale Wing Model with Thermoclad Coating -
Dynamic Similitude of

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

Attention: MCPHFC (WCSF-6)

F 102 PO File No T-9

Via: Air Force Plant Representative
San Antonio Air Materiel Area
Convair, A Division of General Dynamics Corporation
San Diego 12, California

Reference: (a) NACA Memo for Associate Director from B. R. O'Kelly, subject, "Conference at Pilotless Aircraft Research Division concerning degree of similitude between third 1/9-scale Convair F-102 flutter models and the prototype", dated 16 February 1955

(b) NACA Letter to AFDFR/NACA, Langley AFB, Va., subject, "Results of vibration tests of third set 1/9 scale Convair F-102 flutter wings", dated 25 February 1955

Enclosure: (A) Figure 1 - Wing mode shapes and frequencies measured on two uncoated sets of 1/9 scale models mounted on simulated fuselage.

(B) Figure 2 - Cantilever modes and frequencies for 1/9 scale wing panels with and without thermoclad coating.

1. Paragraph 4 of Reference (a) indicates the procedure to be followed for establishing the validity of the third pair of 1/9 scale YF-102 wings and elevons with respect to dynamic similarity. The procedure as stated in Reference (a) was agreed to by Air Force, NACA and Convair representatives present at the conference noted in Reference (a).

2. Reference (b) contains the results of tests described in part (d) of paragraph 4 of Reference (a).

3. Enclosures (A) and (B) contain the results of tests described in parts (a), (b) and (c) of paragraph 4 of Reference (a) which are submitted for comparison with results noted in paragraph 2 above. The test conditions for the two "cantilever" tests differ sufficiently to influence the mode shapes. The NACA rocket casting in which the model wings are anchored introduces flexibility not present in the Convair test since the Convair test was made with the wings cantilevered from a test jig.

4. A comparison of the test results discussed above indicates the satisfactory similitude of the third set of wing models. Permission is therefore requested to proceed with flight testing at the earliest possible date.

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

W. W. Fox
W. W. Fox
Senior Project Engineer

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900

TO SPEED DELIVERY OF YOUR REPLY, PLEASE ADDRESS COMMUNICATION TO ATTENTION OF
MCPHFC(WCSFG/19)
WFB/eh

F 102 PO File No T-9

SUBJECT: Contract AF 33(600)-5942
F-102A Flight Test Flutter Program

THRU: Air Force Plant Representative

TO: Convair
Division of General Dynamics Corporation
P. O. Box 1950
3156 Pacific Highway
San Diego 12, California

To W.W. Fox + copies
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Ref: (a) Convair ltr 6-5319, dtd 8 Nov 54
(b) Convair ltr 6-4670 dtd 1Oct54, w/incl.

1. The comments of reference (a) have been received and have been noted.

2. Convair report ZU-8-025, dtd 25 Aug 54, as submitted by reference (b) has been reviewed and is granted engineering approval with the following contingencies:

a. That the fin and rudder frequencies of the F-102A fin-rudder model are reduced by 15%, in order to incorporate a margin of safety in the transonic region.

b. Reference is made to page 9 of the referenced report. It is not believed that the results of theoretical calculation will be sufficiently reliable to indicate whether the wings with external stores are critical from a flutter viewpoint. Model tests of wings with external tanks may not give results applicable to the airplane since during these tests the fuselage mass and geometry (which may be of considerable importance for this condition) will not be simulated. At this time it appears that the only conclusive

"MORE AIR FORCE PER DOLLAR"

AMC ltr MCPHFC(WCSFG/19) dtd [illegible] to Convair, San Diego, Calif., Subj: "Contract AF 33(600)-5942, F-102A Flight Test Flutter Program"

results for the wing with external store configuration can be obtained from flight flutter tests for at least three fuel weight conditions: external tanks full, external tanks half-full with the fuel c.g. as far forward as possible (dive), and external tanks half-full with the fuel c.g. as far aft as possible (climb). It is therefore requested that Convair submit a more detailed flight flutter program for the external store configuration.

FOR THE COMMANDER:

Bruce C. Downs Col. USAF

for CARL F. DAMBERG
Colonel, USAF
Chief, Aircraft Division

IND.
AFPR Office, [illegible] NA, Convair - A [illegible] FEB 15 1955
San Diego 12, Calif
To: Convair - A Div. of Gen Dynamics Corp, San Diego
Attn: Contr. Adm.
Parts - los
G. F Group
2
Forward
for [illegible] CARTHY Jr., Colonel, USAF
[illegible]

FPW

In reply refer to
WWF:RLF:sb
6-5319

8 November 1954

Subject: Contract AF33(600)-5942
Consolidated Vultee Aircraft Corporation
San Diego 12, California
Model F-102A Airplane
Test Program for F-102A Free Flight Rocket Flutter Models

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

Attention: MCPHFC (WCSF-6)

F-102 PO File No T-9

Via: Air Force Plant Representative
San Antonio Air Materiel Area
Consolidated Vultee Aircraft Corporation
San Diego 12, California

Reference: (a) NACA ltr from Director to AFDFR, Langley AFB, dtd. 13 October 1954, Subject "Free Flight Flutter Test of 1/9 Scale Dynamically Similar Convair F-102 Wings and Elevons"

1. The following comments are with regard to questions raised in Reference (a) with respect to the drawings submitted of the 1/5 scale F-102A fin flutter model:

a. Part -49 has sufficient joggle to permit angular travel in each direction of approximately six (6) degrees, which is adequate.

b. A dummy omniloop antenna is being installed on the dummy fin and is so indicated in the latest issue of WT-54-8423.

2. With regard to the model test program, Table I is presented to indicate the engire F-102 flutter model test program as presently planned by Convair. As noted, Air Force approval is requested for modifications and additions to the program.

TABLE I

F-102 Flutter Model Test Schedule (Subject to Air Force Approval)

Order of Test	Specimen Characteristics	Mach No.	Est. Ship. Date	Est. Firing Date
1. (D-18-1872)	Wing - Orig. Config. f = 243 (25 cps)	1.35	Rec'd.NACA 12-17-53	Fired 2-25-54
2. (D-18-1873)	Wing - Orig. Config. f = 243 (25 cps)	2.1	Rec'd NACA 2-24-54	Fired 5-6-54
3. (D-18-1254)	Wing - Orig. Config (Coated) f = 243 (25 cps)	2.1	11-5-54	Jan. '55
4.	Fin - F-102A Config.	2.1	12-15-54	Jan. '55
5.	Wing - Reflex Tip & Cambered L.E. f = 243 (25 cps)	2.1	12-1-54	Feb. '55
6.	Wing - Reduced Stiffness f = 206 (21 cps)	2.1	1-15-55	Apr. '55
7.	Wing - Orig. Config. with External Tanks f = 243(25 cps)	1.3	2-1-55	May '55
8.	Wing - Spare		Not Scheduled	
9.	Wing - Reduced Stiffness - Spare		Not Scheduled	
10.	Fin - Spare		Not Scheduled	

Note:

1. AF approval for alternate use of wing D-18 test vehicles for fin model test is requested.

2. AF authorization for a total of ten test vehicles is requested.

3. Test #7 to be conducted if calculations made by Convair indicate the necessity.

4. Specimens for test #8, 9 and 10 will be used in the event of failure of one of the previous specimens or to investigate questionable airplane configuration changes.

Notes: (continued)

5. "f" indicates elevon rotational frequency in cycles/sec. and parenthesis enclose corresponding full scale airflow value.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION

W. W. Fox
Senior Project Engineer

cc: AFPR (3)
LADFO

cc: USAF (10)
Dayton Office (2)
Contracts
J. J. Alkazin (2)
Al Gross
R. L. Benson
M. W. McCreary
O. C. Priest
M. Dublin
R. L. Fefferman
Engr. Files
Model 8 P/O (4)

Original to M. Dublin

To W. [illegible]

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D. D. DENCY, JR.

NACA - Langley

Langley Field, Va.
June 18, 1954
Rec'd AFDFR 24 June 1954

MEMORANDUM For Associate Director

F - 102 P O File No T-9

Subject: Visit of Convair representative to discuss the rocket model investigation of flutter tests of F-102 scale model wings.

Reference: Memorandum for Associate Director regarding telephone conversation with Convair - San Diego concerning results of second flutter test of model wings of F-102 airplane, dated May 26, 1954, by David G. Stone.

1. On June 10, 1954, Mr. Mike Dublin of Convair - San Diego visited Pilotless Aircraft Research Division for the purposes of discussing the results of the two rocket model flutter tests and the future program of flutter tests of the dynamically scaled model wings of the F-102 airplane. This visit was the result of the agreement as stated in the telephone call of the reference memorandum. Representatives of the Wright Aeronautical Development Center were unable to attend. Others present were Mr. Jean A. Roche of the Air Force Development Field Office, and Mr. B. R. O'Kelly and Mr. D. G. Stone of PARD.

2. The second model, which is of most importance, experienced a sudden failure of the wings at a Mach number between 1.9 and 2.0. We were in agreement that this failure was structural and the most likely cause was the effect of aerodynamic heating which lead to buckling and failure of the unsupported skin panels or caused the bonding agent (Shell Epon VI) to lose its strength. Therefore further flutter tests could not be made until a fix was found to overcome the effects of aerodynamic heating. It was decided that Convair would immediately start some laboratory tests to determine whether the skin buckled and failed or the Shell Epon VI bond was losing strength at elevated temperatures. An outline of the laboratory program to be run at Convair will be similar to as follows:

(a) Heat along leading edge of a wing (they have 4 sets of model wings fabricated at Convair) to about 350° F to observe whether the skin buckles or the bond parts. Also, try to vibrate the wing when under this non-uniform temperature distribution, especially near the frequency observed at failure.

(b) Test specimens of Epon VI bonding dural to dural in an oven near 350° F to determine bond characteristics at elevated temperatures.

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Also, repeat tests using Epon IX which reputedly has better high temperature strength characteristics.

(c) Investigate the use of heat protective coatings on metal such as Phenoline "300", lithium, and Converplate.

The fixes necessary then may be singly or any combination of the following:

(a) Stiffen the unsupported panels which would tend to violate the dynamic scaling.

(b) Bond the dural structure together with Shell Epon IX instead of Epon VI.

(c) Cover the wings with a protective coating which might tend to violate the dynamic scaling.

3. Assume then that a fix can be found, the rocket model flutter test program would shape up as follows. The next model (No. 3 to be tested) would have a flat chord plane and the stiffest elevon spring restraint (the model is the same test as number 2) and the No. 4 model would have reflex tips and cambered leading edge. All tests would be to a Mach number of 2.0, and no more tests will be made to only M = 1.3 inasmuch as it is Convair's opinion that the airplane is flutter-free up to M = 1.29 as a result of their current flight test program on the F-102 and the initial rocket model test. This then may eliminate two rocket models that were planned originally and leave only two more additional tests (Nos. 5 and 6) which will be wings of reduced stiffness (the wings of these models have not been assembled). Also, it is planned to add two telemeter channels to each of all the models to measure skin temperature somewhere near the leading edge and at the rear skin of the wing ahead of the elevon. At this time it appears that all models will have the stiffest spring restraint on the elevons (and the full-scale airplane will retain the stiffest restraint) because in the light of the slight buzz obtained on the first rocket model test and experience with the airplane it would be dangerous to reduce the restraint of the elevon.

4. Inasmuch as WADC was not in on these decisions and Mr. Dublin was going to visit WADC, he said he would relate the results of this conference to WADC and let me know if the Air Force concurred. On June 16, 1954, I received a telegram from Mr. Dublin stating WADC was in agreement with the above program. After his visit at PARD Mr. Dublin visited the Structures Research Division to discuss briefly aerodynamic heating and laboratory techniques to simulate heating.

WRIGHT AIR DEVELOPMENT CENTER WCSF

DATE: 24 June 54
INFO. CYS. TO:

David G. Stone
David G. Stone
Aeronautical Research Scientist

JAS
PRH
DGS.bas

JEAN G. GOPPERT, Major, USAF
AF Dev. Field Representative
At the NACA Laboratory
Langley Air Force Base, Va.

ADDRESS ANY REPLY
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W.W. Fox

[illegible]

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
San Diego, California

April 20, 1954

Memo to: E.D. Shannon

From: M. Dublin

F-102 PO File No T-9

Subject: F-102 Flight Flutter Test Program

The examination of vibration records obtained during flights 33 through 37 leads to the following recommendations for subsequent flight flutter testing.

a. Make a dive to achieve M≃1.30 at approximately 30,000 feet with symmetric excitation sweep. Since the speed varies during the 30 second period required for the sweep it is suggested that the dive commence at 47,000 feet and that the vibrators be turned on at approximately 40,000 feet.

b. Repeat item (a) with anti-symmetric excitation.

c. The necessity for obtaining flutter data at higher dynamic pressures will be determined from the results obtained from (a) and (b) above and consideration of practical dives that can be made at lower altitudes.

d. For these tests, use the present recording instrumentation (i.e., as used in Flight #17).

M. Dublin

cc. E.D. Shannon
A. Burstein
R.L. Bayless
P.M. Prophett
R. Johnson/V. Allwardt (EAFB)
W.W. Fox
M. Dublin
R.L. Fefferman
Dyn. Files

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CONVAIR

In reply refer to:
6-1384
RRH:MD:md

26 March 1954

Pilotless Aircraft Research Division
National Advisory Committee for Aeronautics
Langley Field, Virginia

Attention: Mr. David G. Stone

F-102 P O File No T-9

Subject: Telephone Conversation Concerning Initial Rocket Model Flutter Test of F-102 Wings Referred to in Memorandum from David G. Stone to the Associate Director, NACA, Langley Field, dated 8 March 1954

1. A copy of the subject memorandum has been received and is hereby acknowledged.

2. In paragraph 1. of the subject memorandum it was stated that the "buzz" on the elevon of the actual F-102 airplane was "about half the amplitude as obtained on the rocket model", which was ±3°. This is in error; the maximum amplitude on the actual airplane elevon was ±.15°, i.e., fifteen hundredths of a degree plus or minus.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION

R. R. Hoover
Project Engineer

Addressee (2)

cc: AMC-MCPPAF-6 (WCOWF-6)
AFPR/CVAC

cc: M. Dublin/R. Fefferman
Dyn. Files
Engr. Files
Mod. 8 P/O (2)

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Contract Administration

WCPAF-6(WCSF-6)
RGA/eh

10 March 1953

SUBJECT: (Uncl) Contract AF 33(600)-5942
1/9 Scale Rocket Flutter Models of Reduced Stiffness

THRU: Air Force Plant Representative

TO: Consolidated Vultee Aircraft Corporation
P. O. Box 1950
3156 Pacific Highway
San Diego 12, California

Copies to: Dublin(2)
Reply
Bayless
Albrayin(2)
E.F.

Ref: CVAC ltr 6-6224, dtd 23 Dec 53.

F-102 P O File No T-9

1. The intent in recommending tests of models which have an overall 15% reduction in stiffness was not to investigate the possibility of flutter under the supposition that the stiffness might actually be reduced in flight due to, say, aerodynamic heating effects. The idea behind this recommendation was to determine whether or not the model (and hence the F-102) had a 15% margin of safety against flutter in terms of the stiffness of the overall structure. For aircraft with subsonic performance only, it is sufficient to determine the flutter margin of safety in terms of speed; i.e., it is required to show that the lowest critical flutter speed of the aircraft is at least 15% above the maximum flight speed. However, for aircraft with supersonic performance, the idea of a flutter margin in terms of speed has little significance since critical flutter conditions may occur at Mach numbers below that corresponding to the maximum speed. Hence it is more desirable to determine the flutter margin of safety for the F-102 in terms of structural stiffness, and this, not the simulation of aerodynamic heating effects, was behind the recommendation for testing flutter models with reduced stiffness.

2. It is therefore considered that the Contractor's request to reduce only the torsional stiffness of the two additional 1/9 scale flutter models should be reconsidered and Convair should proceed with the fabrication of the reduced stiffness models in which all

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547CPF-06956

structural elements are uniformly cut down in stiffness by 15%. If, however, Convair can present valid reasons why it is not feasible to do this, then the original request to reduce only the torsional stiffness of the two additional models by decreasing the wing skin gage will be reconsidered. However, it will then be recommended that the elevon rotational restraint be proportionally reduced as well.

FOR THE COMMANDER:

H. H. ROWE, JR.
Colonel, USAF
Chief, Aircraft Branch
Procurement Division

[illegible] Convair [illegible] MAR 18 1954
San Diego [illegible]
[illegible] CVAC, Attn: Contr. Adm.

Forwarded

STEP[illegible] Colonel USAF
AF Plant [illegible]

54MCPP-66956

CONVAIR

In reply refer to:
RRH:CWR:md
6-707

19 February 1954

Subject: Precision Free Flight Range;
Proposal for

To: Commander
Air Materiel Command
Wright-Patterson Air Force Base, Ohio

Attention: MCPPAF-6 (WCOWF-6) F-102 PO File No T-9

Via: Air Force Plant Representative
San Antonio Air Materiel Area
Consolidated Vultee Aircraft Corporation
San Diego 12, California

Enclosure: (A) Three (3) copies CVAC Technical Memo A-Misc.-10
"A Proposal for a Precision Free Flight Range for Airplanes"

1. Enclosure (A) is forwarded for Air Force evaluation and comment. It is believed that this information may be of value in planning future test facilities.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION

R. R. Hoover
Project Engineer

cc: AFPR (4)
LADFO

2733

Incl. (F) to 11-2-2327

[redacted]

CONVAIR
A Division of General Dynamics Corporation
(San Diego)

UNCLASSIFIED

Airplane Serial No. 53-1781

Convair Flight Test and Maintenance Period	-	21 Months
Portion Attributable to New Task	-	0 Month
Portion Attributable to Overrun	-	1 Month

(a) Assignment:

(1) To be utilized for armament mechanism operation and structure loads, captive missile loads, missile launching, 2.75 and 2.0-inch rocket firing, and ATM development for negative G flight condition. After completion of this program and modification of the airplane, the airplane is to be utilized for Rocket Type MB-1 tethered ground firing and airborne firing as well as stability flight tests.

Convair Flight Test and Maintenance Months 20 Months

(2) Instrumentation installed by Convair includes initially a 12-instrument photo panel, two 26-channel oscillographs, dual RF channels of telemetry, four stadiametric cameras, wire recorder, flight safety panel, three strike cameras, time correlation system, and associated wiring, source instruments and mounting structures.

For the latter phase of testing, the following is installed by Convair: one 12-instrument photo panel, two 26-channel oscillographs, dual RF channels of telemetry, four cameras, system "D" amplifiers, wire recorder, flight safety panel, and "S" Band radar system and the required wiring source instruments and support structure.

(b) Overrun:

One flight test month added task (for a net total of 21 months) is covered by this proposal due to the delay in performance of the flight test assigned because of non-availability of MB-1 rockets to support the previous schedule.

(c) Additional Instrumentation:

Additional instrumentation required for the item a. program includes:

Sound pressure system was added to the instrumentation configuration of the airplane.

Classification Changed to:
UNCLASSIFIED
Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.
Reclassified by: L. G. Avitt Dept. 130-1 Date 4-4-67

UNCLASSIFIED

[redacted]

Airplane Serial No. 53-1787

Convair Flight Test and Maintenance Period	-	19 Months
Portion Attributable to New Task	-	0 Months
Portion Attributable to Overrun	-	0 Months
Portion Transferred to Other Aircraft	-	3 Months

(a) Assignment:

(1) To be utilized for qualitative evaluation of airplane systems and airspeed calibrations, Phase I performance and preliminary flight characteristics, Phase II, roll rate tests, high speed vibration and buffet investigations and other Phase III tests. Following a two month test period for the F-106A program (not a part of this proposal), the airplane will be utilized for flight test techniques development as pertaining to the Convair F-106 interceptor programs.

Convair Flight Test and Maintenance Period 22 Months

(2) Instrumentation installed by Convair includes voice wire recorder, one 28-instrument photo panel, flight safety panel, one 14-channel tape recorder, one dual RF telemetry and the required end instruments, wiring and mounting structures.

(b) Transferred Tasks:

The task for this airplane is the same as outlined above except that the Convair flight test and maintenance period has been reduced three (3) months (for a net total of 19 months) due to the transfer of stability derivitives testing to airplane S/N 54-1398.

(c) Additional Instrumentation:

Additional instrumentation required for the item a. program includes two 26-channel oscillographs.

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Airplane Serial No. 53-1788

Convair Flight Test and Maintenance Period	-	26½ Months
Portion Attributable to New Task	-	0 Months
Portion Attributable to Overrun	-	3½ Months

(a) Assignment:

(1) To be utilized for functional development of armaments including mechanisms, dynamic response to armament, firings of missiles and 2.75 and 2.0 inch rockets, fire control jump angle and captive missile loads, and miscellaneous flight testing and photo test support activity.

Convair Flight Test and Maintenance Period 23 Months

(2) Instrumentation installed by Convair includes a voice wire recorder, one 28-instrument photo panel, one 26-channel oscillograph, time correlation system, flight safety panel, smoke generating system, armament recording cameras, strike cameras, and the required mounting structure, wiring and instruments.

(b) Overrun:

The task assignment covered by this proposal includes the above task in its entirety with the following additions:

Convair flight test and maintenance period is increased 3½ months (for a total of 26½ months) due to the requirement for a photo test support aircraft at HAFB.

(c) Additional Instrumentation

Additional instrumentation required for the item a. program includes two 27-channel oscillographs installed.

Airplane Serial No. 53-1793

Convair Flight Test and Maintenance Period - 15 Months
Portion Attributable to New Task - 0 Months
Portion Attributable to Overrun - 3 Months

(a) Assignment:

(1) To be utilized for armament hardware performance airplane stability and response to armament operation, missile firing, rocket separation, trajectory and dispersion with 24 rocket doors and 36 rocket doors, armament bay air loads and corrosion effects.

Convair Flight Test and Maintenance Period 12 Months

(2) Instrumentation installed by Convair include a voice wire recorder, one 28-instrument photo panel, flight safety panel, two 26-channel oscillograph system, smoke generating equipment, time correlation system, strike and armament recording cameras and required source instruments, mounting structure and wiring.

(b) Overrun:

(1) Two months added task due to ECP 1165 - "Conversion from 2.00 inch to 2.75 inch rockets", the delay resulting from non-availability of parts for implementing ECP 1165 and vibration and flutter testing of missile bay doors with debris deflectors installed.

(2) Extension of the missile firing program for one month.

Airplane Serial No. 53-1794

Convair Flight Test and Maintenance Period - 18 Months
Portion Attributable to New Task - 0 Month
Portion Attributable to Overrun - 2 Month

(a) Assignment:

(1) To be utilized for structural integrity demonstration and flight loads and air loads determination performed in two phases covering approximately 58 test points.

Convair Flight Test and Maintenance Period 16 Months

(2) Instrumentation installed by Convair includes an eighteen-instrument photo panel, three 14-channel magnetic tape recorder, a dual RF telemetry, voice wire recorder, a wing bending camera and the required source instruments, wiring and mounting structures.

The instrumentation system to be utilized for the measurement of air loads is designed to provide flat response (within 5%) in the measurement of pressure fluctuations occuring at rates up to 40 cps. A manifolding system serving all pressure gages will be employed in order to provide increased resolution and to insure accurate calibration of the entire system.

The capability of the recording systems will be such that the maximum required number of correlated data functions (approximately 250) can be simultaneously and continuously recorded.

(b) Overrun:

One month added task for repair of damage sustained by the airplane and its instrumentation in a landing incident.

One month extension required to complete the original task, item (1).

This proposal also covers modification of the airplane by Experimental Factory for changes found necessary as a result of static article testing conducted at WADC by the Air Force.

Airplane Serial No. 53-1791

Convair Flight Test and Maintenance Period	-	19 Months
Portion Attributable to New Task	-	0 Month
Portion Attributable to Overrun	-	4 Months
Portion Transferred from Other Aircraft	-	1 Month

(a) Assignment:

(1) To be utilized for Phase I, III, IV tests, flutter and vibration tests (particularly in regards to the high speed vibration investigation), external fuel tanks, spin and stall tests and systems development.

Convair Flight Test and Maintenance Period 14 Months

(2) Instrumentation installed by Convair includes wing vibrator equipment, voice wire recorder, one 28-instrument photo panel, one 84-channel tape recorder and the required wiring, and instruments and mounting structure.

The task for which this airplane is assigned includes all of the above testing with the following changes:

(1) Spin and stall tests were cancelled and replaced by Air Force evaluation of the external fuel tanks.

(2) Controls system development testing has been added.

(b) Overrun:

Repair of damage sustained by the airplane in a landing incident which required one-half month.

Extension of the Convair external fuel tanks testing and addition of controls system development testing for 3½ months.

(c) Task Transferred from Other Aircraft

Control system development testing transferred from airplane S/N 53-1798 (1 month).

(d) Additional Instrumentation

Additional instrumentation required for the item a. program includes:
(1) One 50-channel oscillograph.
(2) One Dual telemetry system.
(3) Two armament bay viewing cameras.

Airplane Serial No. 53-1797

Convair Flight Test and Maintenance Period	-	10½ Months
Portion Attributable to New Task	-	0 Months
Portion Attributable to Underrun	-	½ Month

(a) Assignment:

(1) To be utilized to demonstrate the Rocket Type MB-1 extension armament system for the F-102A airplane.

Convair Flight Test and Maintenance Period 11 Months

(2) Instrumentation installed by Convair includes one photo panel, four oscillographs, a voice wire recorder, strike camera, forward and side viewing cameras, kinescope and range timing equipment, system D amplifier for fast response and the required source instruments, wiring and mounting structure. After XMG-14 fire control system has been installed buffer amplifiers and a radar scope camera is to be installed.

(b) Underrun:

One-half month is deleted from the flight test program and will be proposed under ECP 1008R1.

(c) Two additional months of airplane modification for XMG-14 installation were required and are noted for record purposes but are not a part of this proposal - reference ECP 1008.

Airplane Serial No. 53-1798

Convair Flight Test and Maintenance Period - 13 Months
Portion Attributable to New Task - 0 Months
Portion attributable to Overrun - 0 Months
Portion Transferred to Other Aircraft - 1 Month

(a) Assignment:

(1) To be utilized for antenna test (including tests with external fuel tanks, installed) a control system and component development and specification compliance tests.

Convair Flight Test and Maintenance Period 14 Months

(2) Instrumentation installed by Convair for antenna tests include radio transmitter-receiver, amplifier, signal generators, power supply, indicators and wiring, and for control systems test include a photo panel, dual RF telemetry system, three oscillographs and required source instruments, wiring, and mounting structure.

(b) Portion Transferred to Other Aircraft:

The task for which this airplane is assigned includes all of the above testing except that part of the controls systems testing was transferred to airplane Serial No. 53-1791.

Flight test months assigned to this airplane are reduced to a net total of 13 months due to a transfer of one month testing to airplane S/N 53-1791.

Note: Preparation for delivery of this airplane to the Air Force for bailment to North American Aviation following completion of the Convair test program is not included in this proposal and is the subject of separate correspondence.

Airplane Serial No. 53-1799

Convair Flight Test and Maintenance Period	-	11 Months
Portion Attributable to New Task	-	0 Month
Portion Attributable to Overrun	-	2 Months

(a) Assignment:

(1) To be utilized for Weapons System Demonstration which consists of checkout and calibration of weapon system and firing of instrumented missiles against parachute and drone targets. A final demonstration flight will be a salvo firing of tactical missiles against a drone target. Snap-up demonstration is also to be conducted.

Convair Flight Test and Maintenance Period 9 Months

(2) Instrumentation installed by Convair include a photo panel, wire recorder, two oscillographs, three strike cameras and required wiring, source instruments and mounting structures.

(b) Overrun

(1) One-half month additional time is required for Hughes Aircraft Company range gate timer testing.

(2) One and one-half additional months are required tank for completion of snap-up maneuvers and co-altitude tests.

(c) Additional Instrumentation:

Additional instrumentation required for the item (a) program includes One dual telemetry system.

Airplane Serial No. 53-1806

Convair Flight Test and Maintenance Period - 9½ Months
Portion Attributable to New Task - 0 Months
Portion Attributable to Overrun - 1½ Months

(a) Assignment:

(1) To be utilized to demonstrate the Rocket Type MB-1 ejection armament system for the F-102A airplane.

Convair Flight Test and Maintenance Period 8 Months

(2) Instrumentation installed by Convair includes a photo panel, two oscillographs, a voice wire recorder, System "D" amplifiers for fast response, kinescore equipment, range timing correlation, strike and armament bay viewing cameras and the wiring, course instruments and mounting structures required. After XMG-14 fire control system has been installed a radar scope camera and one additional oscillograph will be installed.

(b) Overrun:

One and one-half months extension of the original task required for development of the Eject-10 gun system and delay in the availability of production configuration MB-1 rockets.

Three additional months of airplane modification for installation of XMG-14 system were required and are noted here for record purposes but are not a part of this proposal - refer ECP 1008.

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Airplane Serial No. 53-1813

Convair Flight Test and Maintenance Period - 8 months
Portion Attributable to New Task - 0 month
Portion Attributable to Underrun - 0 month

(a) Assignment:

(1) To be utilized for flutter, directional stability and rolling performance tests with a gloved larger vertical fin installed and for constant speed drive negative "g" tests. The airplane is to be modified after these tests for installation of a side control stick. Following the modification, one month functional check-out by Convair, the airplane will be flight tested by USAF at Edwards Air Force Base for a four month period; during these 4 months, Convair will maintain the airplane.

Convair Flight Test and Maintenance Period 8 Months

(2) Instrumentation installed by Convair includes eleven-housed cameras, photo panel, dual RF telemetry system, wire recorder and the required source instruments, wiring and mounting structure.

(b) The task assigned to this airplane includes the above tests and the following addition:

High-low landing lights evaluation.

The flight test months assigned remain unchanged by this proposal since the one month added task for landing light evaluation was accomplished within the original time assigned and the side control stick evaluation was completed ahead of schedule.

Note: One additional modification month was required for the installation of the side control stick and is noted here for record purposes but is not a part of this proposal.

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Airplane Serial No. 54-1380

Convair Flight Test and Maintenance Period	-	11½ Months
Portion Attributable to New Task	-	0 Months
Portion Attributable to Overrun	-	3½ Months

(a) Assignment:

(1) To be utilized for power plant development.

Convair Flight Test and Maintenance Months	8 Months

(2) Instrumentation requirements for this airplane include two RF channels of FM/FM telemetry, one photo panel, and the associated and instruments, wiring, junction boxes and mounting structures.

(b) Overrun:

3½ Months added task for emergency landing gear extension tests and development testing of GFAE equipment. (Sundstrand Constant Speed Drive)

(c) Additional Instrumentation.

Additional instrumentation required for the item a. program includes:

(1) One voice recorder.

(2) One tape recorder.

(3) One 26-channel oscillograph.

Airplane Serial No. 54-1388

Convair Flight Test and Maintenance Period	10½ Months
Portion Attributable to New Task	0 Months
Portion Attributable to Overrun	4½ Months

(a) Assignment:

(1) To be utilized for specification compliance and electronic environment testing for the production extended ramp and MG-10 installation as follows:

(a) Heating and Ventilating, including pilot comfort, electronic compartment cooling and vibration, windshield and canopy defog.

(b) All weather operation including radome, windshield, and engine inlet scoop anti-ice and windshield rain clearing.

Convair Flight Test and Maintenance Months 6 Months

(2) Instrumentation requirements for this airplane include one photo panel and recording camera, two 26-channel oscillographs, two RF channels of FM/FM telemetry and the associated end instruments, wiring junction boxes and mounting structures.

(b) Overrun:

The task assignment for this airplane includes the above tasks and 4½ months overrun for the following additions:

(1) Windshield defog development testing. Ref. ECP 1204.

(2) F102 hydraulic systems tests transferred from airplane 54-1401.

Airplane Serial No. 54-1385 Continued

(c) Additional Instrumentation:

Change to instrumentation requirements as shown in (a)2 are as follows:

(1) One voice recorder added.

(2) One 26-channel oscillograph deleted.

Airplane Serial No. 54-1390

Convair Flight Test and Maintenance Period	-	12 Months
Portion Attributable to New Task	-	1 1/2 Months
Portion Attributable to Overrun	-	1/2 Month

(a) Assignment:

(1) To be utilized for flutter stability, control and performance testing to evaluate the big tail installation and tactical improvement testing. The tactical improvement testing will be conducted to evaluate weight saving and such configuration changes as result from tactical operations by the USAF.

Convair Flight Test and Maintenance Months 10 Months

(2) Instrumentation requirements for this airplane include one photo panel and recording camera, two RF channels of FM/FM telemetry and the associated end instruments, wiring, junction boxes and mounting structures.

The task assignment for this airplane includes the above task and the following redefinition of task:

Provide production support to:

(a) Gain lead time over F102A airplane Serial No. 56-1137 to insure the installation and operation of the production MG-10 configuration.

(b) Confirm or modify MG-10 installation procedures, flight check-out procedures and evaluate functional performance of aircraft provisions and MG-10.

(c) Conduct specification conformance verification to further resolve ring-out and production flight test problems.

(d) Provide "Unsatisfactory Report" support for any problem areas, reported by Air Force, Hughes Aircraft Corporation and Convair testing.

(b) New Task:

One and one-half months added task to accomplish main landing gear critical stress determination.

(c) Overrun:

One-half month extension for accomplishment of original task as set forth in (a) above.

(d) Additional Instrumentation:

Additional instrumentation requirements for the item a. program include:

(1) Two 26-channel oscillographs

(2) One voice recorder

Airplane Serial No. 54-1398

Convair Flight Test and Maintenance Period - 8 Months
Portion Attributable to New Task - 0 Months
Portion Attributable to Overrun - 4 Months

(a) Assignment:

(1) To be utilized for air loads--unsymmetrical--on the big tail.

Convair Flight Test and Maintenance Months 4 Months

(2) Instrumentation requirements for this airplane include one photopanel and recording camera, one 26-channel oscillograph, two RF channels of FM/FM telemetry and the associated and instruments, wiring, junction boxes and mounting structures.

(b) Overrun:

Four months extension of the original program required to complete the flight loads instrumentation chekedown and asymetrical loads demonstration.

(c) Additional Instrumentation:

(1) One tape recorder added.

(2) One voice recorder added.

(3) One 26-channel oscillograph deleted.

Airplane Serial No. 54-1401

Convair Flight Test and Maintenance Period	-	13 months
Portion Attributable to New Task	-	1 months
Portion Attributable to Overrun	-	0 months

(a) Assignment:

(1) To be utilized for spin testing, to demonstrate specification compliance with the big tail and extended ramp configuration and to demonstrate the production airplane configuration operation to design limits including automatic flight control, Data Link and MG-10 Demonstration and hydraulic and electrical systems.

The task assignment for production airplane configuration tests to design limits is further defined as follows:

(a) Flight testing to develop snap-up limit capabilities and to explore overall system limit capabilities for F102A/MG-10 system compatibility with SAGE.

1. Data is required to define flight envelopes for integration into SAGE concept.

2. Reduce the time lag in supplying the required Convair-SAGE data to minimize the amount of "Degradation of F102A/MG-10 System Capabilities".

3. Insure complete system verification of MG-10 and F102A aircraft provisions for the F102A Serial No. 56-1137 configuration.

4. Back-up support of airplane Serial No. 54-1390 and Hughes Aircraft Co. F102A airplane engaged in similar tasks within the test period assigned for 54-1401.

Convair Flight Test and Maintenance Months — 12 months

(2) Instrumentation requirements for this airplane include one photo-panel and recording camera, one 26-channel oscillograph, two RF channels of FM/FM telemetry and the associated end instruments, wiring, junction boxes and mounting structures.

(b) New Task:

One additional month required for the spin test program due to Air Force changes in, and check out of, spin recovery equipment.

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Trainer Airplane Serial No. 54-1351

Convair Flight Test and Maintenance Period	-	12 months
Portion Attributable to New Task	-	0 month
Portion Attributable to Overrun	-	0 months
Portion Attributable to Underrun		1 months

(a) Assignment:
   (1) To be used for investigation of buffet, performance evaluation, and Phase I, Phase II and Phase III testing, including anti-icing, rain clearing and production systems tests.

Convair Flight Test and Maintenance Period 13 Months

   (2) Instrumentation installed by Convair includes a 24-instrument photo-panel, flight safety panel, tape recorder, wire recorder, time correlation system, flight programmer, and the required wiring, source instruments and mounting structure.

(b) Underrun:
   (1) The original production systems tests and all weather tests were reduced by one month.

(c) This proposal also covers:
   (1) Two months additional modification time required by Convair test base to complete the specification conformance changes.

   (2) Two months added task for rain clearing ground tests conducted at San Diego.

(d) Additional Instrumentation:

Additional instrumentation requirements for the item a. program includes:

(1) One fin camera.
(2) Two 26-Channel oscillographs.

Trainer Airplane Serial No. 54-1353

Convair Flight Test and Maintenance Period		
Portion Attributable to New Task	-	4 months
Portion Attributable to Overrun	-	0 months
Portion to other Aircraft	-	0 months
	-	1 month

(a) Assignment:

(1) To be utilized for Phase I buffet evaluation with modified ("cut-down") canopy and Phase III and roll coupling tests with original canopy and gloved tail, and external fuel tanks tests.

Convair Flight Test and Maintenance Period 5 Months

(2) Instrumentation installed by Convair includes one RF telemetry system, a 24-instrument photo-panel, 26-channel oscillograph, wiring, time correlation system, source instruments, and mounting structures.

Following the above Convair flight test program and instrumentation modification the airplane will be delivered to the Air Force for Phase IV tests by AFFTC. Instrumentation will be revised to delete the telemetry system and replace installed source instruments with new source instruments.

(b) Transferred Task:

Flight test period assigned for this airplane is reduced by one month, for a total of four months, as a result of the transfer of the external fuel tanks tests to airplane 54-1361

(c) This proposal also covers:

(1) Two months additional modification time required by Convair test base to complete the conversion to Phase IV instrumentation and preparation for Air Force acceptance.

(2) One month additional modification by Palmdale Production departments for completion of the gloved tail modification.

(d) Additional Instrumentation:

Additional instrumentation required for the item a. and b. program includes:

(1) One voice recorder.

Trainer Airplane Serial No. 54-1354

Convair Flight Test and Maintenance Period	-	10 Months
Portion Attributable to New Task	-	1 Month
Portion Attributable to Overrun	-	0 Month

(a) Assignment:

(1) To be utilized for armament testing, including MG-10T vibration environment tests and effect of armament on MG-10T system.

Convair Flight Test and Maintenance Period 9 Months

(2) Instrumentation installed by Convair includes a 24-instrument photo panel, 50-channel oscillograph, five armament cameras, wire recorder, wiring, source instruments, and mounting structures.

(b) New Task:

The flight test period assigned to this airplane is increased one month, for a total of ten months, required for modification of the GFAE constant speed drive system necessary for the original flight test program.

(c) Overrun:

The flight test assignment is extended one month to complete the original task as described in paragraph (a) above.

Trainer Airplane Serial No. 54-1361

Convair Flight Test and Maintenance Period = 6½ Months
Portion Attributable to New Task = 0 Month
Portion Attributable to Overrun = 1½ Month

(a) Assignment:

(1) To be utilized for structural integrity demonstration and heating and ventilating tests.

Convair Flight Test and Maintenance Period 5 Months

(2) Instrumentation requirements for this airplane include a dual RF telemetry system, an oscillograph system, a wire recorder, pilot's panel instruments and the associated end instruments, wiring, and mounting structures.

(b) **Overrun:**

The flight test period assigned to this airplane is increased **1½ months** for a total of 6½ months, required to complete the shake down structural demonstration instrumentation and **external fuel tank tests.**

(c) This proposal also covers the following additional modification tasks:

(1) One and one-half months modification by Convair test base department following completion of the assigned test program for installation of external fuel tanks and Phase IV instrumentation.

(2) Instrumentation for the Air Force Phase IV Tests will require removal of the telemetry system and revision of the end instruments and associated wiring and provisions.

(d) Additional Instrumentation:

Additional instrumentation required for the item a. program includes:

One 24-instrument Photo Panel.

Trainer Airplane Serial No. 54-1362

Convair Flight Test and Maintenance Period	-	7 Months
Portion Attributable to New Task	-	0 Months
Portion Attributable to Underrun	-	1 Month

a. Assignment:

(1) To be utilized for stability and control tests, spin tests, performance evaluation and Phase III tests of the production airplane configuration.

Convair Flight Test and Maintenance Period 8 Months

(2) Instrumentation requirements for this airplane include a dual RF telemetry system, an oscillograph system, a wire recorder, pilots panel instruments and the associated end instruments, wiring, and mounting structures.

b. Underrun:

The flight test assignment for this airplane has been reduced from a total of 8 months to 7 months as a result of task re-alignment.

c. Additional Instrumentation:

Instrumentation requirements for the Item a. program are changed as follows:

Delete one oscillograph system.

The currently authorized Convair Flight Test and maintenance period for the following six (6) aircraft remain unchanged by this proposal and are included herein for record purposes only:

Airplane Serial No. 52-7994

Convair Flight Test and Maintenance Period - 1 Month
Portion Attributable to New Task - 0 Month
Portion Attributable to Overrun - 0 Month

Assignment:

Test assignment was to demonstrate the evaluate the basic flying and operating qualities of the YF-102 airplane. This airplane crashed after one month of testing and the test assignments were transferred to Airplane Serial No. 52-7995.

Instrumentation:

Instrumentation installed by Convair includes a 40-instrument photo panel, four 26-channel oscillographs, dual RF channels of telemetry, wire recorder, flight safety panel, flutter vibration equipment, flight programmer, time correlation system, and required wiring, source instruments and mounting structures.

Airplane Serial No. 52-7995

Convair Flight Test and Maintenance Period - 18 Months
Portion Attributable to New Task - 0 Month
Portion Attributable to Overrun - 0 Month

Assignment:

To be utilized for safety of flight tests, initial power plant performance, cooling, pitch and yaw rate damp systems, minimum flutter investigation, phase II, phase III, development testing for the F-102A airplane and to demonstrate and evaluate the basic flying and operating qualities of the YF-102 airplane. Engine cooling and drag modifications and testing. Three tail cone configurations and an extended nose were tested. Also, cambered leading edge and reflex tips were tested.

Instrumentation:

Instrumentation installed by Convair includes a 40-instrument photo panel, four 26-channel oscillographs, dual RF channels of telemetry, wire recorder, flight safety panel, flight programmer, and flutter vibration equipment, time correlation system, and the required wiring, mounting structure and source instruments.

Airplane Serial No. 53-1779

Convair Flight Test and Maintenance Period - 4 Months
Portion Attributable to New Task - 0 Months
Portion Attributable to Overrun - 0 Months

Assignment:
To be utilized for dynamic response and stability and control tests, as related to the pitch and yaw damper systems.

Instrumentation:
Instrumentation installed by Convair includes three 18-channel oscillographs, wire recorder, flight safety panel, and provisions for mounting three strike cameras. Time correlation system, wiring, source instruments, and mounting structures are also included.

Airplane Serial No. 53-1780

Convair Flight Test and Maintenance Period - 17 months
Portion Attributable to New Task - 0 months
Portion Attributable to Overrun - 0 months

Assignment:
To be utilized for testing and refinement of the flight control and automatic flight control systems.

Instrumentation:
Instrumentation installed by Convair includes three 26-channel oscillographs, dual RF channels of telemetry, 30-instrument photo panel, flight safety panel, flight programmer, time correlation system, wire recorder, and associated wiring, mounting structures and source instruments.

Airplane Serial No. 53-1782

Convair Flight Test and Maintenance Period = 1 month
Portion Attributable to New Task = 0 month
Portion Attributable to Overrun = 0 month

Assignment:
To be utilized to demonstrate the functional characteristics of the production antenna installations and to demonstrate operation of the emergency fuel system, ring type dacron drag chute and the electrical system power controller.

Instrumentation:
Instrumentation installed by Convair includes a flight safety panel, three 18-channel oscillographs, wire recorder, three strike cameras, time correlation system, and the required source instruments, wiring and mounting structure.

NOTE: The flight test assignment for this airplane has been completed and the airplane has been finally accepted.

Airplane Serial No. 53-1783

Convair Flight Test and Maintenance Period = 10 months
Portion Attributable to New Task = 0 months
Portion Attributable to Overrun = 0 months

Assignment:
To be utilized for qualitative testing of dry air anti-icing, defogging and cabin ventilation and pressurization, completion of communications and navigation systems tests transferred from airplane S/N 53-1782, and Air Force evaluation of the airplane in various icing conditions.

Instrumentation:
Instrumentation installed by Convair include a voice wire recorder, one 28-instrument photo panel, three 18-channel oscillographs, one dual RF telemetering system, NACA icing rate meter, relative wind transducer, mounts for four 16 MM camera, glide path antenna and the required and instruments, wiring and mounting structure.

C O N V A I R

SAN DIEGO

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ZC-8-142-19
1 June 1958

FINAL REPORT

YF-TF AND F-[illegible] ACCELERATED DEVELOPMENT FLIGHT TEST PROGRAM ACTIVITIES

OCTOBER 195[illegible] THROUGH MAY 1958

PREPARED BY ______________ GROUP Engineering Flight Test Planning
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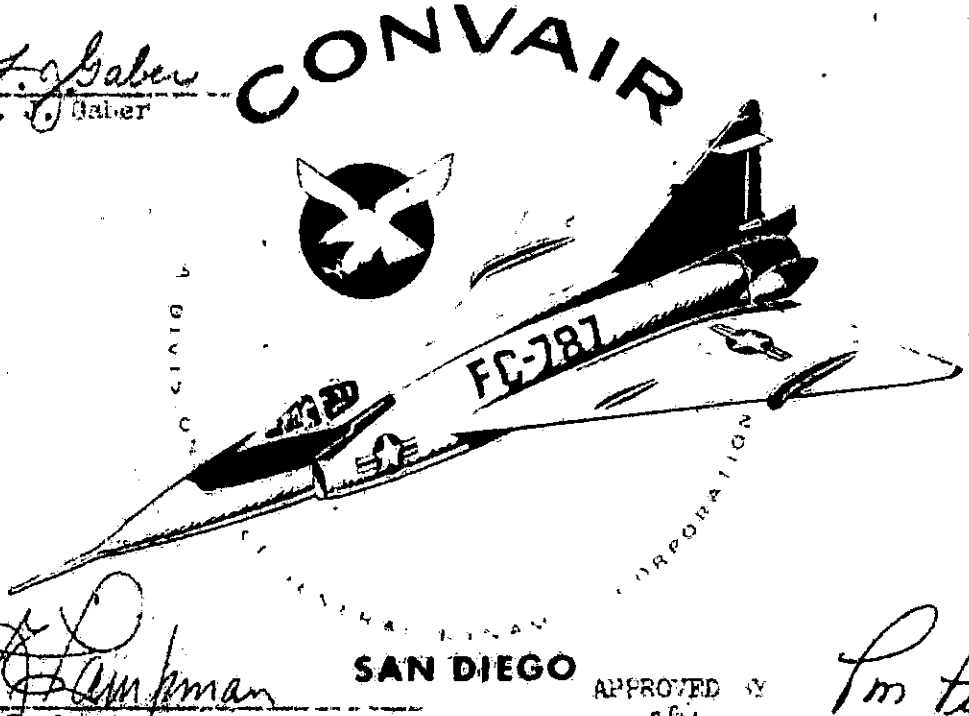
CONVAIR
FC-787
DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

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NOTE: Appendix B, which was a compilation of published flight test reports on test items from programs that had not been fully reported, has been integrated into Appendix C.

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INTRODUCTION

This report comprises a compendium of histories of F-102 series aircraft covering 54 months of a Convair Accelerated Development Flight Test Program. Starting with the initial flight of the YF-102 in October 1953 and ending with the product improvement flight of a TF-102A in May 1958, this report summarizes prominent flight test achievements by individual aircraft.

For a more detailed report on individual aircraft contributions to the development program, reference is made to the weekly Operations Report (ZC-8-512) and the reports of test results covered in Appendix C.

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MAJOR TEST PHASES ACCOMPLISHED

Evaluated following configurations:

YF-102 - Symmetrical airfoil-constant diameter fuselage
YF-102A- Area rule fuselage - cambered wing-small tail
F-102A - Production aircraft (both Case X and XX camber wing and large tail)
TF-102A- Two-place side-by-side cockpit.

Flutter and vibration clearance to 655 knots and Mach 1.5 achieved
Developed "yellow canary" tail fairing
Inlet duct buzz problem investigated and corrected
TF-102A buffet reduced with vortex generators
Control system developed to satisfactory configuration
External fuel tank system demonstrated
Other aircraft systems and components developed to satisfactory configurations
Roll coupling characteristics evaluated and corrected
Critical air load distribution determined
Aircraft structure demonstrated satisfactorily
Spin characteristics evaluated and demonstrated satisfactorily
Armament firing demonstrated to following independent maximum conditions:

	ALTITUDE (FT)	MACH NUMBER	CALIBRATED AIRSPEED (KTS)
GAR-1 missile	52,140	1.42	356.5
2-inch rocket	53,040	1.08	254.5
2.75-inch rocket	53,520	1.44	351.0

Weapon system demonstrated through Joint Program with Hughes Aircraft Company
MB-1 rocket successfully fired by both rail and ejection systems
Armament firing snap-up techniques developed.

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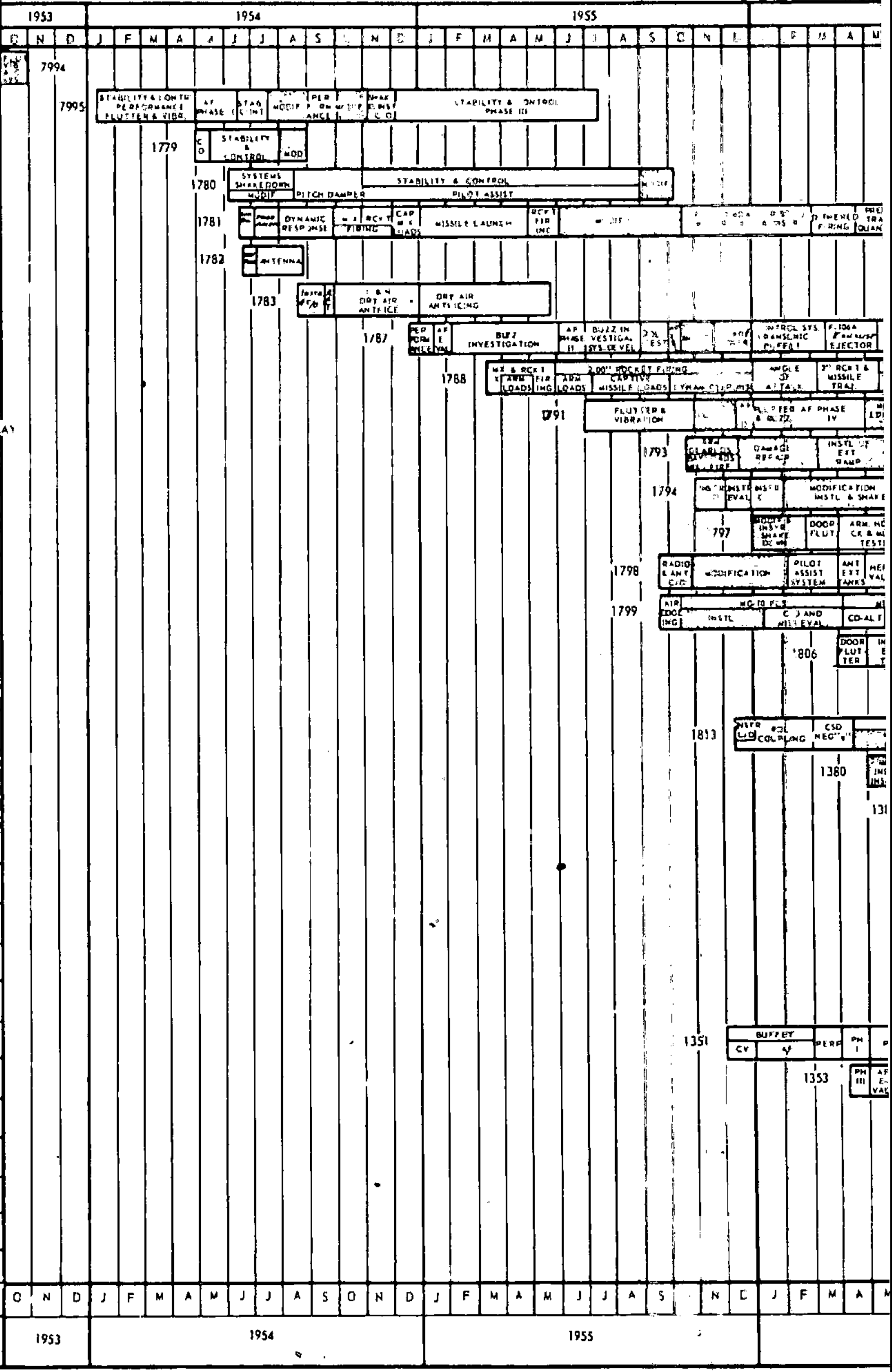
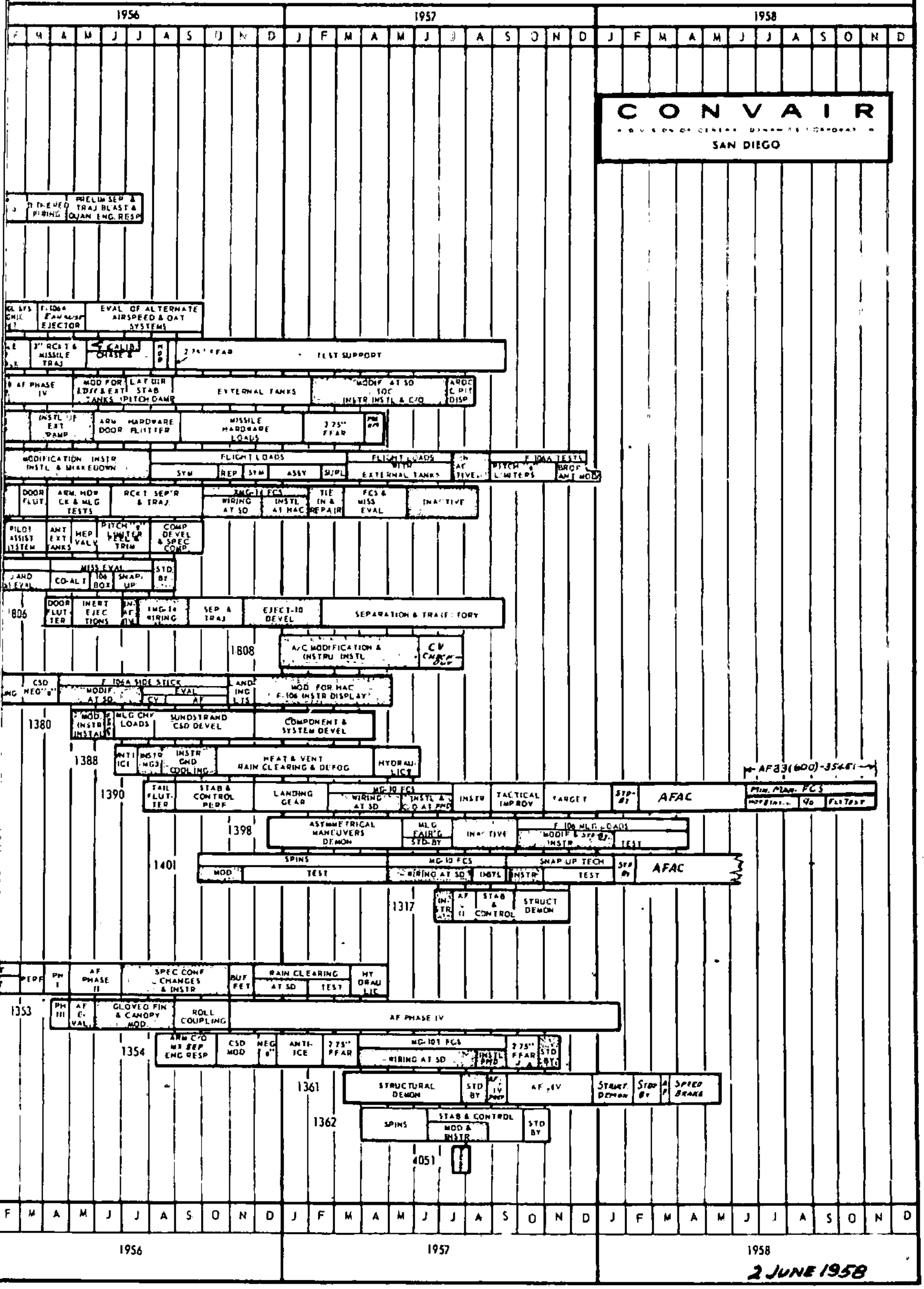
CONVAIR JF- & JTF-102A FLIG

A C S N	Mod B-	TASK ASSIGNMENT
1994	80	PHASE AN
1995	-80	PHASE ANI
1779	-82	STABILITY AND CONTROL
1780	-82	PHASE III CONTROLS
1781	-82	GAR MISSILE MB-1 ROCKET
1782	-82	ANTENNA
1783	-82	PHASE III COOLING
1787	-90	PHASE I, II, AND III
1788	-90	ARMAMENT TEST SUPPORT
1791	-10	PHASE I, III, & IV EXTERNAL TANKS F-106 COCKPIT DISPLAY
1793	-10	ARMAMENT DEVELOP/HI-PRESSURE PNEUM.
1794	-10	FLIGHT LOADS/F106 PITCH "G" LIMITER
1797	-10	MB-1 ROCKET (RAIL)
1798	-10	CONTROLS
1799	-10	COMPLETE SYSTEMS EVAL.
1806	-10	MB-1 ROCKET (EJECTION)
1808	-10	F-106A ARDC COCKPIT DISPLAY
1813	-10	LARGE TAIL/CSD F-106 SIDE STICK/INSTR.DISPLAY
1380	-10	POWER PLANT AND COMPONENTS DEVELOPMENT
1388	-10	ALL WEATHER HEAT AND VENT HYDRAULICS
1390	-10	STAB & CONT/TACT IMPR/MIN MAN FCS
1398	-10	STRUCTURAL DEMON F-106 MLG LOADS
1401	-10	SPINS/ SNAP-UP TECHNIQUES
1317	-10	CASE XX WING EVAL.
1351	-12	PHASE I, II, AND III
1353	-12	PHASE III AND IV
1354	-12	ARMAMENT
1361	-12	STRUCTURAL DEMON PHASE IV/SPEED BRAKES
1362	-12	SPINS STABILITY AND CONTROL
4051	-12	EXTERNAL TANKS



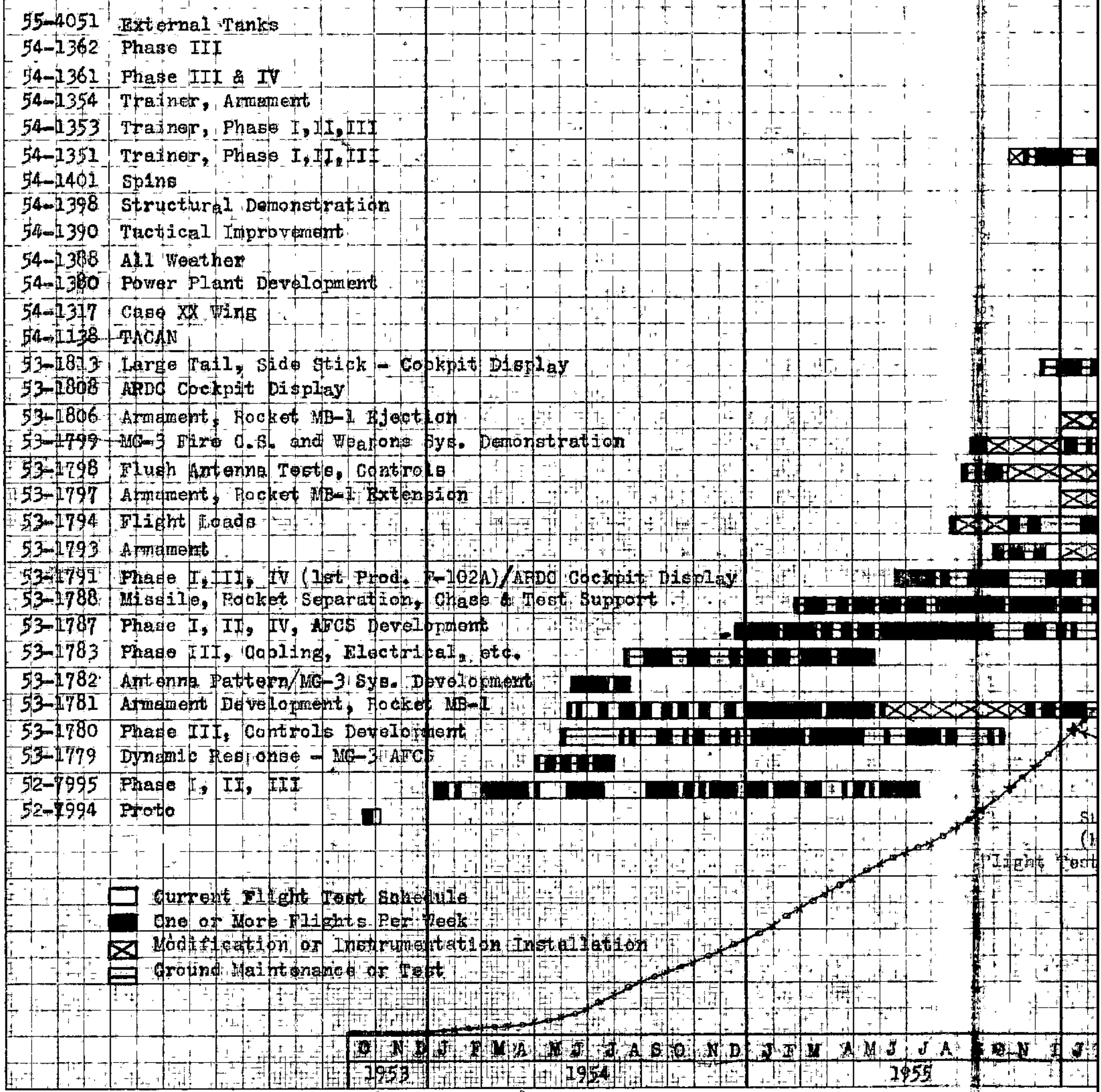
FLIGHT TEST PROGRAM



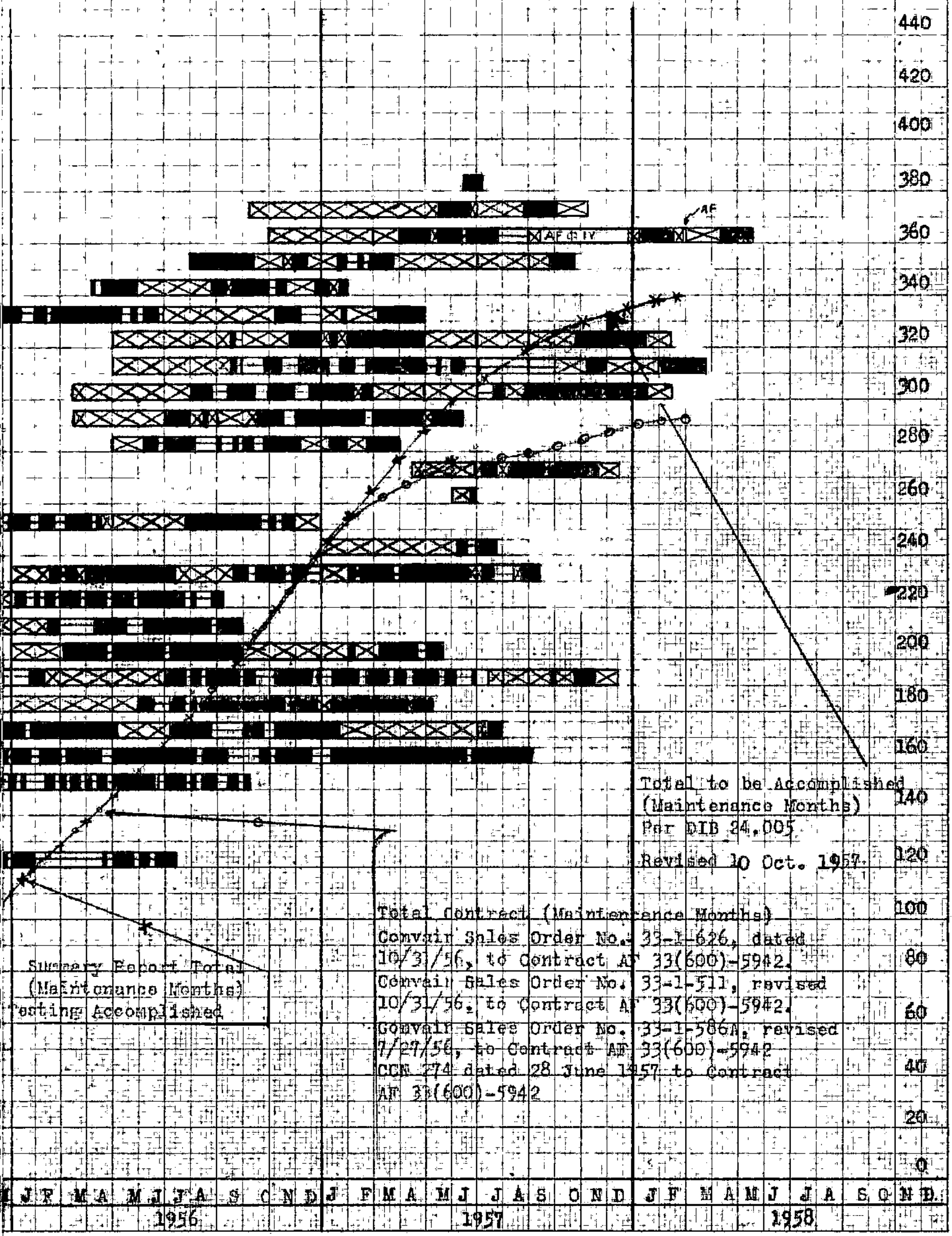
CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

2 JUNE 1958

CONVAIR F-102 AND TF-102 ACCEL
AIRCRAFT ASSIGNMENT SCHEDU



CCELERATED DEVELOPMENT PROGRAM
HEDULE AND ACTIVITY STATUS

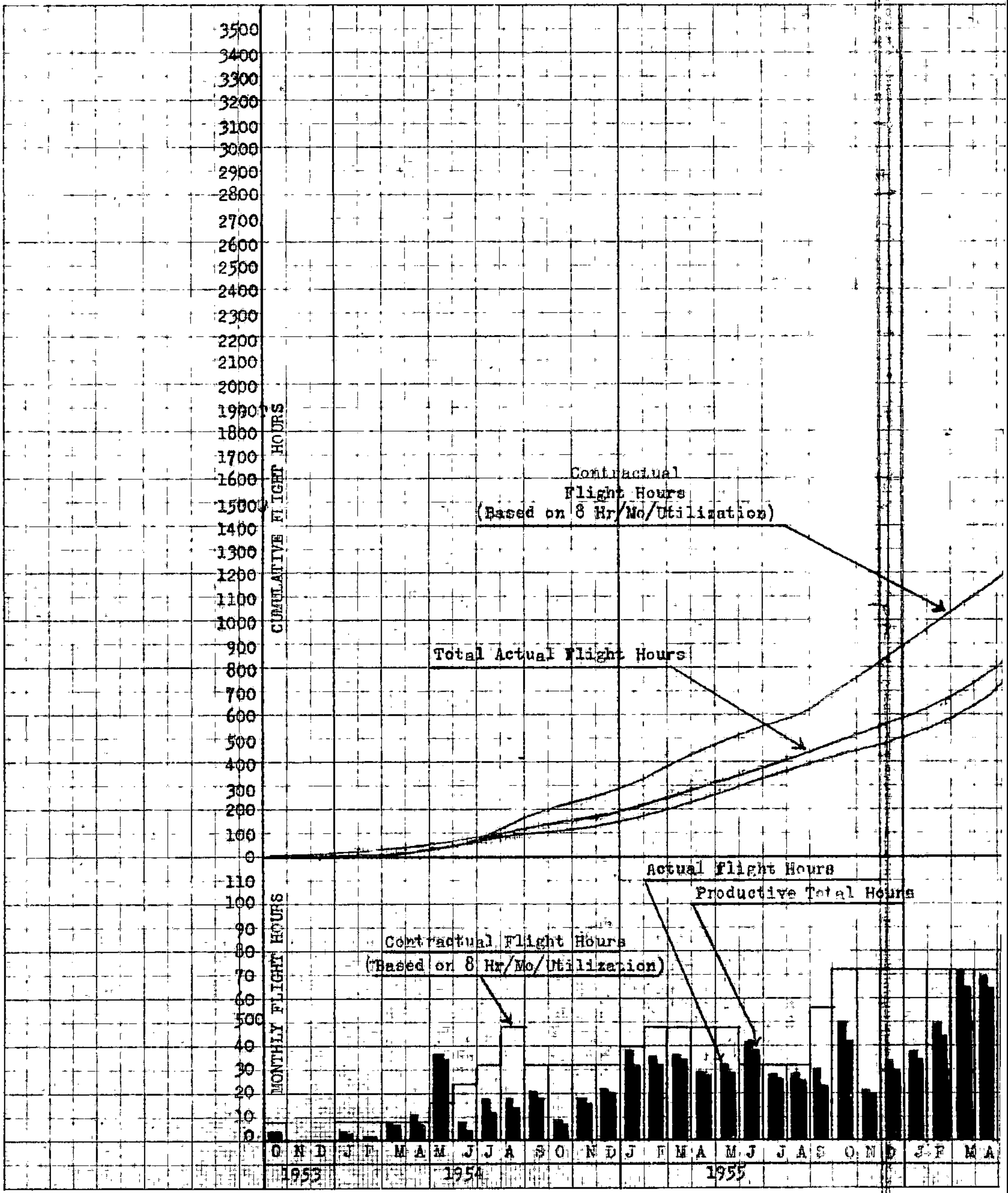


Total to be Accomplished
(Maintenance Months)
Per DIB 24.005
Revised 10 Oct. 1957

Total Contract (Maintenance Months)
Convair Sales Order No. 33-1-626, dated 10/31/56, to Contract AF 33(600)-5942.
Convair Sales Order No. 33-1-511, revised 10/31/56, to Contract AF 33(600)-5942.
Convair Sales Order No. 33-1-586A, revised 7/27/56, to Contract AF 33(600)-5942
CCN 774 dated 28 June 1957 to Contract AF 33(600)-5942

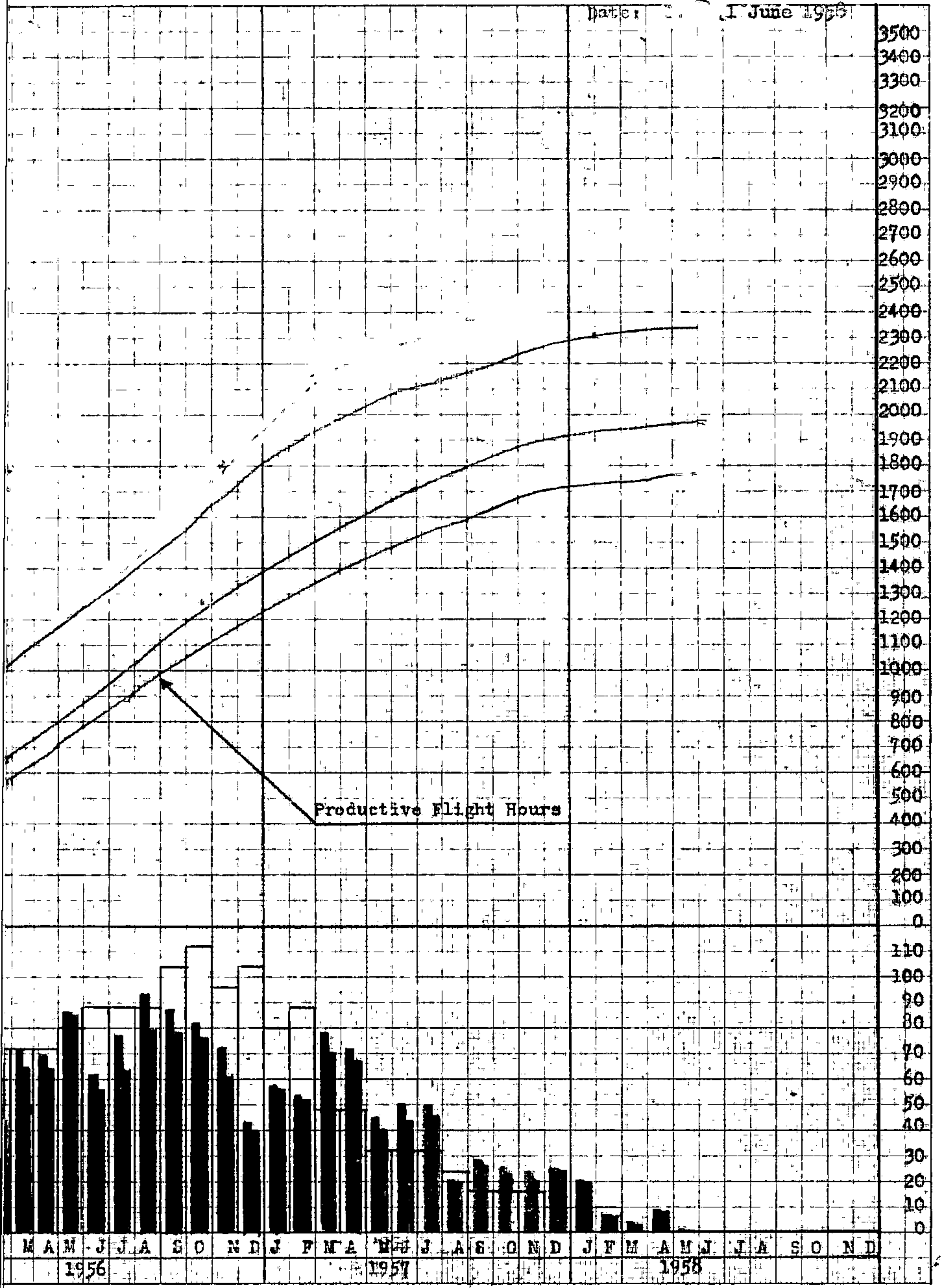
Summary Report Total
(Maintenance Months)
Testing Accomplished

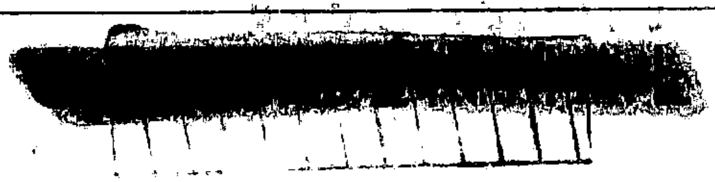
CONVAIR YF, TF & F-102 ACCELERATED
ESTIMATED (8 Hr/Mo/Util) vs ACTUAL & PRODU



ATED DEVELOPMENT PROGRAM
PRODUCTION FLIGHT HOURS FLOWN

Page: 9
Report No. ZC-8-142-19
Model: Y[illegible] & F-102A
Date: 1 June 1958





CHRONOLOGY OF SIGNIFICANT EVENTS IN THE YF-102, F-102A, AND TF-102A FLIGHT TEST DEVELOPMENT PROGRAM

24 October 1953:	Initial flight of YF-102. Airplane crashed 2 November 1953 during take-off for flight seven.
11 January 1954:	Initial flight of second YF-102.
27 January 1954:	Supersonic speed of Mach 1.06 was attained.
5 March 1954:	Flutter investigation tests were initiated at supersonic speeds using shakers to evaluate damping characteristics.
13 April 1954:	Buffeting was reduced and uncontrollable yaw oscillations in the vicinity of Mach 1.0 were eliminated by extension of tail pipe and modification of elevon to fuselage fillet.
13 April 1954:	Altitude performance and general handling characteristics at high lift coefficient were improved by addition of cambered leading edges.
2 May 1954:	Mach number of 1.29 was attained.
4 May 1954:	Initial flight of Model type 8-82 airplane.
4 May to 1 June:	Air Force Phase II tests were conducted. Airplane was flown by ten Air Force pilots and one NACA pilot.
4 June 1954:	Initial major experience with compressor stall occurred.
19 June 1954:	Initial fly-away of production aircraft from Contractors factory in San Diego.
12 July 1954:	Flutter investigation for type 8-80 and 8-82 airplane completed. Maximum Mach number 1.3 and maximum indicated airspeed 631 MPH.
22 July 1954:	Flight characteristics during opening of missile bay doors were demonstrated to have no unusual effects up to Mach number of 0.92.
29 July 1954:	Completed initial hardware development of pitch and yaw damper system, and obtained basic response data with this equipment.

[redacted]

24 August 1954: The ram air turbine emergency hydraulic system was successfully operated in flight.

25 August 1954: The first air starts were accomplished. No difficulties were encountered.

9 September 1954: A GAR-1 missile was fired at 25,000 feet pressure altitude.

11 September 1954: Second 8-80 airplane began testing with an extended nose and large fuselage elevon fillets. Subsequent testing proved predicted improvement in supersonic drag and in maximum power climb performance.

15 September 1954: A simulated landing was made using ram air turbine hydraulic pressure only. Satisfactory control was maintained down to 140 knots.

29 September 1954: A Mach number of 1.33 was attained at about 30,000 feet pressure altitude.

9 November 1954: Four 2.75" FFAR rockets were fired at 25,000 feet pressure altitude.

18 November 1954: The first night flight of an F-102 airplane was made.

19 November 1954: The first hooded instrument flight was accomplished.

2 December 1954: Flight testing of the "pilot assist" system was started.

14-17 December: Three Air Force pilots and one Hughes pilot flew evaluation flights of the double-headed control stick.

20 December 1954: The first flight of an 8-90 airplane was made at Edwards AFB.

21 December 1954: The first 8-90 airplane attained a Mach number of 1.2 in level flight at 35,000 feet pressure altitude.

29 December 1954: The first 8-90 airplane exceeded 52,000 feet pressure altitude.

21 January 1955: Highest altitude reached by YF-102A of 55,130 indicated feet.

20 February 1955: First test flight of production F-102A.

3 May 1955: First 2.00" rocket fired from F-102.

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10 May 1955: Highest altitude firing of GAR missile at 52,140 feet.

June 1955: Air Force Phase II testing was completed.

8 June 1955: Longest flight time of 1 hour, 53 minutes, made in YF-102.

23 June 1955: Highest altitude firing of rockets at 53,040 feet.

31 October 1955: First flight of trainer TF-102.

10 December 1955: First flight of enlarged vertical fin on F-102A.

22 December 1955: First MB-1 rocket jettisoned from a fixed pylon on YF-102.

4 January 1956: First Phase VI F-102A delivered to Air Force at EAFB.

18 January 1956: Design speed of Mach number 1.535 reached in F-102.

11 February 1956: First delivery of F-102A for Phase VII testing at Eglin AFB.

4 April 1956: First delivery of F-102A for AF Armament Center testing at Eglin AFB.

14 April 1956: First TF-102A flight with cut-down canopy.

1 May 1956: First tactical F-102A delivered to the Air Force.

1 May 1956: First MB-1 rocket launched from YF-102.

11 May 1956: First GAR-1 missile hit on drone (By Convair at HAFB).

14 May 1956: First MB-1 rocket, less motor, ejected from F-102.

5 June 1956: Compatibility of the F-102A, MG-3 FCS and GAR-1 missile demonstrated by destruction of a QB-17 drone at HAFB.

23 July 1956: Engine inlet scoop anti-ice system evaluation completed.

6 August 1956: Successful "Snap Up Maneuver" missile firings against a high altitude balloon target were performed at HAFB.

1 August 1956: The F-102A reached an altitude of 58,700 feet.

27 August 1956: First Big Tail TF-102A flown.

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[redacted]

10 September 1956: First missile firing from a TF-102A aircraft at HAFB.

17 September 1956: Armament hardware and bay door load tests completed.

Electronic pitch "g" limiter tests completed satisfactorily.

22 October 1956: External fuel tank flutter tests completed.

25 October 1956: TF-102A roll coupling tests completed.

29 October 1956: Aft electronics area cooling tests completed.

19 November 1956: The roll coupling tests completed on the F-102A.

28 November 1956: TF-102A buffet testing completed.

3 December 1956: Emergency fuel transfer tests and main wheel ambient pressure survey have been completed.

10 December 1956: F-102A landing light tests completed. The after-duct cheek fairing buffet investigation, radome anti-ice tests and pilot's panel vibration tests completed on the TF-102A.

12 December 1956: Successful ejected launching of a live MB-1 rocket from an F-102A.

14 December 1956: First emergency ejection of a pilot from an F-102A.

26 December 1956: Part I of 50 percent, 80 percent, and 100 percent symmetrical load factor flight loads program completed.

3 January 1957: External tank jettison tests and in-flight fuel transfer tests completed on the F-102A.

28 January 1957: Cockpit carbon monoxide tests completed.

4 February 1957: Drag chute deployment, take off and landing performance and speed brake effectiveness tests completed on the F-102A.

First spin entry and recovery of the F-102A aircraft.

11 February 1957: Pilot's instrument panel vibration tests, Stratos refrigeration package tests and cabin temperature control response tests completed on F-102A.

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[redacted]

Missile bay doors with debris deflectors installed flutter tests and 2.75-inch rocket exhaust effect on GAR-1 and GAR-2 missiles completed.

The 50 percent and 80 percent asymmetric increments of design limit loads completed on the F-102A.

18 February 1957: 100 percent asymmetric maneuver tests completed on the F-102A.

Engine inlet duct anti-icing system tests completed in the TF-102A.

26 March 1957: Structural demonstration flights started on F-102A.

2 April 1957: Airloads (with external tanks installed) testing started on F-102A.

23 April 1957: High pressure pneumatics system testing completed.

26 April 1957: Spin tests completed on F-102A.

22 May 1957: First spin entry and recovery completed on TF-102A.

19 June 1957: Spin tests completed on TF-102A.

26 June 1957: TACAN equipment evaluation initiated on F-102A.

28 June 1957: First production Case XX wing F-102A delivered to test base.

23 August 1957: Stability and control testing started on Case XX wing F-102A.

5 November 1957: Snap-up techniques evaluation initiated with F-102A.

19 November 1957: Structural demonstration of Case XX wing F-102A completed.

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F-102A FLIGHT

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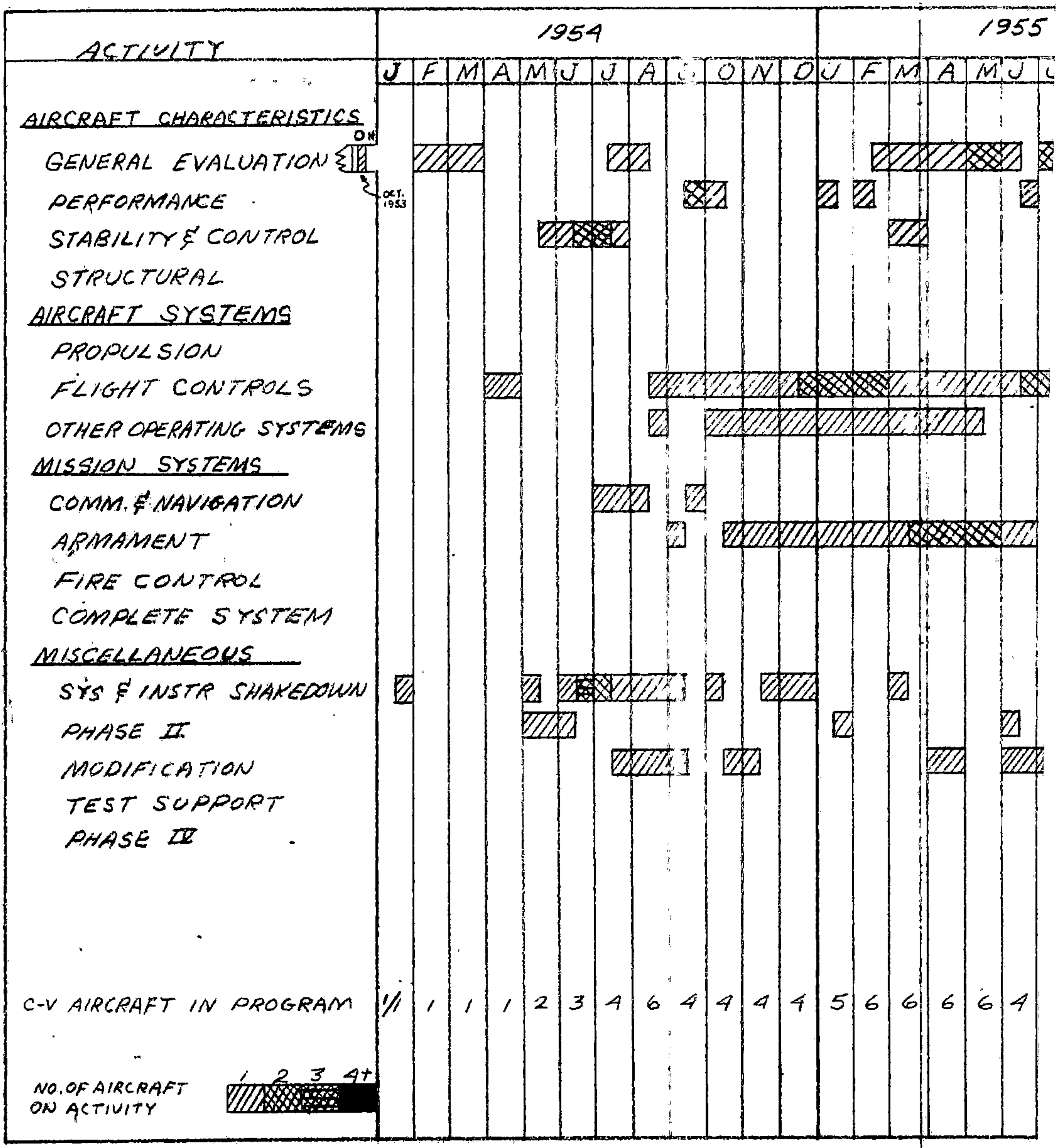

IT TEST PROGRAM ACTIVITY

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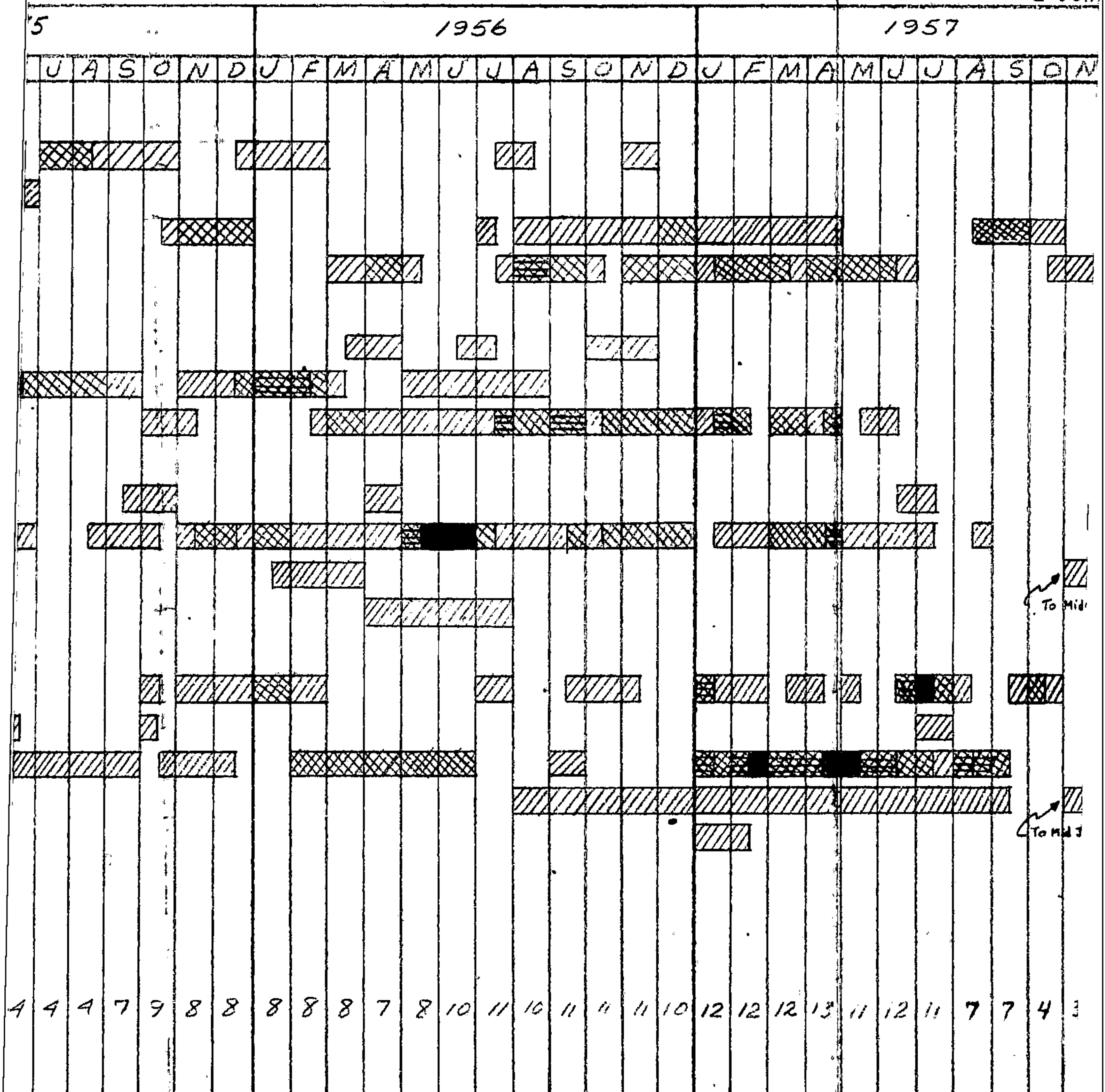
1 Jun



C-8-142-19 Page 7
1 June, 1958

				SUMMARY		
	O	N	D	No A/C	M.M.	% TOT. M.M.
				8	19.5	6.9
				3	3	1.0
▨	▨			9	23.5	8.4
	▨	▨		6	24.5	8.7
				2	4.5	1.6
				6	32.0	11.4
				14	33.5	11.9
				4	5.5	2.0
				5	47.5	17.0
		▨	▨	2	5.0	1.8
	To Mid JAN.'58			1	4	1.4
▨	▨			17	29	10.3
				3	4	1.4
				13	28	10.0
		▨	▨	2	16.	5.3
	To Mid JAN.'58			1	1.5	.5
					281.0	100.0
7	4	3	2½			

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SUMMARY OF FLIGHT TEST ACTIVITIES FOR DECEMBER 1957 THROUGH MAY 1958

During the period December 1957 through May 1958 four (4) Convair test aircraft flew a total of 68.4 hours during 77 flights. Of this total 64.7 hours (productive) were flown toward the accomplishment of the Convair Flight Test Program.

Flight Test Items of Special Interest

Aircraft S/N 54-1401 - Developed snap-up techniques for armament firing.
Aircraft S/N 54-1361 - Structural demonstration program completed; evaluated original and modified large speed brakes.

Summary of Flight Test Operations

A/C S/N	CONVAIR		OTHER AGENCY		(1) FSC & F				TOTALS	
	Test Hour	No.of Flts.	Test Hours	No.of Flts.	Test Hours	No.of Flts.	Prod. Hours	Non-Prod.	Test Hours	No. of Flts.
54-1361	4.3	5	1.0	1	4.8	6	9.2	.9	10.1	12
54-1390					17.1	19	17.1		17.1	19
54-1398	17.7	20			2.1	2	17.8	2.0	19.8	22
54-1401	21.4	24					20.6	.8	21.4	24
	43.4	49	1.0	1	24.0	27	64.7	3.7	68.4	77

(1) Ferry, Shakedown, Chase, and Familiarization

[redacted]

[redacted]

CUMULATIVE CONVAIR ENGINEERING FLIGHT TEST HOURS & FLIGHTS FROM OCTOBER 1953 THROUGH MAY 1958

A/C S/N	CONVAIR		OTHER AGENCY		(1) FSC & F		CUMULATIVE TOTALS			
	Test Hours	No. of Flts.	Test Hours	No. of Flts.	Test Hours	No. of Flts.	Prod. Hours	Non-Prod.	Hours	Flights
52-7994	1.8	6					1.8		1.8	6
52-7995	67.2	113	36.1	46	1.7	4	97.0	8.0	105.0	163
53-1779	9.9	19	.8	1	1.9	4	9.7	2.9	12.6	24
53-1780	46.0	67	5.3	6	1.3	3	50.8	1.8	52.6	76
53-1781	63.6	106			11.1	17	58.0	16.7	74.7	123
53-1782	18.4	24	.3	1	2.4	4	15.7	5.4	21.1	29
53-1783	23.5	35	59.8	57	0.7	2	73.5	10.5	84.0	94
53-1787	130.1	169	35.3	41	4.8	6	159.9	10.3	170.2	216
53-1788	114.8	170	.9	1	75.0	107	177.0	13.7	190.7	278
53-1791	70.1	102	39.6	48	1.9	3	107.5	4.1	111.6	153
53-1793	77.4	115			5.8	7	67.1	16.1	83.2	122
53-1794	55.4	74			4.2	7	55.7	3.9	59.6	81
53-1797	46.3	68			6.9	8	41.9	11.3	53.2	76
53-1798	86.1	93			7.5	10	85.1	8.5	93.6	103
53-1799	43.5	64			5.4	6	41.2	7.7	48.9	70
53-1806	60.5	86			9.0	13	52.8	16.7	69.5	99
53-1808	5.0	4	1.3	1	1.3	2	5.6	2.0	7.6	7
53-1813	49.4	64	54.9	61	3.4	4	101.6	6.1	107.7	129
54-1351	61.2	77	24.0	30	2.5	4	76.7	11.0	87.7	111
54-1353	18.2	26	14.8	18	5.8	8	34.5	4.3	38.8	52
54-1354	41.6	55			5.2	5	39.1	7.7	46.8	60
54-1361	30.9	35	1.0	1	6.7	8	34.4	4.2	38.6	44
54-1362	27.9	32			.9	1	28.3	.5	28.8	33
55-4051	3.7	4			1.3	2	5.0		5.0	6
54-1138	13.2	10			.4	1	13.0	.6	13.6	11
54-1317	19.5	25	12.0	12			30.1	1.4	31.5	37
54-1380	58.3	69					58.3		58.3	69
54-1388	72.8	86			1.8	3	66.3	8.3	74.6	89
54-1390	52.5	72	1.6	2	22.4	24	73.7	2.8	76.5	98
54-1398	55.2	75			5.8	8	55.6	5.4	61.0	83
54-1401	58.4	75			7.6	9	60.0	6.0	66.0	84
TOTAL	1482.4	2020	287.7	326	204.7	280	1776.9	197.9	1974.8	2626

(1) Ferry, Shakedown, Chase, and Familiarization

[redacted]

[redacted]

DEFINITIONS

For the purpose of this report, the following definitions apply:

1. Contract Flight Time - Based on 8 hours/maintenance month.
2. Total Flight Time - Airplane total flight time during the period the airplane is assigned to Convair for flight testing.
3. Productive Flight Time - Flight time during which data in furtherance of the development test program was obtained. Air Force Phase II and Phase IV testing is included in this time.
4. Convair Test - Flights in which the aircraft has taken off with a Convair pilot, in direct pursuit of fulfilling a portion of the scheduled test program (includes aborts).

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APPENDIX A

FINAL SUMMARY OF AIRCRAFT FLIGHT TEST ACTIVITIES IN THE CONVAIR YF-TF & F-102 ACCELERATED DEVELOPMENT TEST PROGRAM

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 52-7994

Primary Assignment

Aircraft was assigned to the Convair Flight Test Program during October 1953. The formal statement of the flight test assignment, as presented in Contract AF 33(600)-5942, as amended in accordance with the terms of negotiations conducted at the Air Materiel Command during the week of 11 June 1956, for review and re-estimate of the F-TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 26 July 1956, is quoted as follows: "Period - One Maintenance Month." "Test assignment was to demonstrate and evaluate the basic flying and operating qualities of the YF-102 airplane. This airplane crashed after one month of testing and the test assignments were transferred to Airplane Serial No. 52-7995." The program (ZC-8-101) for this aircraft was conducted at Edwards Air Force Base, and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was trucked from Convair-San Diego to Edwards Air Force Base on 5 October 1953 to enter the F-102 Flight Test Program. The initial flight was conducted on 24 October 1953 for "shakedown and familiarization". The United States Air Force "conditionally" accepted the aircraft on 28 October 1953. Phase I ground and flight tests for: low speed taxi tests, high speed taxi tests, and shakedown flight tests were conducted and completed during the ensuing period. During take-off on flight number 7, 2 November 1953, the aircraft developed a power loss and made an emergency gear-up crash landing off the runway. The aircraft was damaged beyond economical repair and was subsequently surveyed.

The balance of Phase I ground and flight testing was later transferred to YF-102A S/N 52-7995.

Chronological History

5 October 1953 - Trucked from Convair-San Diego to Edwards Air Force Base
24 October 1953 - Initial flight number 1 conducted
2 November 1953 - Emergency crash landed due to power loss during take-off.

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Page: 2 A
Date: 1 Dec 1953
Report No.: ZC-8-122
Model: YF-102

Convair Report No.:
ZC-8-101

SUMMARY OF ACTIVITY

S/N 52-7994

Primary Assignment: F-102A Prototype

FLIGHT TEST ASSIGNMENTS	1953			1954												1955									
	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O
Instrument Calibration	▨	▨	▨	▨	▨	▨	▨	▨	▨																
Flying Qual.-Structural: Flutter & Vibration	▨ ■	▨	▨	▨	▨	▨																			
Flying Qualities: Dynamic Response			▨	▨	▨	▨	▨	▨	▨																
Aircraft Systems: Fuel Tank Pressurization	▨ ■	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨							
Performance: Power Plant & Lift-Drag Characteristics		▨	▨																						
Flying Qual.: Long, Lat. & Dir. Stability & Control						▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨	▨							

Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 52-7995

Primary Assignment

Aircraft was assigned to the Convair Flight Test Program during January 1954. The formal statement of the flight test assignment, as presented in Contract AF(600)-5942, as amended in accordance with the terms of negotiations conducted at the Air Materiel Command during the week of 11 June 1956, for review and re-estimate of the F/TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 26 July 1956, is quoted as follows: "Period - 18 Maintenance Months". "To be utilized for safety of flight tests, initial power plant performance, cooling, pitch and yaw rate damp system, minimum flutter investigation, Phase II, Phase III, development testing for the F-102A Airplane and to demonstrate and evaluate the basic flying and operating qualities of the YF-102 airplane". The program (ZC-8-103) for this aircraft was conducted at Edwards A.F. Base, and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair - San Diego to Edwards A.F. Base on 10 December 1953 to enter the F-102 Flight Test Program. The U. S. Air Force conditionally accepted the aircraft on 31 December 1953. The initial flight, for flight test purposes, was conducted on 11 January 1954 for shakedown and familiarization.

The primary ground and flight testing tasks of 52-7994 were transferred to this aircraft. During the 18 months the aircraft was in the flight test program it conducted and completed ground and flight testing on the following items:

Phase I Flight Testing

Static Thrust Calibration
Brake Operation Evaluation
Engine Control Operation
Speed Brake Operation
Fuel Tank Pressurization
Engine Equipment Operation
Flutter and Vibration
Noise Level Measurement
Cockpit Noise and Vibration
Airspeed Calibration
Preliminary Flight Characteristics
Aircraft Response to High Roll Rate Man.

Preliminary Aircraft Performance
Wing Tip Structural Load
USAF Phase II Testing
Aircraft Performance
Landing Loads and Distances
Wing Tip - Cambered Not Reflexed
Engine and Structural Cooling
Drogue Chute Tests
Emergency Fuel Control System
Flight Control System Operation
Engine and Associated Equipment Phase III
Ground Test Combustion Starter

[redacted]

[redacted]

Addendums

Landing Gear Hydraulic Pressure
Dynamic Directional Stability
Prim Hydraulic System Oil Temp. w/wo Cooler
Pressure on Extended Nose
Radome Glycol Dispersion
Yellow Canary Pressures
Bleed Air Consumption Effect
Drag Effects in Region of Mach 09
Ram Air Turbine Tests
Longitudinal Sensitivity
Eng. Outer Wall & Nozzle Actuator Temps.
Compressor Stall Test
Eng. Oil Temps. w/wo Air-Oil Cooler
Aft Section Cooling
Airspeed Calibration
Relocation of "Q" Pot. Static Source
Drag Characteristics
Vibration & Acceleration Characteristics
Cable Tension Values
Buffet Charact. & Wing Fence Eval.
Wing Fence Evaluation
Adverse Yaw & Directional Charact.

This aircraft was the second experimental model and performed its early flights with uncambered leading edges and straight wing tips. These were changed to cambered leading edges with reflexed tips on 14 April 1954. On 26 June 1954 the tips were changed to full camber without reflex.

Preliminary aircraft performance and aircraft flight characteristics, including flutter and vibration tests, were completed in the early months of the flight test program. Phase III testing was conducted during the middle and latter portions of the program, and stability and control tests were continued throughout the program as a result of control system changes made to develop the Model 8-10 versions. Roll rate studies were accomplished in connection with flight characteristics and computer analysis.

The primary contributions, as a result of this aircraft's test program, to the F-102 program (the results of which were incorporated into the Model 8-10 versions), are as follows: Developed "Yellow Canary" tail fairing, Demonstrated cambered leading edges of the wings, Obtained information on reflexed and cambered wing tips, and Developed the extended nose.

The test program of this aircraft was completed on 8 July 1955, at which time the aircraft was ferried from Edwards A.F. Base to Convair - San Diego.

Chronological History:

10 December 1953 - Ferried from Convair - San Diego to Edwards A.F. Base.
31 December 1953 - Conditionally accepted by the U.S. Air Force.
11 January 1954 - Initial flight for flight test purposes.
24 January 1954 - Finally accepted by the U. S. Air Force.
1-31 March 1955 - Phase II testing - U. S. Air Force.
8 July 1955 - Completed flight test program and ferried to Convair - San Diego.
December 1955 - Assigned to Convair - Palmdale.

[redacted]

[redacted]

Page: 5A
Date: 1 Aug. 1955
Report No.: ZC-8-142
Model: YF-102

Convair Report No.:
ZC-8-103

SUMMARY OF ACTIVITY

S/N 52-7995

Primary Assignment: Phases I - II - III

FLIGHT TEST ASSIGNMENTS	1954												1955											
	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D
Flying Qual.-Structural: Flutter & Vibration	▨	▨																						
	■	■	■	■		■																		
Performance: Power Plant & Lift-Drag Characteristics			▨	▨	▨	▨																		
	■	■	■	■		■	■																	
Flying Qualities: Dynamic Response							▨	▨	▨						▨									
													■	■										
Flying Qual.: Long., Lat. & Dir. Stability & Control			▨							▨	▨	▨	▨	▨	▨	▨			☰					
	■	■	■	■		■	■	■	■	■	■	■	■	■	■	■	■	■						
Aircraft Systems-Flight Cont.: Pilot Assist System																	▨	▨	▨	▨				
A/C Sys.: Power Plant & Associated Equip(Phase III)	▨	▨			▨	▨													☰	▨				
						■			■	■			■	■		■	■	■						

Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1779

Primary Assignment

Aircraft was assigned to the Convair Flight Test Program during May 1954. The formal statement of the flight test assignment, as presented in Contract AF(600)-5942, as amended in accordance with the terms of negotiations conducted at the Air Materiel Command during the week of 11 June 1956, for review and re-estimate of the F/TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 26 July 1956, is quoted as follows: "Period - Four Maintenance Months". "To be utilized for dynamic response and stability and control tests, as related to the pitch and yaw damper systems". The program (ZC-8-108) for this aircraft was conducted at Edwards A.F. Base, and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair - San Diego to Edwards A.F. Base on 4 May 1954 to enter the F-102 Flight Test Program. The initial flight, for flight test purposes, was conducted on 18 May 1954. The U. S. Air Force conditionally accepted the aircraft on 31 August 1954, and finally accepted it on 24 September 1954.

During the three month period the aircraft was in the flight test program the following ground and flight testing were conducted and completed:

Phase I Flight Testing

Airspeed Calibration - Subsonic Speed Range
Angle of Attack and Angle of Yaw Pickup Calib. (Incomplete)
Dynamic Lateral-Directional Stability Without Damping
Dynamic Lateral-Directional Stability With Damping
Dynamic Longitudinal Stability Without Damping
Dynamic Longitudinal Stability With Damping
Dynamic Response to Armament Bay Door Operation

Addendum

Control System Ground Test

Flight testing for dynamic response and stability and control, as related to the pitch and yaw damper system being the prime consideration, was conducted throughout the flight test period.

The lateral stability tests proved satisfactory and optimum yaw damper gain settings were obtained. The pitch damping system, in its configuration at the end of the development program, did not produce satisfactory damping at all airspeeds at both high and low altitudes with a constant gain setting.

[redacted]

[redacted]

Summary of Activity (Continued)

The final flight, No. 24, of this program was conducted on 30 July 1954 when the aircraft was delivered to Palmdale for modification, prior to delivery to Hughes Aircraft Co. The aircraft was delivered to Hughes on 25 September 1954.

The aircraft was assigned to Convair - Palmdale for chase and target flying in support of the F-102A and TF-102A Production Flight Test and Acceptance Program, in accordance with the provisions of Sales Order No. 94-1-126, dated 1 August 1956, to Contract AF33(600)-29264.

Chronological History:

4 May 1954	- Ferried from Convair - San Diego to Edwards A.F. Base.
18 May 1954	- Initial flight for flight testing purposes.
31 July 1954	- Ferried from Edwards A.F. Base to Convair - Palmdale - Flight Test Program completed.
31 August 1954	- Conditional acceptance by the U. S. Air Force.
24 September 1954	- Final acceptance by the U. S. Air Force.
25 September 1954	- Assigned to Hughes Aircraft Company
1 August 1956	- Assigned to Convair - Palmdale.

Utilization Factors

Hours Flown			
Productive	Non-Productive	Total	No. of Flights
9.7	2.9	126	24

[redacted]

[redacted]

Page: 8A
Date: 1 Aug. 1954
Report No.: ZC-8-142
Model: YF-102

Convair Report No.:
ZC-8-108

SUMMARY OF ACTIVITY

S/N 53-1779

Primary Assignment: Flying Qualities-Dynamic Response/MG-3 AFCS

FLIGHT TEST ASSIGNMENTS	1954								1955												1956				
	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M
Dyn Long, Lat & Dir Stab & Control w/wo Dampings		▨	▨	▨																					
	■	■	■																						
Long, Lateral & Directional Stability & Control		▨	▨	▨																					
	■	■	■																						
Instrument Calibration		▨																							
	■																								

Original Planning ▨   Revised Planning ☰   Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1780

Primary Assignment

Aircraft was assigned to the Convair Flight Test Program during June 1954. The formal statement of the flight test assignment, as presented in Contract AF33(600)-5942, as amended in accordance with the terms of negotiation conducted at the Air Material Command during the week of 11 June 1956, for review and re-estimate of the F/TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 26 July 1956, as quoted as follows: "Period - 17 Maintenance Months". "To be utilized for testing and refinement of the flight control and automatic flight control systems". The program (20-8-111) for this aircraft was conducted at Edwards A.F. Base, and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair - San Diego to Edwards A.F. Base on 28 April 1954 to enter the F-102 Flight Test Program. The U. S. Air Force conditionally accepted the aircraft on 31 May 1954, and finally accepted it on 3 April 1956. The initial flight, for flight testing purposes, was conducted on 5 June 1954.

During the 17 months the aircraft was in the flight test program the following ground and flight testing were conducted and completed:

Phase III - Control Development

Dynamic Lat-Directional Stability
Directional Dynamic Response
Dynamic Longitudinal Stability
Longitudinal Dynamic Response
Full Auth. Elevon Cont. & Prelim. Pilot Assist Tests
Turn Coordinator Effect of Yaw Damper System
Longitudinal Control Forces
Angle of Attack & Angle of Yaw Pickup Cond.
Production OAT Survey
ATM Power Capability
Throttle Idle-Stop Adjustment Check
Cooling Capability of ATM System
ATM System Stability
Ground Operation of ATM

Addendums

Control System Ground Test
Boundary Layer Inlet Duct Recovery
Temp. of Eng. Outer Wall & Nozzle Actuators
Eng. Exhaust Nozzle Manual Override
Roll Rate Limiter Development Tests

Miscellaneous

Tape Recorder
Dynamic Response Characteristics for HAC
I.L.S. Tests

[redacted]

[redacted]

Summary of Activity (Continued)

This aircraft was primarily utilized throughout its flight test program for the testing and refinement of the flight controls and automatic flight control systems. The aircraft was also primarily utilized for obtaining dynamic response data for the development of the Hughes Company's fire control system.

During the middle portion of the program, testing of the pneumatic system ATM power and cooling was conducted along with the pilot assist development tests.

The data obtained from the control, damping and pilot assist systems contributed a great deal toward the development of a satisfactory flight control system for the F-102 aircraft, as it had been proven that the design philosophy in the control system was sound and developed to a degree whereby the only principal problems remaining were component reliability. Consequently, the results of these tests were later incorporated into the F-102 8-10 versions.

The test program of this aircraft was completed on 11 October 1955, and was ferried from Edwards A.F. Base to Convair - Palmdale on 27 October 1955, for utilization for pilot proficiency and target and chase plane missions, per Sales Order No, 63-1-125.

Chronological History:

28 April 1954 - Ferried from Convair - San Diego to Edwards A.F. Base.
31 May 1954 - Conditionally accepted by the U. S. Air Force.
5 June 1954 - Initial flight for flight testing purposes.
11 October 1955 - Completed flight test program.
27 October 1955 - Ferried from Edwards A.F. Base to Convair - Palmdale.
3 April 1956 - Finally accepted by the U. S. Air Force.

[redacted]

[redacted]

Page: 11A
Date: 1 Nov. 1955
Report No.: ZC-8-142
Model: YF-102

Convair Report No.:
ZC-8-111

SUMMARY OF ACTIVITY

S/N 53-1780

Primary Assignment: Phase III, Control Development

FLIGHT TEST ASSIGNMENTS	1954							1955												1956					
	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J
Flying Qual.: Long, Lat & Dir Stability & Control	▨ █	▨ █	▨ █	█	█	█	█	█	█	█	█	█	█	█	█	☰	☰ █								
Flying Qual.-Dyn. Response: Long, Lat & Dir Stab & Cont	▨ █	▨ █	▨ █	█				█	█		█		█	█	█	☰	☰ █								
Aircraft Systems-Flight Controls: Pilot Assist	▨	▨	▨		█	█	█	█	█	█	█	█	█	█	█	☰									
A/C Systems: Pneumatic			▨	▨	▨	▨	▨	▨ █	█	█	▨ █	▨													
A/C Systems: Hydraulics											▨	▨													
A/C Systems: Electrical													▨												
A/C Systems: Power Plant						▨	▨	▨	▨	▨	▨														

Original Planning ▨ Revised Planning ☰ Activity █

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1781

Original Program

Primary Assignment

Aircraft was assigned to the Convair Flight Test Program during June 1954. The formal statement of the flight test assignment, as presented in Contract AF(600)-5942, as amended in accordance with the terms of negotiation conducted at the Air Materiel Command during the week of 11 June 1956, for review and re-estimate of the F/TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 26 July 1956, is quoted as follows: "Period - 20 Maintenance Months". - "To be utilized for armament mechanism operation and structure, loads, captive missile loads, missile launching, 2.75 and 2.0-inch rocket firing and ATM development for negative"G"flight condition. After completion of this program and modification of the airplane, the airplane is to be utilized for Rocket Type MG-1 tethered ground firing and airborne firing as well as stability flight tests. In addition this airplane will be utilized to ascertain rocket temperature data, grain characteristics of propellant, trajectory data, ingesting of rocket exhaust gases to check aircraft engine response". The program (ZC-8-112), as originally planned, was conducted at Edwards A.F. Base from 10 June to 4 October 1954, thence to Holloman A.F. Base until 3 June 1955, and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair - San Diego to Edwards A.F. Base on 19 May 1954 to enter the F-102 Flight Test Program. The initial flight for flight test purposes was conducted on 10 June 1954, for production shakedown. The U. S. Air Force conditionally accepted the aircraft on 30 June 1954.

During the 12 month period (Original Program) the aircraft was in the flight test program, it conducted and completed ground and flight testing on the following items:

8-82 Armament Development, MB-1 Rocket

Armament Bay Door Operation
Armament Door Cyl. Rod End Loads
Door Acceleration Measurement
Armament Launcher Ext. Arm Oper.
Armament Launcher Operation
Missile Separation
Missile Sep. & Traj. at V_{max} & Hp Max.
Missile Trajectory
Rocket Separation and Dispersion
Dynamic Resp. to Armament Oper.
Rocket & Miss. Exhaust Gas Effects
Fuel Venting Pressure Measurements
Preliminary R.A.T. Tests

During the early phases of flight testing at EAFB, the aircraft was utilized for shakedown and familiarization flights, and cockpit smoke evaluation; and for armament mechanism operation, structural loads and performance testing during the middle and latter period.

[redacted]

[redacted]

Summary of Activity (Continued)

The aircraft was transferred from Edwards A.F. Base and ferried to Holloman A.B. Base on 4 October 1954 to continue its flight test program in armament development. During the early stages at Holloman the first missile was successfully launched at 40,000 feet during flight 26 on 28 October 1954, and the first rocket launched during flight 27 on 9 November 1954. Missile and rocket launchings continued for the remaining period of the "original" program. The captive missile loads and missile launching programs were completed, as well as the 2.75" and 2.0" rocket programs. Generally, the armament tests conducted during the entire period were considered satisfactory, and flight characteristics and stability of the aircraft at combat ceiling with the missile bay doors open were good.

The primary contributions, as a result of this aircraft's test program, to the F-102 program are as follows: Satisfactory information and data obtained on the separation, dispersion and trajectory of rockets and missiles; satisfactory response to rocket and missile firing, and the firing of a full load of rockets at maximum altitude without causing engine snuffing; and the development of the ATM to resolve the adverse negative "G" flight condition.

The aircraft was then temporarily transferred from Holloman A.F. Base and ferried to Convair - San Diego, via Luke Field, Arizona on 3 June 1955, for modification to a fixed pylon configuration for MB-1 rockets.

Chronological History:

19 May 1954 -	Ferried from Convair - San Diego to Edwards AF Base.
10 June 1954 -	Initial flight for flight testing purposes.
30 June 1954 -	Conditionally accepted by the U. S. Air Force.
3 & 4 October 1954 -	Ferried from EAFB to Holloman AF Base.
3 June 1955 -	Ferried from Holloman AF Base to Convair - San Diego for modification to a fixed pylon configuration

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Page: 14A
Date: 1 July 1955
Report No.: ZC-8-142-
Model: YF-102

Convair Report No.:
ZC-8-112

SUMMARY OF ACTIVITY

S/N 53-1781

Primary Assignment: 8-82 Aux. Sys. Oper.-Armament Development, Rocket MB-1

FLIGHT TEST ASSIGNMENTS	1954							1955												1956					
	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J
Armament Mechanism Operation & Structure Loads		▨	▨																						
	■	■	■						■		■	■													
Captive Missile Loads				▨	▨																				
							■	■																	
Missile Launchings			▨			▨	▨	▨	▨	▨															
				■	■			■	■	■	■	■													
Rocket Launchings			▨								▨	▨	▨	▨											
						■						■													
Fly Qual: Dyn Response Long, Lat&Dir Stab&Cont-Arm Oper.		▨	▨			▨	▨	▨																	
	■	■	■						■			■													

Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1781

Reinstated Program

Primary Assignment

Aircraft was reassigned to the Convair Flight Test Program during October 1955, after it was modified to a fixed pylon configuration for MB-1 type rockets at Convair-San Diego during June thru September 1955. The formal statement of the flight test assignment, as presented in Contract AF(600)-5942 is the same as that listed in the original program of S/N 53-1781 filed in Appendix A of this report. However, for the convenience of the reader, that portion pertaining to this reinstated program is re-quoted herein as follows: "After completion of this program, unquote (the original program), quote, and modification of the airplane, the airplane is to be utilized for Rocket type MG-1 tethered ground firing and airborne firing as well as stability flight tests. In addition this airplane will be utilized to ascertain rocket temperature data, grain characteristics of propellant, trajectory data, ingesting of rocket exhaust gases to check aircraft engine response". The program (ZC-8-131), as originally planned, was conducted at Holloman Air Force Base from 11 October 1955 to 13 July 1956; and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair-San Diego to Holloman Air Force Base on 11 October 1955 to re-enter the F-102 Flight Test Program. The initial flight (No. 88) for flight test purposes, after its re-entery into the program, was conducted on 17 October 1955.

During the nine month period (Reinstated Program), the aircraft was in the flight test program it conducted and completed ground and flight testing on the following items:

8-82 Armament Development, MB-1 Rocket

Tethered Test Firing
Ramp Jettison Test of the MB-1 Launcher
Air Jettison of Inert Rockets

During the early portion of the program the aircraft was primarily utilized for obtaining dynamic response with pylon fairing on and off, directional stability characteristics, taxi and flight testing with rocket installed, flight stability with/without rocket, pylon airflow study with/without rocket installed, and rocket environmental check flights. The aircraft was then prepared for tethered launchings to launch two MB-1 rockets. These launchings were conducted and completed satisfactorily during April 1956. During the latter portion of the program the aircraft was primarily utilized to obtain armament hardware, aircraft response, separation, and ballistic information on the MB-1 rocket (modified), with a secondary purpose of gaining a certain confidence level in the MB-1 round. Of the ten rockets originally programmed for firing, eight were successfully launched

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[redacted]

S/N 53-1781 Summary of Activity - (Continued)

during this latter period. As observed by pilots in the cockpit, no two identical launchings were observed out of the eight launchings conducted. However, the last round, No. 8, appeared to have the most satisfactory separation and trajectory characteristics. The remaining two rounds were subsequently deleted from the aircraft's program and rescheduled for firing on S/N 53-1797.

The primary contributions to the F-102A program, as a result of this aircraft's program, are as follows: First launching of an MB-1 Rocket from a YF-102 tethered aircraft, and first launching of an MB-1 Rocket (Modified) from an airborne YF-102 aircraft. In the latter instance a great deal of confidence was realized in the ability to launch the rocket safely from an airborne aircraft.

The aircraft completed its flight test program during flight No. 121 on 6 July 1956, and was ferried from Holloman Air Force Base to Convair-Palmdale on 13 July 1956 for utilization for pilot proficiency, target and chase plane missions, per Sales Order No. 63-1-125.

Chronological History:

11 October 1955 - Ferried from Convair-San Diego to Holloman Air Force Base.
1 May 1956 - First airborne launching of an MB-1 Rocket (Mod.) from a YF-102 aircraft
13 July 1956 - Ferried from Holloman Air Force Base to Convair-Palmdale

[redacted]

[redacted]

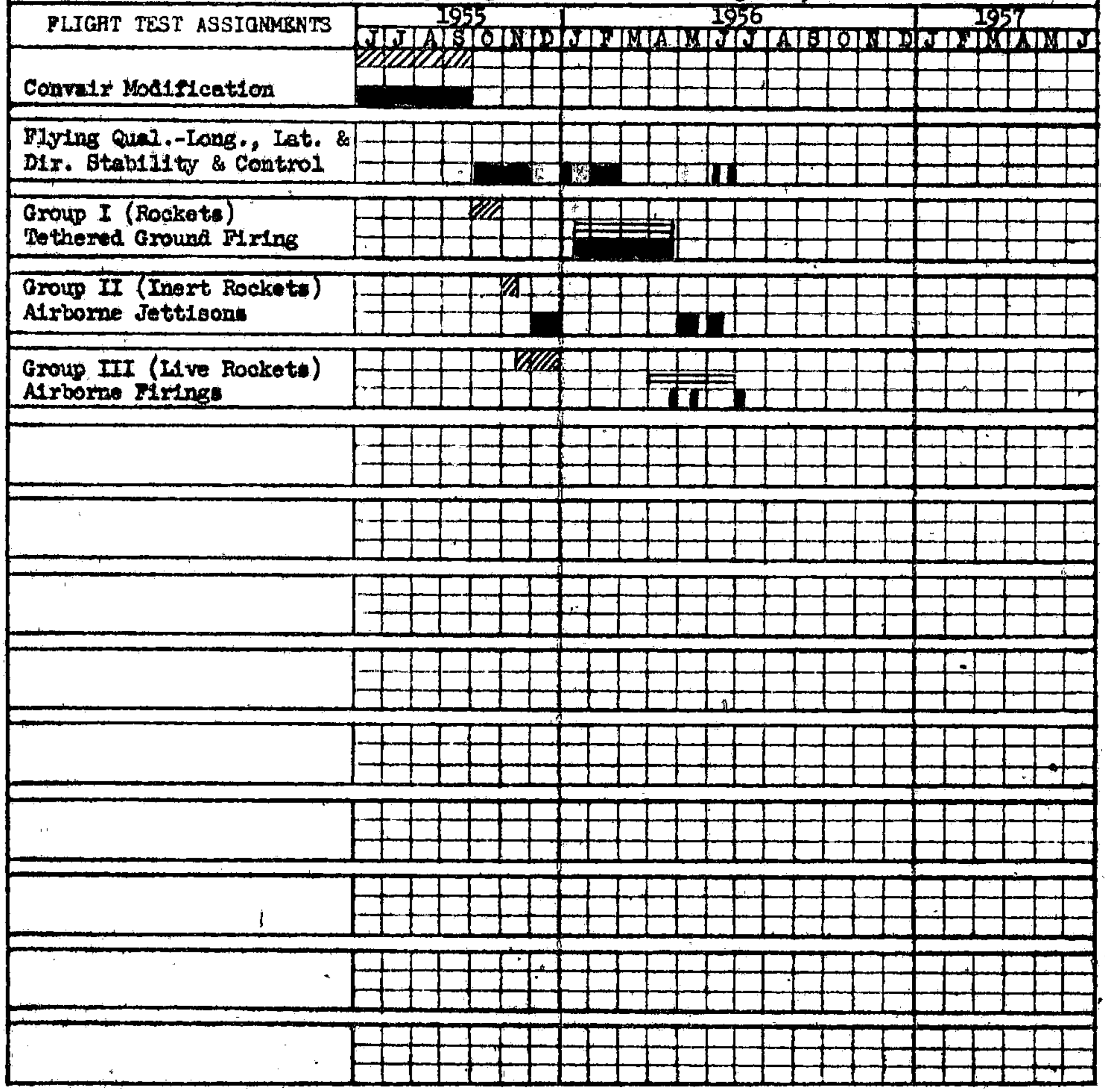
Page: 17A
Date: 1 Nov. 1956
Report No.: ZC-8-142
Model: YF-102

Convair Report No.:
ZC-8-131

SUMMARY OF ACTIVITY

S/N 53-1781

Primary Assignment: 8-82 Aux. Sys. Oper.-Armament Development, Rocket MB-1



Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1782

Primary Assignment

Aircraft was assigned to the Convair Flight Test Program during June 1954. The formal statement of the flight test assignment, as presented in Contract AF(600)-5942, as amended in accordance with the terms of negotiation conducted at the Air Materiel Command during the week of 11 June 1956, for review and estimate of the F/TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 26 July 1956, is quoted as follows: "Period - One Maintenance Month". "To be utilized to demonstrate the functional characteristics of the production antenna installations and to demonstrate operation of the emergency fuel system, ring type dacron drag chute and the electrical system power controller." The program (ZC-8-113) for this aircraft was conducted at Edwards A. F. Base, and is graphically illustrated, as planned and performed, of the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair - San Diego to Edwards A. F. Base on 19 June 1954, to enter the Flight Test Program. The initial flight for flight test purposes was conducted on 28 June 1954, for production shakedown. The U. S. Air Force finally accepted the aircraft on 21 October 1954.

During the two month period the aircraft was in the flight test program it conducted and completed ground & flight testing on the following items:

Antenna Pattern/MG-3 System Development

Antenna - Functional Characteristics - AN/ARN-18 Glide Path
" - " " - AN/ARN-14 Omni
" - " " - AN/ARN-25 UHF/DF
" - " " - AN/ARC-34
" - " " - AN/ARN-12

Addendums

Glide Path Antenna - Location Study

The aircraft was originally scheduled to enter the flight test program during the latter portion of July 1954. However, the aircraft came off the production line ahead of schedule. Subsequently, it entered the flight test program accordingly.

Performance testing for production shakedown flights, airspeed calibration, static source checks and pilot familiarization flights, along with some communications & navigation

[redacted]

[redacted]

Addendum (Continued)

Glide Path Antenna - Location Study (Continued)

testing, were conducted during the first half of the program. Compressor stall investigations were conducted concurrently with the communications & navigation tests during the balance of the program. The aircraft demonstrated operation of the emergency fuel system, ring type dacron drag chute and electrical system power controller at different intervals throughout the program. Satisfactory results were obtained on all phases of functional equipment testing of the program.

The primary contributions, as a result of this aircraft's test program, to the F-102 program are as follows: Determined that all antennas were compatible; and Completed a portion of the F-102 antenna specification compliance testing.

The test program of this aircraft was completed on 19 August 1954, at which time the aircraft was ferried to Convair - Palmdale for modification prior to its delivery to Hughes Aircraft Company.

Chronological History:

19 June 1954 -	Ferried from Convair - San Diego to Edwards A. F. Base.
28 June 1954 -	Initial flight for flight testing purposes.
19 August 1954 -	Completed flight test program & ferried to Convair - Palmdale.
21 October 1954 -	Finally accepted by the U. S. Air Force & transferred to Hughes Aircraft Company.

[redacted]

Convair Report No.:
ZC-8-113

SUMMARY OF ACTIVITY

S/N 53-1782

Primary Assignment: Aux. Sys. Oper.-Electronics-Antenna Pattern/MG-3 Sys. Development

FLIGHT TEST ASSIGNMENTS	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J
Communications & Navigation		■	▨ ■																						
Performance - General		■																							
Instrumentation Calibration	■																								

Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

UNCLASSIFIED

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1783

Primary Assignment

Aircraft was assigned to the Convair Flight Test Program during August 1954. The formal statement of the flight test assignment, as presented in Contract AF 33(600-5942, as amended in accordance with the terms of negotiation conducted at the Air Materiel Command during the week of 11 June 1956, for review and re-estimate of the F/TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 26 July 1956, is quoted as follows: "Period - 10 Maintenance Months". "To be utilized for qualitative testing of dry air anti-icing, defogging and cabin ventilation and pressurization, completion of communications and navigation system tests transferred from airplane S/N 53-1782, and Air Force evaluation of the airplane in various icing conditions." The program (ZC-8-114) for this aircraft was conducted at Edwards AFB, and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair-San Diego to Edwards AFB on 8 July 1954, to enter the F-102 Flight Test Program. The initial flight for flight test purposes was conducted on 16 August 1954. The United States Air Force conditionally accepted the aircraft on 30 July 1954, and finally accepted it on 24 January 1956.

During the 10-month period the aircraft was in the flight test program it conducted and completed ground and flight testing on the following items:

Phase III, Cooling, Electrical, Etc.

- Airspeed Calibration-Subsonic Speed Range
- OAT Pickup Location Study
- Ground Test of Wing, Tail, & Engine Inlet Anti-Icing System
- Wing, Tail, & Engine Inlet Anti-Icing System & Windshield (NESA) Anti-Icing System (Climb & Descent Tests)
- Wing, Tail & Engine Inlet Anti-icing System & Windshield (NESA) Anti-icing System (Level Flight Tests)
- Radome Anti-icing
- Windshield & Canopy Defogging System
- Heating & Vent Equip. Spec. C/O
- Cabin Pressure Specification C/O

Addendums

- Temperature of Engine Outer Wall Nozzle Actuators
- Windshield Canopy Defogging System Test
- Cabin Flight Test
- Antenna Flight Test
- Performance of R.A.T. Emergency Hydraulic Power Unit

[redacted]

~~UNCLASSIFIED~~

[redacted]

Addendum (continued)

The original flight test program for this aircraft was changed extensively during the first half of the program period. Instrumentation calibration, production items, engine and structural cooling, and pilot assist phases of the test program were periodically deleted in order to arrange for the delivery of the aircraft to WADC on 11 January 1955. Dry air anti-icing, defogging, and cabin ventilation and pressurization tests were necessarily limited to qualitative testing as a result of this arrangement.

While at WADC, U. S. Air Force pilots evaluated the aircraft in various icing conditions. The majority of flights were conducted under artificial conditions created behind tanker type aircraft. Preliminary information indicated that the aircraft, without wing & fin anti-icing systems, would perform satisfactorily under artificial as well as actual icing conditions that were encountered. As a result of these tests the anti-icing equipment was deleted from the 8-10 aircraft with a considerable saving in weight.

The primary contributions, as a result of this aircraft's test program, to the F-102 program are as follows: That the F-102 aircraft could be utilized in icing conditions without wing & fin anti-icing systems, completed antenna specification compliance testing, and demonstrated satisfactory instrument and night flying characteristics.

The anti-icing tests were completed by WADC on 3 May 1955, and the aircraft ferried from Wright-Patterson A. F. Base, Ohio on 14 May 1955, arriving at Edwards A. F. Base on 15 May 1955. The aircraft was then ferried from Edwards A. F. Base to Convair - Palmdale on 20 May 1955, for utilization for pilot proficiency and target and chase plane missions, per Sales Order No. 63-1-125.

Chronological History:

8 July 1954 - Ferried from Convair - San Diego to Edwards A. F. Base.
30 July 1954 - Conditionally accepted by the U. S. Air Force.
16 August 1954 - Initial flight test.
8-11 January 1955 - Ferried from Edwards A. F. Base to WADC at W-P. A. F. Base, Ohio.
3 May 1955 - Completed flight test program.
14-15 May 1955 - Ferried from WADC at W-P A. F. Base, Ohio, to Edwards A. F. Base.
20 May 1955 - Ferried from Edwards A. F. Base to Convair - Palmdale.
24 January 1956 - Finally accepted by the U. S. Air Force.

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Page: 23A
Date: 1 Feb. 1956
Report No.: ZC-8-142
Model: YF-102

Convair Report No.:
ZC-8-114

SUMMARY OF ACTIVITY

S/N 53-1783

Primary Assignment: Phase III - Aircraft Systems - Cooling, Electrical, etc.

FLIGHT TEST ASSIGNMENTS	1954					1955												1956							
	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A
Instrumentation Calibration of Production Items	▨																								
Dry Air Anti-icing, defogging cabin vent & pressurization	▨	▨	▨ ■	▨ ■	▨ ■				■	■															
Engine & Structural Cooling & Cooling Drag						▨	▨	▨																	
Fly. Qual-Dyn. Response: Long, Lat & Dir Stab & Control									▨	▨	▨														
Pilot Assist System												▨	▨	▨											
Performance & Fly Qual: Actual Icing Flights						■	■	■	☰ ■	☰															
Communications & Navigation Systems																									

Original Planning ▨ Revised Planning ☰ Activity ■

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1787

Primary Assignment

Aircraft entered the Convair Flight Test Program during December 1954. The formal statement of the flight test assignment, as presented in Contract AF-33(600)-5942, as amended in accordance with the terms of negotiations conducted at the Air Materiel Command during the week of 11 June 1956, for review and re-estimate of the F/TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 31 Oct. 1956, is quoted as follows: "Period - 19 Maintenance Months". "To be utilized for qualitative evaluation of airplane systems and air speed calibrations. Phase I performance and preliminary flight characteristics; Phase II, roll rate tests, high speed vibration and buffet investigations and other Phase III testing. Following a two month test period for the F-102B program (not a part of this proposal), the airplane will be utilized for flight test techniques development as pertaining to the Convair F-102 interceptor programs". The program (ZC-8-118), for this aircraft was conducted at Edwards Air Force Base, and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair-San Diego to Edwards Air Force Base on 15 December 1954, to enter the F-102 Flight Test Program. The initial flight, for flight test purposes, was conducted on 20 December 1954. The U. S. Air Force conditionally accepted the aircraft on 23 December 1954.

During the 22 months the aircraft has been in the flight test program, it has been utilized for flight testing the following primary phases of its program:

Phase I, II, III, AFCS Development/Flight Test Techniques Development

Phase I flight testing by the Contractor to demonstrate, through ground tests and initial flight tests, that the complete aircraft and supporting equipment meets engineering specifications and will function. Test Item 1 - Ground Tests; Test Item 2 - Aircraft System Shakedown; Test Item 3 - Air Speed Calibration; Portions of Test Item 4 - Aircraft Performance; and Test Item 5 - Preliminary Flight Characteristics.

Phase II flight testing by the U. S. Air Force for compliance with contractual requirements, quantitative performance, and qualitative handling characteristics (aircraft, material, equipment). Portions of Test Item 4 - Aircraft Performance.

Phase III flight testing to overcome deficiencies in the aircraft and equipment found during Phases I and II. Test Item 6 - Systems Development on the Ram Air Turbine and Pilot Assist System; and Test Item 7 for Dynamic Response.

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S/N 53-1787 Summary of Activity - (Continued)

Addendum Items: High Speed Vibration (Buzz) Investigation, Ejector Tail Pipe Evaluation, and Wing Vent Pressure Evaluation.

The Phase III Test Item 8 - Stability and Control Tests, scheduled for this aircraft, were transferred to S/N 53-1791.

The primary progress made to date to the F-102 program, as a result of this aircraft's test program are as follows: Phase I performance tasks and preliminary flight characteristics were satisfactorily accomplished, confirming the performance predictions for the Model 8-10 aircraft; qualitative evaluations of the aircraft's systems were conducted, which greatly enhanced the refinement of the systems, including tests required to develop the manual mode surface controls system to a satisfactory degree, as well as the trim servo providing a stable stick force gradient in the sonic, transonic, and supersonic speed ranges; other testing has resolved the lateral acceleration warning light pitch "g" limiter tie to the pilot assist, beep trim development of the pitch attitude and rate gain optimization, up elevator function and supersonic buffeting; a great deal of information was obtained on high speed vibration or "buzz"; considerable development testing was conducted to improve the tail pipe ejector configurations, from which the data obtained indicated a thrust improvement during non-after burning subsonic flights, as well as performance characteristics being determined for transonic and supersonic speed ranges; the alternate air speed study has indicated promise of an alternate system consisting of a chin pitot probe with a pair of fuselage static sources. Further testing included a second alternate; the chin nose boom system. Modified wing fence evaluation tests did not reveal any major adverse effects, as compared to the original wing fence configuration, to the aircraft's general performance. However, a general overall evaluation between the two configurations revealed that the original configuration produced more stable effects to the aircraft's handling characteristics. As a result, a decision was made to retain the original configuration.

During the first two and one-half weeks in September, 1956, the aircraft was on a ground work status for instrumentation calibrations for elevon position measurement techniques. During the remainder of the month, elevon position, skin thermocouples, Topp transducer, and alternate air speed tests were conducted. Flight testing of measurements of elevon position and control systems deflections, Topp angle of attack transducers, and comparison of skin-temperature thermocouples were all completed during the period. Flight testing of forward center strike camera fairing buffet investigation was conducted at the end of the period, as well as a pilot check-out flight and chase support mission. One flight was conducted during the early portion of October, 1956, to investigate aircraft buffet with forward center strike camera fairing. With the fairing modified to a frontal angle of 68°, the buffet, which was experienced throughout the airframe on the previous flight, was reduced from severe to moderate at supersonic speeds.

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S/N 53-1787 (Continued)

Future Activity

Aircraft completed its flight test program on 2 October 1956 at the termination of flight 216, and was placed on a ground work status for instrumentation removal in preparation for delivery to Convair-Palmdale. Aircraft was assigned to the Convair-Palmdale facility for chase and observation flights, as a high and low altitude target for check-out of fire control systems, and other operational and maintenance purposes per Convair Sales Order No. 94-1-126, dated 1 August 1956.

Chronological History

15 December 1954 - Ferried from Convair-San Diego to Edwards Air Force Base
20 December 1954 - Initial flight test conducted
23 December 1954 - Conditionally accepted by the United States Air Force
15-21 January 1955 - Phase II testing - United States Air Force
8-18 March 1956 - Modification for F-106A exhaust ejector at Edwards AFB
19 March 1956 - Commenced F-106A exhaust ejector tests
25 April 1956 - Completed F-106A exhaust ejector tests
2 October 1956 - Completed flight test program at the termination of flight 216.

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UNCLASSIFIED

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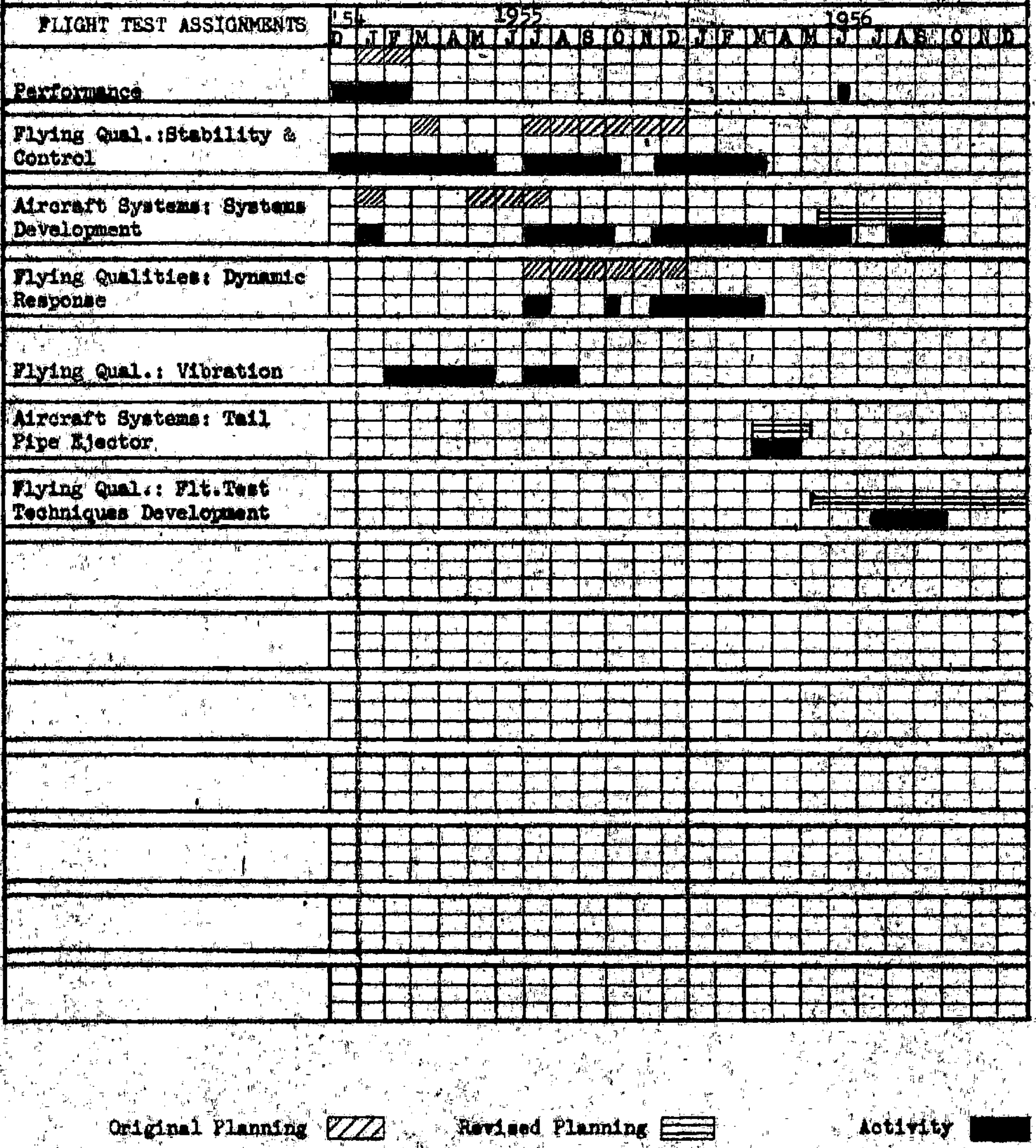
Page: 27A
Date: 1 Nov. 1956
Report No.: ZC-8-142-
Model: YF-102A

Convair Report No.:
ZC-8-118

SUMMARY OF ACTIVITY

S/N 53-1787

Primary Assignment: Phase I, II, III, AFCS Development/Flight Test Techniques Dev.



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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1788

Primary Assignment

As stated in Convair Sales Order No. 33-1-626, dated 31 October 1956, this aircraft is "to be utilized for functional development of armaments including mechanisms, dynamic response to armaments, firings of missiles and 2.75-inch and 2.0-inch rockets, fire control jump angle and captive missile loads and for photo chase and test support."

Summary of Activity

This aircraft has completed its portion of the flight test schedule. During its period in the Flight Test Program flights were conducted on a variety of tests; the initial testing were missile and rocket firings. Following this was an extensive 2.00-inch rocket firing period conducted concurrently with testing for captive missile loads and dynamic response. Angle of attack testing was initiated at the beginning of 1956; both measurements and calibrations were obtained. Testing was again conducted during the latter part of the program for rocket and missile firing during which time missile trajectory data were obtained. The program concluded with 2.75-inch rocket firings.

As a result of the flight testing on this aircraft the suitability of the F-102 as a missile and rocket launcher has been determined. Functional development of the armament sytem has been furthered as a result of extension of altitude and Mach number ranges. Information was obtained for setting fire control system parameters on 2.00-inch rocket jump angle, angle of attack calibration, 2.75-inch rocket jump angle, and GAR-1 missile trajectory. Effects of missile and rocket exhaust gases on missile air domes were determined.

The aircraft was used solely on photo test support missions upon completion of the regular program 31 August 1956. Photo test support flights were concluded at the end of August 1957.

Chronological History

26 January 1955 - Ferried from Convair-San Diego to Convair-Palmdale
31 January 1955 - Conditionally accepted by the U.S. Air Force
22 February 1955 - Ferried from Convair-Palmdale to Holloman Air Force Base
24 February 1955 - Initial flight test conducted
31 August 1956 - Armament Program completed at the termination of Flight 195.
29 August 1957 - Photo test support missions completed.

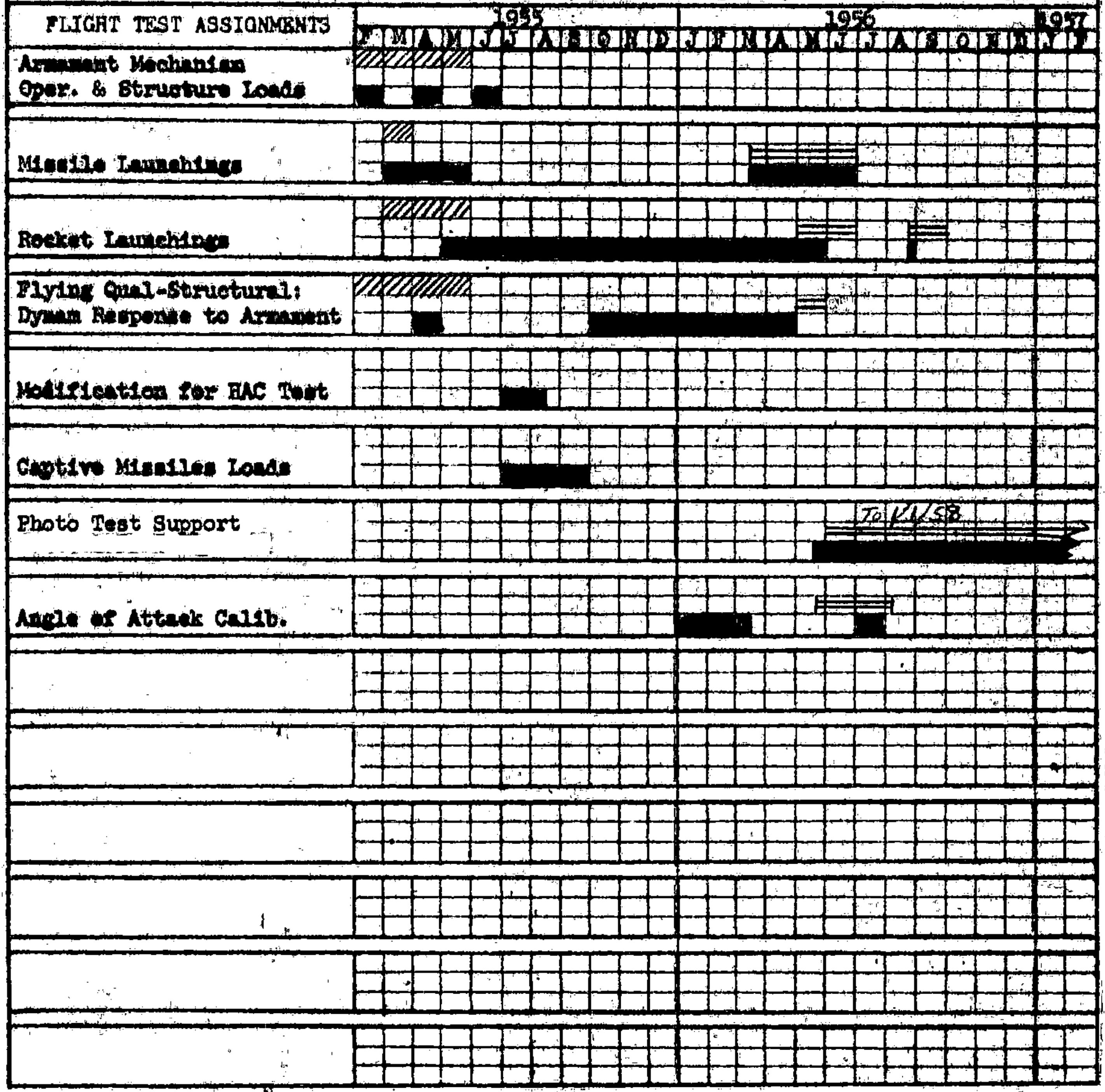
[redacted]

Convair Report No.:
ZC-8-119 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 53-1788

Primary Assignment: 8-90 Aux. Sys. Oper.-Armament Development, Missiles & Rockets



Original Planning ▨ Revised Planning ▤ Activity ■

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1791

Primary Assignment

The flight test assignment for this aircraft as presented in Convair Sales Order No. 33-1-626, dated 31 October 1956, states that it is "to be utilized for Phase I, III, and IV tests, flutter and vibration test (particularly in regards to the high speed vibration investigation) external fuel tanks (2), spin and stall tests and systems development." Added program: ARDC Advanced Instrument Display.

Summary of Activity

The primary progress made to date to the F-102 program, as a result of this aircraft's test program, is as follows:

During the flutter tests, to remove speed restrictions, the aircraft demonstrated an unsatisfactory buzz condition above Mach 1.20. As a result, extensive flight testing between July 1955 through January 1956 consisted of buzz investigation. This condition was corrected by installing a "Buzz Fix", which primarily consists of the following: Extended engine inlet ramp, engine inlet duct inter-wall beef-up, and bifurcation of the engine inlet duct wedge. After these modifications were made the aircraft satisfactorily conducted buzz free flights up to the specification Mach number 1.5.

After the initiation of the F-102 Flight Test Program a decision was made to extend the range of the F-102 aircraft by the addition of two external fuel tanks. The addition of the external fuel tanks to the F-102 aircraft posed a problem of proving that they could be utilized up to the established limitations. This aircraft was selected, to establish aircraft performance (structurally and aerodynamically), with external fuel tanks installed. The in-flight flutter tests were conducted and data were obtained on the full, 1/2 full forward loading, and 1/2 full aft loading configurations. Testing during November and December consisted of external fuel tank jettison tests, main landing gear wheel well tests, and in-flight fuel transfer tests. During one fuel tank jettison test, both tanks were jettisoned simultaneously while flying 36,515 feet true pressure altitude and .953 M_T.

ADDED PROGRAM

The aircraft was delivered to Edwards Air Force Base 12 July 1957 for functional check of the ARDC Advanced Instrument Display, Programmer, VOR, ADF, and DME.

Flight tests were conducted on the above and results were satisfactory excepting the inoperable DME.

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[redacted]

Brief Summary of Aircraft Activity S/N 53-1791 - (Continued)

Future Activity

The aircraft was delivered to WADC.

Chronological History

20 June 1955 - Ferried from Convair-San Diego to Edwards Air Force Base
24 June 1955 - Initial flight test conducted
29 June 1955 - Conditionally accepted by the U.S. Air Force
14 February 1956 - Beginning of Phase IV testing by the U.S. Air Force
4 May 1956 - Completion of Phase IV testing by the U.S. Air Force
27 December 1956 - Completed Convair External Fuel Tank testing
12 July 1957 - ARDC Advanced Instrument Display functional check begun
31 July 1957 - Completed Convair flight test program

[redacted]

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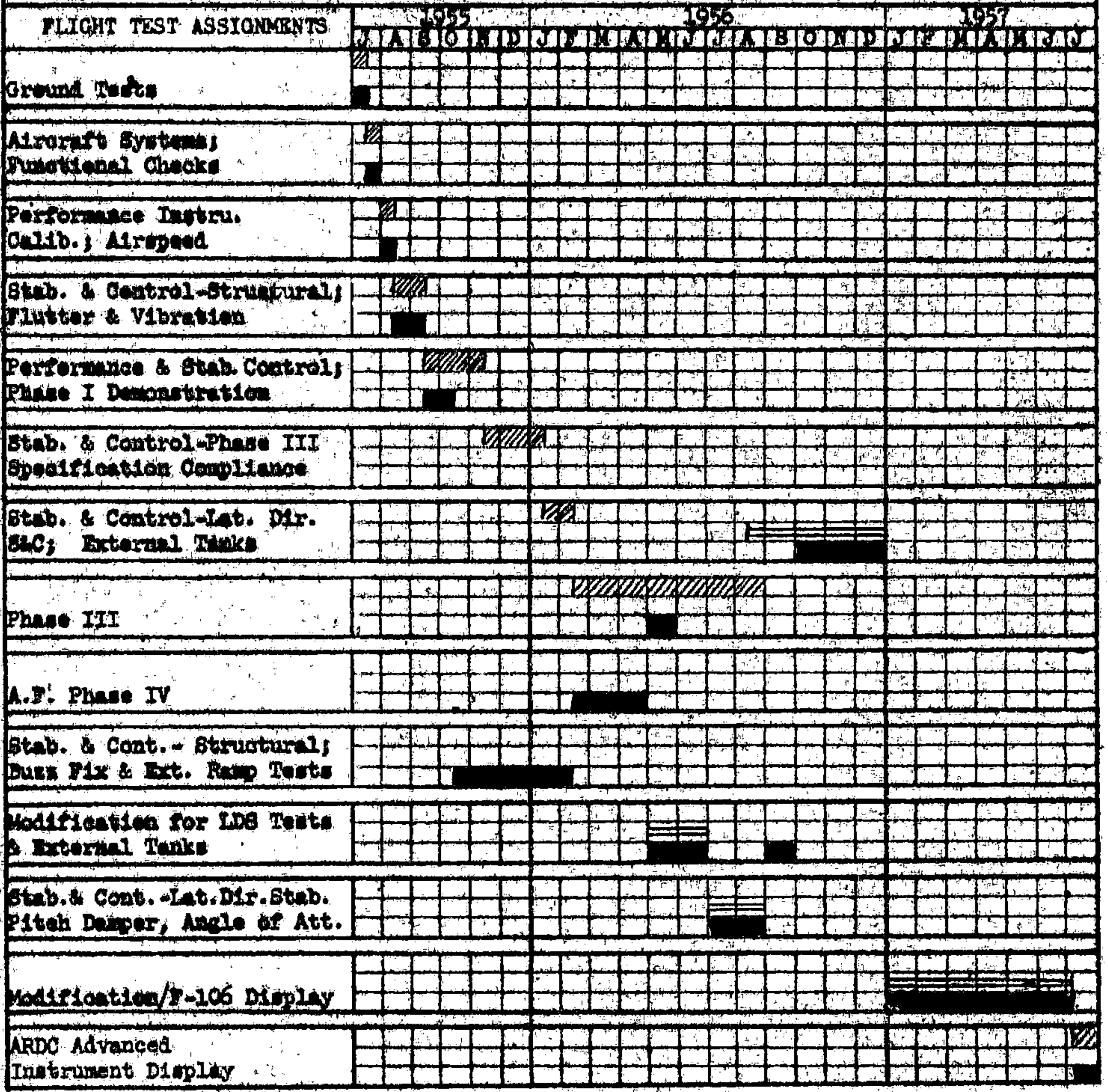
Page: 32A
Date: 1 Sept. 1957
Report No.: ZC-8-142-
Model: F-102A

Convair Report No.:
ZC-8-120 (Flt. Test Program)
ZC-8-153 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 53-1791

Primary Assignment: 8-10 Phase I-III-IV, Inlet Extended Ramp



Original Planning ▨ Revised Planning ☰ Activity ■

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1793

Primary Assignment

This aircraft, as stated in Convair Sales Order No. 33-1-626, dated 31 October 1956, is "to be utilized for armament hardware performance, airplane stability and response to armament operation, missile firing, rocket separation, trajectory and dispersion with 24 rocket doors and 36 rocket doors, armament bay air loads and corrosion effects."

Summary of Activity

During the first three-month period the aircraft was in the program, structural loads were imposed on the armament displacement mechanism and measured with dummy missiles in place, while the aircraft was on the ground. The loads imposed, in this respect, were found to be within the design limits of the aircraft. Armament hardware checks were made in subsonic and supersonic flights at loads up to 3 "g". The loads were well within design limits and subsequently lifted the restrictions on the armament doors and launcher gear. This was the first aircraft to launch GAR-1 missiles. One missile launching flight (No. 13) was being concluded, when the aircraft undershot the runway on 13 December 1955, causing considerable damage to the aircraft. The fuselage was shipped to Convair-San Diego, via truck, for repair and modification on 10 January 1956, per Convair Sales Order No. 33-2-327.

After undergoing extensive repairs and modification at Convair-San Diego, aircraft was again released for flight on 7 June 1956 and ferried to Edwards Air Force Base for U. S. Air Force acceptance, which was accomplished the same date. Aircraft was then ferried from Edwards Air Force Base to Holloman Air Force Base on 8 June 1956 to re-enter the armament flight test program, and was utilized for the balance of the month for shakedown and flight testing of armament hardware loads. Flights were then conducted to demonstrate operation and performance of the F-102A aircraft as a launching platform for GAR-1 and GAR-2 missiles throughout the design envelope. Tests were also conducted concurrently to determine the airloads and vibrations on the missile launchers. At the turn of the year, the rocket doors were modified for 2.75" FFAR. The addition of debris deflectors required door flutter tests to lift limitations on all aircraft equipped per ACA-8-1784.

A number of 2.75" rockets (approximately 500) were fired covering the entire design envelope to demonstrate operation and performance of the aircraft as a rocket firing platform and to provide jump angle data for the Hughes air data computer.

Two tests were made to investigate buffet problems on F-106 forward and side camera fairings.

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY S/N 53-1793 (Continued)

The aircraft has been transferred to tactical status by the Air Force.

Chronological History:

13 July 1955 - Ferried from Convair-San Diego to Convair-Palmdale
17 August 1955 - Conditionally accepted by the U. S. Air Force
19-21 October 1955 - Ferried from Convair-Palmdale to Holloman Air Force Base
27 October 1955 - Initial flight for flight testing purposes
13 December 1955 - Aircraft damaged during landing operation of Flight No. 13
10 January 1956 - Fuselage shipped vis truck from Holloman to Convair-San Diego
7 June 1956 - Ferried from Convair-San Diego to Edwards Air Force Base
7 June 1956 - U. S. Air Force accepted aircraft
8 June 1956 - Ferried from Edwards Air Force Base to Holloman Air Force Base
21 December 1956 - Completed missile firing program.
26 April 1957 - Completed Flight Test Program

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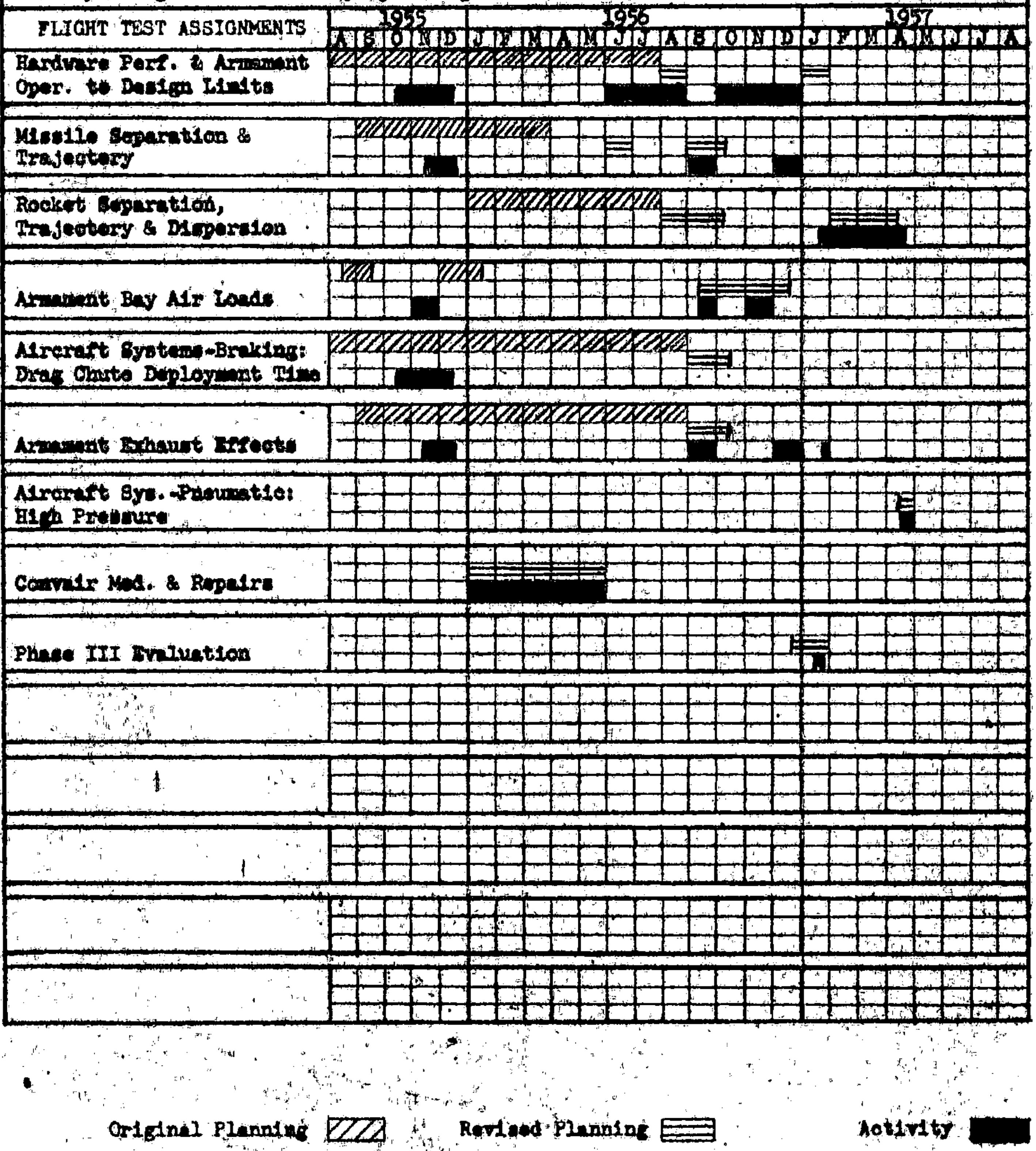
UNCLASSIFIED

Convair Report No.:
ZC-8-121
ZC-8-242 Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 53-1793

Primary Assignment: Auxiliary System Operation - Armament Phase III



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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1794

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-626, dated 31 October 1956, is quoted as follows: "To be utilized for structural integrity demonstration and flight loads and air loads determination performed in two phases covering approximately 58 (WADC) test points." Added Program: Pitch "g" limiter and Broficon antenna tests (F-106A items).

Summary of Activity

After the initial functional test flight, the aircraft was on a ground work status for an engine change from a J-57-P41 to a J-57-P23. Extensive instrumentation installation, rework and checks, with numerous delays encountered due to fuel leaks in the pressure manifolds were experienced during September and October 1955. Extra instrumentation was required to investigate flight loads instrumentation drift. During November the aircraft conducted five flights for functional and instrumentation check outs. Three flight loads tests were conducted during December 1955, January 1956, and February 1956. The aircraft was then ferried to Convair-San Diego on 10 February 1956, for structural modification that would bring the aircraft up to design limit (100 percent) load specification. This consisted of beefing-up the nose, replacing bolts at spar #6, and applying buzz-fix; install differential linkage; and rework instrumentation for wing pressure manifold and in-flight zero calibration.

The aircraft was at Convair-San Diego undergoing modification and instrumentation for approximately four and one-half months. It was then ferried, combined with a shakedown flight, back to Edwards Air Force Base on 3 July 1956 to re-enter the flight test program.

The aircraft continued its flight loads program on a continuous basis since that time with excellent results. Testing consisted of Phase I flight tests under the following flight conditions: Symmetrical pullouts and pushovers, asymmetrical rolling pullouts, and finally high drag pushovers and pullouts. A three week inactivity period in October occurred when the nose structure was being repaired as a result of damage sustained during a landing with inoperable nose landing gear.

The symmetrical maneuvering portion (Parts I and II) of the program was completed in December 1956. The asymmetrical portion of Part I (required points) was completed during February 1957. Part II (supplemental points) asymmetrical testing was completed during March 1957. The aircraft was then placed on work

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Summary of Action (Continued)

status for installation of external fuel tanks and subsequently performed external tank structural demonstration and flight loads testing.

The aircraft continued structural demonstration with external fuel tanks empty and full through May and June. Upon completion of the flight loads program in late June, the aircraft was on an inactive status. The instrumentation necessary only for the flight loads program was removed and the aircraft was prepared for F-106A "g" limiter tests.

One flight to evaluate the Stromberg-Carlson and Crosley pitch "g" limiters was conducted the latter part of September. The data was transmitted to CV-SD for evaluation and the aircraft remained inactive during the evaluation period and while awaiting the return of a Crosley pitch "g" limiter unit. After the aircraft's return to flight status, eight flights were conducted completing the flight test evaluation of F-106 type pitch "g" limiters. The aircraft was delivered to San Diego in mid November for Broficon antenna modification.

Experimental Factory modification was initiated and essentially completed on the Broficon antenna installation when an Air Force decision was made to cancel any further work or tests on this particular aircraft. Authority for cancellation was received via CCN8N-2536 against Contract AF 33(600)-30169 on 18 December 1957. The aircraft was then removed to a storage area in San Diego awaiting Air Force disposition.

Future Activity

Awaiting Air Force disposition.

Chronological History

15 August 1955 - Ferried from Convair-San Diego to Edwards Air Force Base
25 August 1955 - Initial flight test, for flight test purposes
30 August 1955 - Conditionally accepted by the U. S. Air Force
10 February 1956 - Ferried from Edwards AFB to Convair-San Diego
3 July 1956 - Ferried from Convair-San Diego to Edwards Air Force Base
14 December 1956 - Completed Part I Symmetrical Flight Loads testing
13 February 1957 - Asymmetric demonstration completed
8 March 1957 - Supplemental flight loads conditions completed
27 June 1957 - Completed flight loads testing with external tanks
15 November 1957 - Completed pitch "g" limiter testing-ferried to San Diego
18 December 1957 - Broficon antenna modification and subsequent flight tests cancelled.

[redacted]

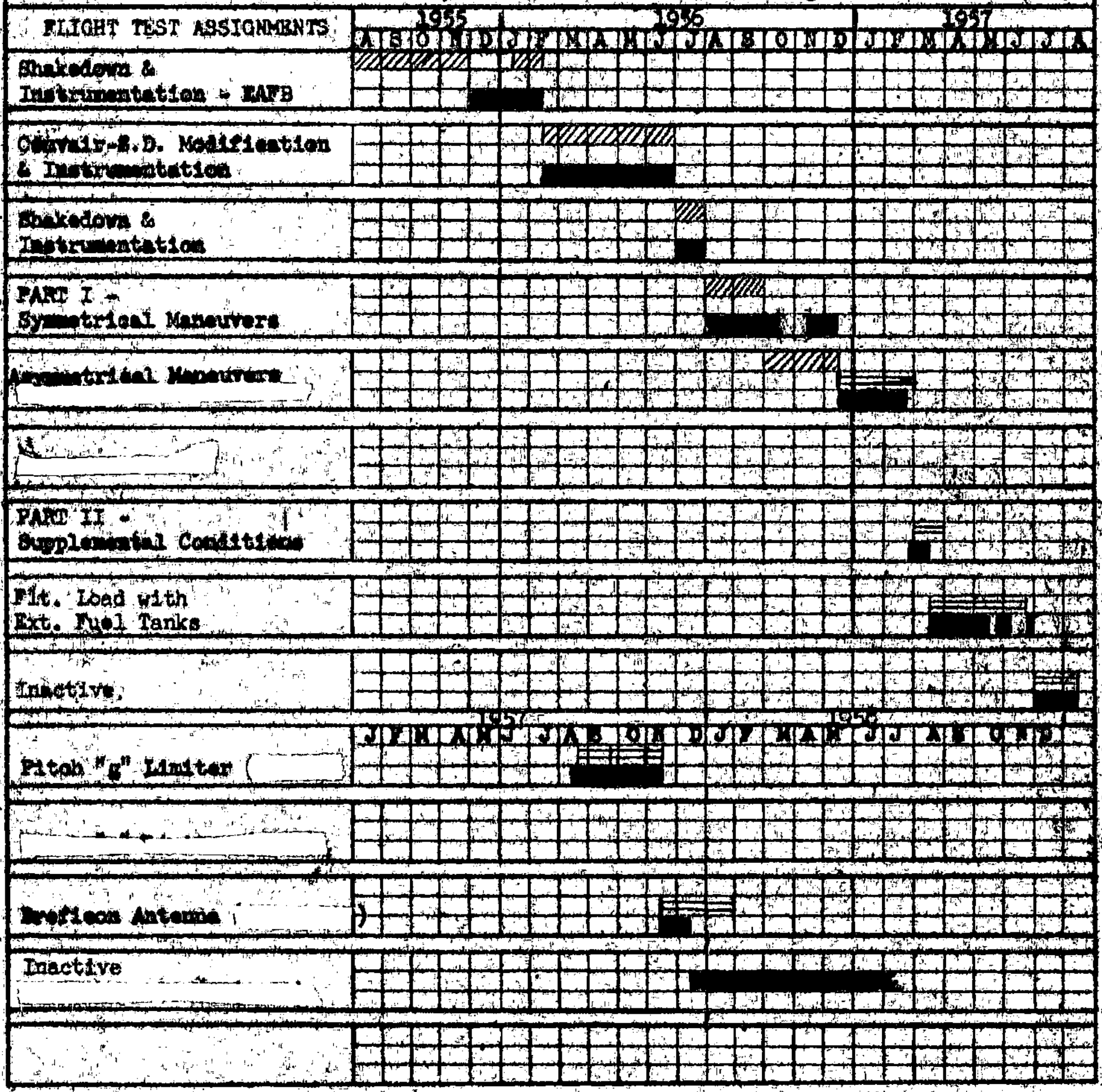
[redacted]

Convair Report No.:
ZC-8-122, Rev. #2, ZC-8-289
(Flight Test Program)

SUMMARY OF ACTIVITY

S/N 53-1794

Primary Assignment: 8-10 Stability & Control - Structural: Flight Loads



Original Planning ▨ Revised Planning ☰ Activity ■

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1797

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-626, dated 31 October 1956, is quoted as follows: "To be utilized to demonstrate the Rocket Type MB-1 extension armament system for the F-102A airplane".

Summary of Activity

In the endeavor to accomplish the test objectives, tests on missile bay door flutter investigation, armament hardware checks, main landing gear checks, and MB-1 rocket separation and trajectory have been completed. In the latter instance, the modified MB-1 rockets were launched from an "extended" position for the primary purpose of obtaining separation, trajectory, and engine response to rocket exhaust dust. The aircraft was ferried from Holloman Air Force Base to Convair-San Diego on 28 September 1956, where it underwent XMG-14 and instrumentation wiring installation and then to Hughes Aircraft Company where the XMG-14 AWCS installation was completed.

The aircraft was ferried from Hughes Aircraft Company, Culver City, to Holloman Air Force Base the first of February. The aircraft was laid up during part of February as a result of rail and structural damage incurred when the rocket was retracted into the missile bay without snubbing during rocket clearance checks. During the last part of February, instrumentation was converted to XMG-14 miss evaluation.

Testing during March and part of April consisted of radar, armament, instrumentation, and aircraft check flights. MB-1 rocket launching tests were conducted the middle of the month. The remainder of the rocket firing program was cancelled. The XMG-14 radar system was converted to the ejection system configuration for radar miss evaluation the last week in April.

During May four (4) flights were conducted to obtain a strike camera installation check and radar evaluation. Several radar lock-ons were obtained but could not be maintained. It was decided at the Holloman Armament Conference, 23 May 1957, to cancel the flight test program. The FCS was removed, aircraft weighed, ballasted, and inspected, and the aircraft was ferried to Palmdale for final acceptance.

Chronological History

21 January 1956 - Ferried from Convair-San Diego to Edwards Air Force Base
21 January 1956 - Conditionally accepted by the U.S. Air Force
10 February 1956 - Ferried from Edwards Air Force Base to Holloman Air Force Base
12 March 1956 - Initial flight test conducted
28 September 1956 - Ferried from Holloman Air Force Base to Convair-San Diego
1 February 1957 - Ferried from Culver City to Holloman Air Force Base
15 May 1957 - Completed flight test program.

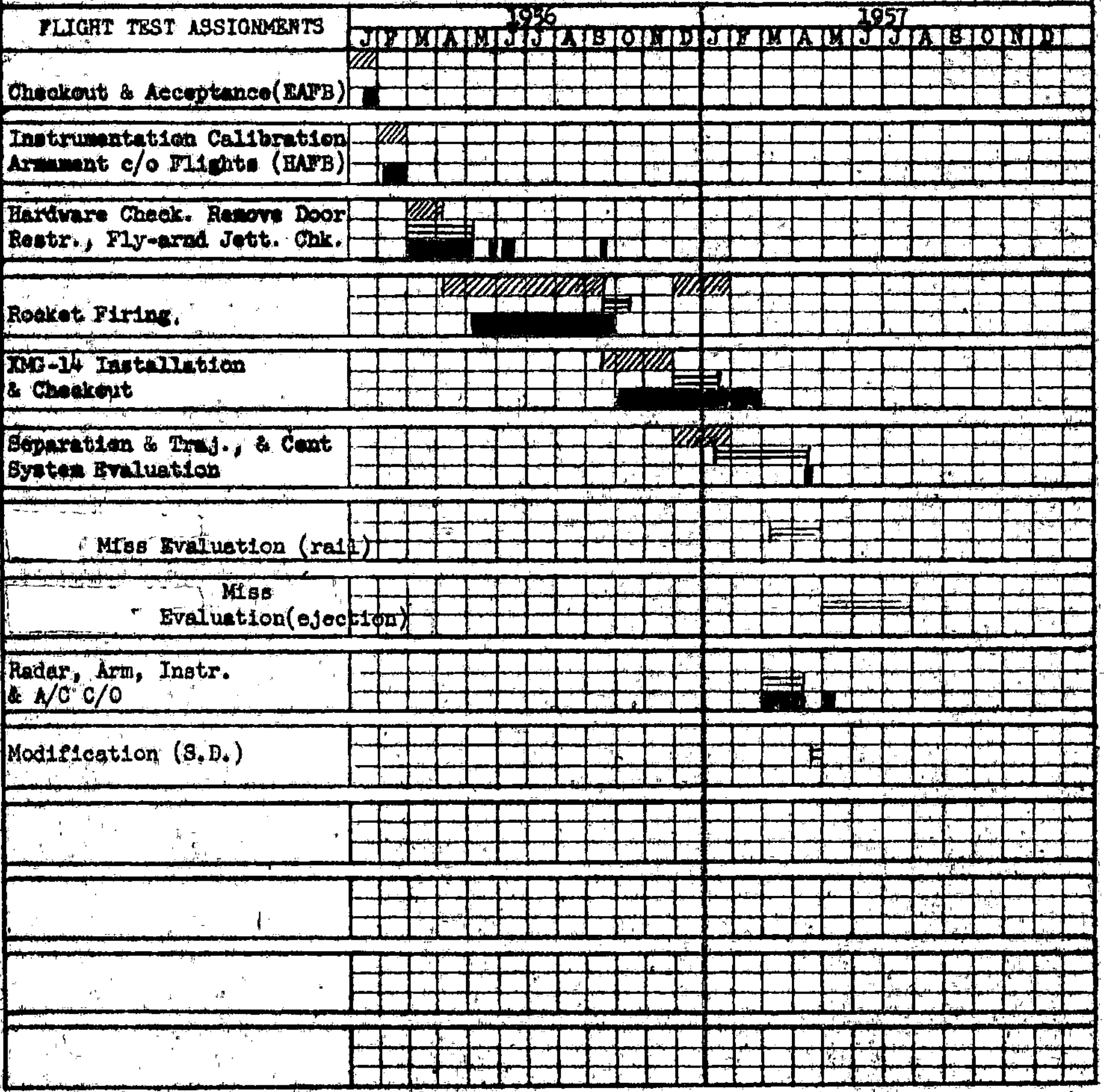
[redacted]

Convair Report No.:
ZC-8-166 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 53-1797

Primary Assignment: 8-10 Auxiliary Sys. Operation-Armament Development, Rocket MB-1



Original Planning ▨ Revised Planning ☰ Activity ■

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1798

Primary Assignment

Aircraft entered the Convair Flight Test Program during August 1955. The formal statement of the flight test assignment, as presented in Contract AF 33(600)-5942, as amended in accordance with the terms of negotiations conducted at the Air Materiel Command during the week of 11 June 1956, for review and re-estimate of the F/IF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 31 October 1956, is quoted as follows: "Period - 14 Maintenance Months". "To be utilized for antenna test (including tests with external fuel tanks, installed) a control system and component development and specification compliance tests.". The program (ZC-8-107), for this aircraft was conducted at Edwards AFB, and is graphically illustrated, as planned and performed, on the page following this summary report.

Summary of Activity

Aircraft was ferried from Convair-San Diego to Convair-Palmdale on 9 August 1955. The United States Air Force conditionally accepted the aircraft on 31 August 1955. Aircraft was then ferried to Edwards AFB to enter the F-102 Flight Test Program on 10 September 1955. The initial flight, for flight test purposes, was conducted on 22 September 1955.

During the 15 months the aircraft has been in the flight test program, it has been utilized for flight testing the following primary phases of its program.

Antenna and Controls

Antenna testing - AN/ARN-14 Omnidirectional Antenna, AN/ARC-34 UHF Communications, Data Link Communications, APX-27 Air-to-Air IFF, AN-APX-6 Air-to-Ground IFF, and AN/ARN-12 Marker Beacon. The Addendum 1 - AN/APX-6 Antenna System Study (F-106A), and Addendum 2 - AN/ARN-18 Glide Path Antenna System Test were partially completed.

Controls Testing, Phase I, Test Items 1 - Ground Tests; Test Items 2 - Control System Flight Evaluation; Test Item 3- System development including components development; Test Item 4 - Stability and Control. Addendum Items include, 1- Production Air Speed Calibration; 2- Engine Fuel Control Evaluation; 3-O.A.T. Calibration; 4- Ram Air Turbine check-out; 5- Drag chute deployment time; 6- Reduced rudder steering forces and 7- Landing Gear Loads tests.

The antenna program was completed during September and October 1955. Flights were then conducted for check-out of specification compliance changes, pilot assist, pitch damper, yaw damper, turn coordinator, and air speed and outside air temperature calibration, thus completing the initial flight evaluation of the control system. Trim servo qualitative tests, and the ground and flight testing of the HEP

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S/N 53-1798 Summary of Activity (Continued)

(Hydraulic Elevon Package) valve were completed satisfactorily. In the latter instance, the installation of the HEP valve produced an appreciable improvement in the performance of the direct manual and manual mode elevon system.

The initial assignment of this aircraft was the flight determination of navigational and communications antenna radiation patterns. This program, with the exception of the portion involving external tanks, was satisfactorily completed by the end of October 1955. The airplane was then scheduled for an extended modification period in preparation for its principal assign,ent of control system evaluation and development.

The testing of the flight control system started with flight 19 in February 1956, and continued, without major interruptions, through September. The program consisted of two major phases: an exhaustive evaluation of the system and its sub-systems; followed by a development phase. During the first phase, complete flight investigations of the performance of the Direct Manual System, the Feel and Trim Systems, the Turn Coordinator, Pitch and Yaw Dampers and the Convair Pilot Assist System were conducted. Ground tests to determine system responses were also conducted. The final antenna tests to determine external tank effects on radiation patterns were conducted concurrently with the flight evaluation.

In the course of the second, or development phase, gain settings of the stability augmentation system were optimized and the final F-102A control system configuration established. The Hydraulic Elevon Package (HEP)valve was tested and approved for production and considerable testing was done on the feel and trim systems, culminating in the CONTR 110-8 Snelley Unit and the production trim servo system. Test data were obtained for the design of the aileron control differential linkage and the longitudinal differential linkage was also evaluated.

During the final months of the program, emphasis was placed on the development of the electronic pitch "g" limiter, the investigation of transonic buffet and lateral-directional instability, and the determination of the aft C.G. limit for satisfactory flying qualities. Stability and control data for specification compliance proofing was also collected, as was data on landing gear air loads and system operation. In the last weeks, the electronic pitch g limiter was cleared for production incorporation and an aft C.G. takeoff limit of 30.5 percent M.A.C. established.

The aircraft completed its program during flight 93 on 28 September 1956. The last flight, number 94, was conducted for chase support for S/N 54-1353.

The aircraft was delivered to another test agency function in November 1957.

[redacted]

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[redacted]

S/N 53-1798 - (Continued)

Chronological History

9 August 1955 - Ferried from Convair-San Diego to Convair-Palmdale
31 August 1955 - Conditionally accepted by the U. S. Air Force
10 September 1955 - Ferried from Convair-Palmdale to Edwards Air Force Base
22 October 1955 - Initial flight test conducted.
28 September 1956 - Completed flight test program

[redacted]

UNCLASSIFIED

[redacted]

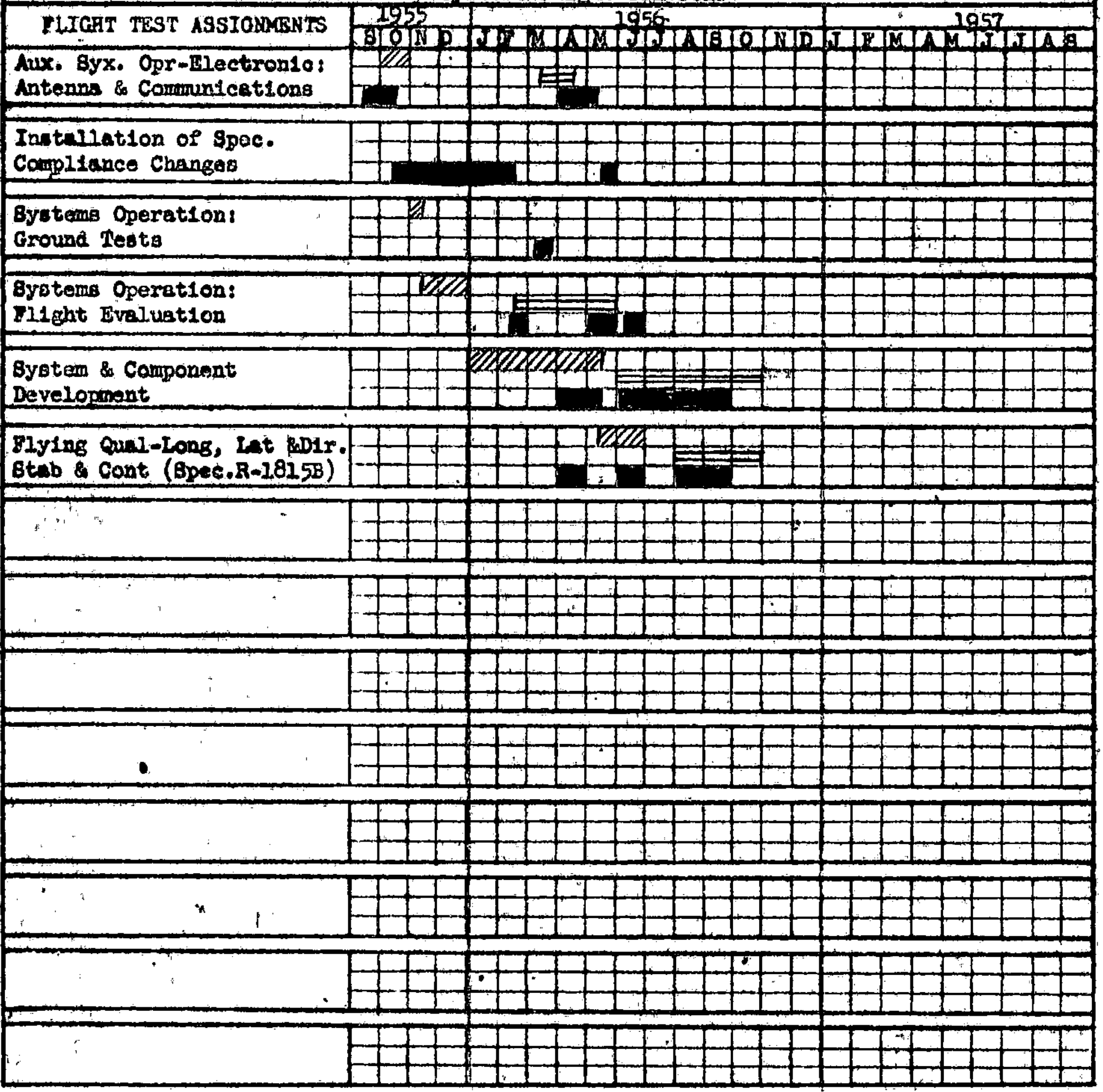
Page: 41A
Date: 1 Nov. 1956
Report No.: ZC-8-142-
Model: F-102A

Convair Report No.
ZC-8-167

SUMMARY OF ACTIVITY

S/N 53-1798

Primary Assignment: 8-10 Aircraft Systems-Flight Controls



Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1799

Primary Assignment

Aircraft entered the Convair Flight Test Program during September 1955. The formal statement of the flight test assignment, as presented in Contract AF 33(600)-5942, as amended in accordance with the terms of negotiations conducted at the Air Materiel Command during the week of 11 June 1956, for review and re-estimate of the F/TF-102A Flight Test Program, as presented in Convair Sales Order No. 33-1-626, dated 26 July 1956, is quoted as follows: "Period - 9 Maintenance Months". "To be utilized for Weapons System Demonstration which consists of check-out and calibration of weapon system and firing of instrumented missiles against parachute and drone targets. A final demonstration flight will be a salvo firing of tactical missiles against a drone target. Snap-up demonstration is also to be conducted.". The program (ZC-8-143), for this aircraft was conducted at Edwards AFB and Holloman AFB as follows:

Part I - Was conducted at Edwards AFB for check-out and weapon system calibration tests.

Part II - Was conducted at Holloman Air Development Center for demonstration firing of instrumented missiles against parachute and drone targets. The final demonstration flight was a salvo firing of tactical missiles against a drone target. Snap-up demonstration also was conducted.

The program is graphically illustrated, as planned and performed, on the page following this summary report.

Activities for the overall flight test program of this aircraft were accomplished jointly by Convair and Hughes.

Summary of Activity

Aircraft was ferried from Convair-San Diego to Convair-Palmdale on 22 August 1955. Aircraft was then ferried to Edwards AFB on 21 September 1955, to enter the F-102 Flight Test Program. The initial flight, for flight test purposes, was conducted on 23 September 1955. The United States Air Force conditionally accepted the aircraft on 30 September 1955. Aircraft was ferried from Edwards AFB to Convair-Palmdale on 5 October 1955 for installation of the MG-3 System and instrumentation, and returned to Edwards AFB on 4 November 1955. It was then placed on a ground work status on 7 November 1955 for the following work items: install and calibrate instrumentation for the WSD flight test program, and accomplish specification conformance changes.

Aircraft actively commenced its flight testing again on 6 January 1956, as flight No. 6, for a functional test flight, and MG-3 radar and instrumentation check-out. During its ground and flight testing at Edwards it completed the following test items:

[redacted]

[redacted]

S/N 53-1799 Summary of Activity (Continued)

Preparation and Equipment Check-out - EAFB

Flight Test Items 1 - Fire control system cooling, main landing gear side brace load test; experimental determination of launcher restrictor size; installation and check-out of test instrumentation; calibration, harmonization, and ground check-out of Fire Control System; installation of strike camera, missile rail trunnion mislocation tests and correction; shakedown flights for aircraft, FCS and T.M. missiles; simulated attacks to check complete system operation with missiles; airspeed calibration and air data computer check; miss evaluation - simulated attacks co-altitude; snap-up performance evaluation; and photo panel installation.

Hughes pilots performed several snap-up maneuvers, establishing the aircraft's control parameters. One Convair flight was made to obtain data for optimum snap-up operation. These maneuvers were conducted without difficulty to the aircraft's systems or engine.

The aircraft was then ferried from Edwards AFB to Holloman AFB on 26 March 1956. During its ground and flight testing at Holloman it completed the following items:

Weapons System Demonstration - HAFB

Flight Test Items 2 - Ground system check-out with missiles; co-altitude system verification-single missile firings; co-altitude salvo firing-instrumented missiles; co-altitude salvo firing-tactical missiles; snap-up performance flights; snap-up system verification and miss-evaluation; snap-up salvo firings; and production missile range rate servo evaluation.

The aircraft successfully completed co-altitude firings at parachute and drone targets, and then proceeded with WSD flights against QB-17 drones. In the latter instance one missile without a "warhead" passed through the vertical fin of one of the drones, and in another instance a drone was completely destroyed with a missile containing a "warhead". The fire control system snap-up and co-altitude tests against balloon targets were successfully concluded during the final stages of this aircraft's flight test program, as well as 106-box testing. Conclusions were that the GAR-1 missiles could be successfully fired from the aircraft during snap-up maneuvers, and guide properly to targets located as high as 17,000 above the launching platform, and that co-altitude firings (to determine the ability of the weapon system to fire missiles at the maximum level flight velocity of the interceptor), could be successfully accomplished.

The aircraft completed its primary flight test program at the conclusion of flight 71 on 2 August 1956. It was then utilized for chase support missions until being ferried to Convair-San Diego on 4 September 1956 during flight 74. Aircraft then underwent a 263 equipment inspection check prior to being transferred to the Hughes Aircraft Company.

[redacted]

[redacted]

S/N 53-1799 Summary of Activity - (Continued)

Chronological History:

22 August 1955 - Ferried from Convair-San Diego to Convair-Palmdale
21 September 1955 - Ferried from Convair-Palmdale to Edwards Air Force Base
23 September 1955 - Initial flight test conducted
30 September 1955 - Conditionally accepted by the U.S. Air Force
5 October 1955 - Ferried from Edwards Air Force Base to Convair-Palmdale
4 November 1955 - Ferried from Convair-Palmdale to Edwards Air Force Base
6 January 1956 - Again commenced flight testing
26 March 1956 - Ferried from Edwards Air Force Base to Holloman Air Force Base
2 August 1956 - Completed flight test program at the Termination of flight 71
4 September 1956 - Ferried from HAFB to Convair-San Diego

[redacted]

Page: 45A
Date: 1 NOV 56
Report No.: ZC-8-142
Model: F-102A

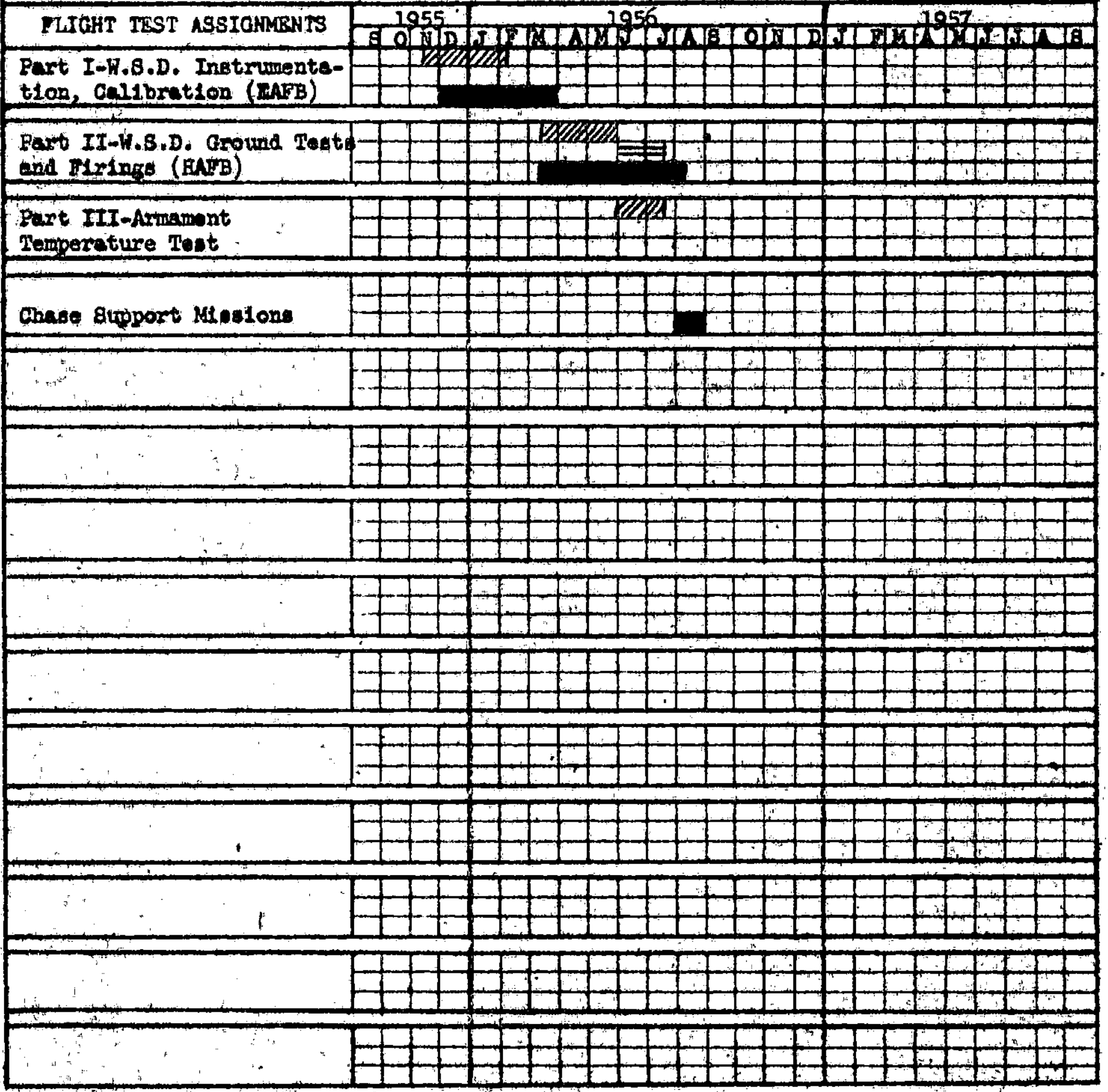
[redacted]

Convair Report No.
ZC-8-143

SUMMARY OF ACTIVITY

S/N 53-1799

Primary Assignment: 8-10 Auxiliary Systems Operation - Weapons System Demonstration



Original Planning ▨ Revised Planning ▤ Activity ■

[redacted]

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1806

Primary Assignment

The flight test assignment for this aircraft as presented in Convair Sales Order No. 33-1-626, dated 31 October 1956, is quoted as follows: "To be utilized to demonstrate the Rocket Type MB-1 ejection armament system for the F-102A airplane".

Summary of Activity

The missile bay door flutter investigation tests were conducted throughout April 1956. Data obtained from these tests resulted in the aircraft's armament bay doors being cleared to the test program requirements of Mach 1.2 at 20,000 feet. During May and July 1956 the aircraft was primarily utilized for rocket ejection system development, utilizing six out of twelve originally programmed, inert rockets to determine operation of the aircraft's rocket ejection gear. Data obtained from these tests drew a conclusion that gear (modified) was satisfactory for rocket ejections. As a result of this the balance of six inert rocket releases were postponed. Consequently, this aircraft was returned to Convair-San Diego for XMG-14 Aircraft and Weapon Control System wiring and instrumentation wiring installation.

The aircraft was ferried from Convair-San Diego to Holloman Air Force Base on 20 September 1956 to re-enter the flight test program. After instrumentation and flight controls checks the aircraft was used to conduct air inert rocket ejection tests. During December the first in-flight live motored MB-1 rocket was ejected successfully.

In January a decision was made that the Patushin ejection system latch mechanism was unsatisfactory and therefore, Convair designed, fabricated, and qualified at San Diego a new frictionless brake overcenter release mechanism.

During the March-April period a total of 10 MB-1 live motored rockets were ejection launched. Trajectories appear to be within the limits set for the program. The primary problems encountered have been rocket fin actuation failures and EJECT-10 system maintenance. Ejection system operating pressures have been brought into an acceptable pressure band by the Convair brake-over-center latch mechanism. An engine change was begun the last week in April due to an engine accessory drive failure. Live motored MB-1 ejections were continued through May and June. The aircraft was down for a short period the middle of June for negative "g" valve modification to the Sundstrand Unit and a 50 hour inspection.

After an engine change the first week of July, rocket launches were continued. During an attempt to determine the cause of the malfunction of the EJECT-10 system which occurred the latter part of July, the aircraft was inactive for a three week

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Brief Summary of Aircraft Activity S/N 53-1806 - (Continued)

period. Pit firings were performed during the investigation. The aircraft was down once again the latter part of August for an engine change which was necessitated as a result of excessive tail pipe temperatures during a practice snap-up maneuver flight. Testing was again resumed and rocket launches and snap-up maneuver investigation flights were performed. One flight for evaluation of ground cameras and airborne instrumentation for snap-up was flown in early September. The flight test task was then terminated by mutual agreement with AMC. The aircraft was ferried to Palmdale on 10 September 1957.

Chronological History

21 February 1956 - Ferried from Convair-San Diego to Edwards Air Force Base
29 February 1956 - Conditionally accepted by the U.S. Air Force
17 March 1956 - Ferried from Edwards Air Force Base to Convair-San Diego
5 April 1956 - Ferried from Convair-San Diego to Holloman Air Force Base
6 April 1956 - Initial armament flight test conducted
13-14 July 1956 - Ferried from Holloman Air Force Base to Convair-San Diego
20 September 1956 - Ferried from Convair-San Diego to Holloman Air Force Base
12 December 1956 - First live motored MB-1 rocket ejected
9 September 1957 - Completed flight test program.

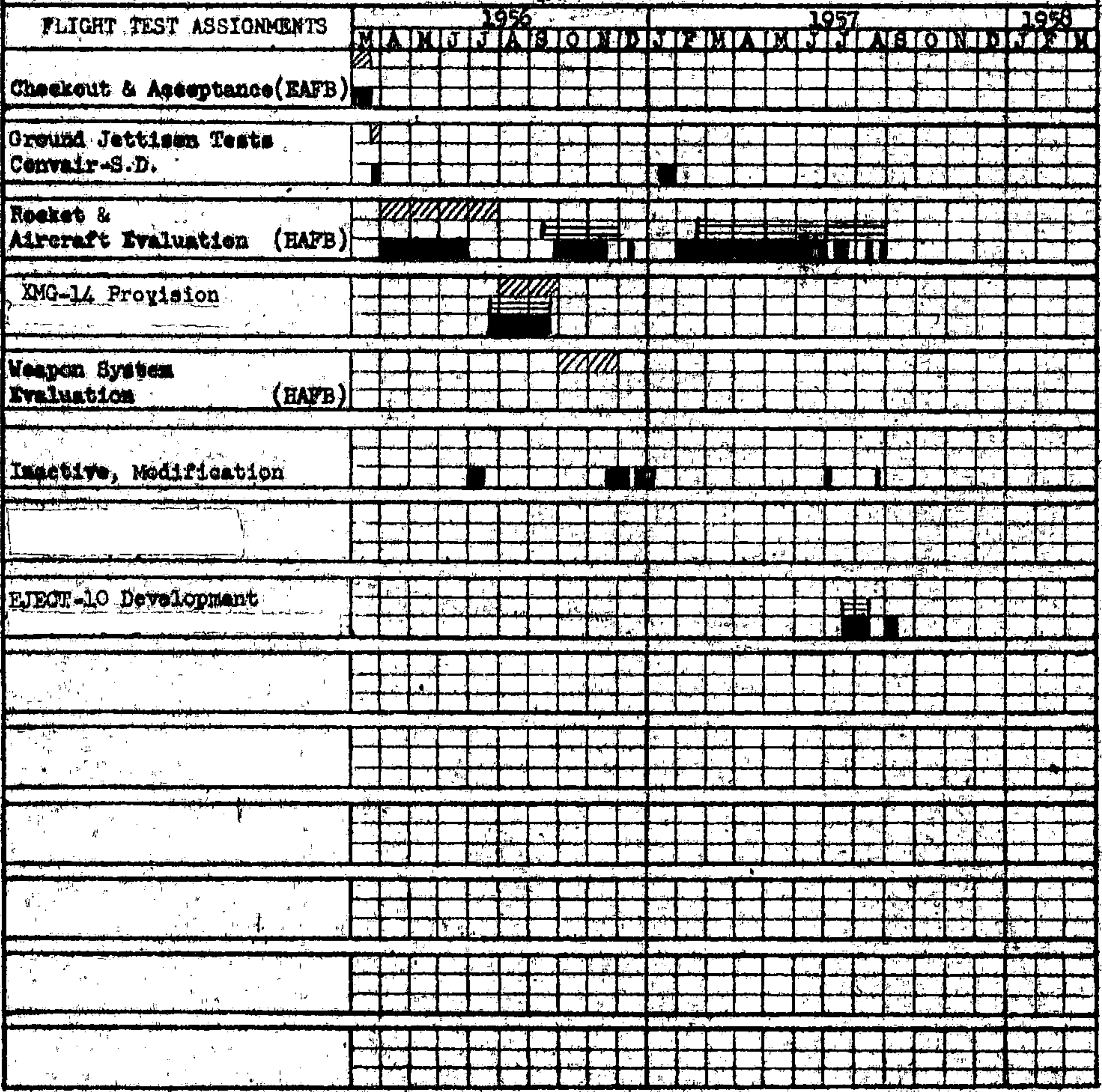
[redacted]

Convair Report No.:
ZC-8-168 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 53-1806

Primary Assignment: 8-10 Auxiliary System Operation-Armament Development Rocket MB-1



Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1808

Primary Assignment

Functional check of the Air Research Development Command Advanced Cockpit Instrument Display.

Summary of Activity

After undergoing instrumentation and cockpit instrument installation at San Diego, the aircraft was delivered to Edwards Air Force Base, 6 June 1957.

Numerous wiring changes were made on the ARDC Instrument Display (R & D Project) and the accompanying instrumentation. Test flights were conducted to determine the functioning of both cockpit instruments and instrumentation. Later the aircraft was put on a ground work status for instrument check out and for installation of manual slewing switches on several ARDC instruments.

Testing during the first part of July indicated that the manual slewing switches functioned well. The ARDC Advanced Instrument Display operated satisfactorily during the final functional test flight. The aircraft was ferried to WADC the latter part of July.

Chronological History

6 June 1957 - Ferry flight from San Diego to Edwards Air Force Base
15 July 1957 - Completed flight test program
23 July 1957 - Ferried to WADC

[redacted]

[redacted]

Page: 47A
Date: 1 Sept 1957
Report No.: ZC-8-142-
Model: TF-YF-F-102

Convair Report No.:
ZC-8-153

SUMMARY OF ACTIVITY

S/N 53-1808

Primary Assignment: ARDC Advanced Cockpit Instrument Display

FLIGHT TEST ASSIGNMENTS	1957							1958																		
	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J
ARDC Cockpit display	▨ ■	▨ ■																							

Original Planning ▨ **Revised Planning** ☰ **Activity** ■

[redacted]

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 53-1813

Primary Assignment

The flight test assignment for this aircraft as presented in Convair Sales Order No. 33-1-626, dated 31 October 1956, is quoted as follows: "To be utilized for flutter, directional stability and rolling performance tests with a gloved larger vertical fin installed and for constant speed drive negative "g" tests."

Upon completion of this program the aircraft was utilized for evaluation of side stick. The formal statement of this assignment, as presented in Contract Change Notice #222, dated 2 April 1956, to Contract AF 33(600)-5942, is quoted as follows: "The flight test task for F-102A airplane S/N 53-1813 as established by Item 15 is hereby revised to include the following: Contractor shall accomplish a side control stick test installation for evaluation by the Government and shall maintain the airplane during evaluation by the Government, etc.". Following the modification period, and a one-month functional check out by Convair pilots, the aircraft was scheduled for flight testing by the United States Air Force, during which time Convair would maintain the aircraft.

Summary of Activity

Testing during the primary program included: stability and control consisting of static directional stability, dynamic lateral-directional stability, and roll coupling including stability and control test points, and lateral response to rolling maneuvers; additional testing was conducted on elevon camera fairings, and the Sundstrand constant speed drive negative "g" valve. Roll tests were conducted by Convair and United States Air Force pilots at both subsonic and supersonic speeds at various altitudes. This program was concluded on 18 April 1956.

Upon completion of the side stick installation at Convair-San Diego, the aircraft was ferried to Edwards Air Force Base to re-enter the flight test program. Convair completed their functional flight testing of the installation during August 1956. United States Air Force pilots evaluated the side stick installation for a two-month period. Testing was also accomplished to evaluate revised landing light configurations.

[redacted]

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[redacted]

S/N 53-1813 - Continued

Chronological History

10 December 1955 - Ferried from Convair-San Diego to Edwards Air Force Base
10 December 1955 - Initial flight test conducted
29 December 1955 - Conditionally accepted by the U. S. Air Force
19 April 1956 - Ferried from Edwards Air Force Base to Convair-San Diego
20 July 1956 - Ferried from Convair-San Diego to Edwards Air Force Base
11 August 1956 - Convair completed functional flight testing of the F-106A Side Stick installation during Flight No. 57.
11 August 1956 - U. S. Air Force commenced flight testing of the F-106A Side Stick installation during Convair Flight No. 58 and USAF Flight No. 1
18 October 1956 - F-106A Side Stick installation testing completed
27 November 1956 - Revised landing light configuration completed - Completion of Convair testing.

[redacted]

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1317

Primary Assignment

The assignment for this aircraft as covered in CCN 275 dated 24 June 1957, is angle of attack, stability and control, and structural demonstration with the Case XX wing.

Summary of Activity

Air Force evaluation flights were conducted shortly after the aircraft's arrival at Edwards Air Force Base the latter part of June. This evaluation continued through to the latter part of July. The airdraft was on a stand-by status waiting for the return of the engine from aircraft S/N 56-1001 (which was used during Air Force performance comparisons).

Convair initiated angle of attack testing the middle of August and completed this phase of testing after two flights (pending data analysis). Instrumentation for stability and control testing was then activated and testing was conducted the remainder of August.

Stability and control were completed in the early part of September, however, due to instrumentation malfunctions portions of this testing had to be repeated. No flights were conducted the middle of September; the aircraft was on work status for repair of cracked leading edge spar rail. Upon completion of the work task, stability and control testing was resumed, this time concurrently with the structural demonstration flights. The aircraft was on instrumentation work status the first half of October. Stability and control testing and structural demonstration flights were conducted through October and completed 19 November 1957. Testing included 50, 80, and 100 percent bank to bank rolls, rudder kicks, pull ups and pushovers, etc. Additional stability and control tests and turn coordinator control tests with the lag roll deleted were completed during November on a by-product basis.

The aircraft was de-instrumented and readied for disposition, leaving Edwards AFB 13 December 1957.

Chronological History

28 June 1957 - Ferried to Edwards Air Force Base
23 July 1957 - Completed Air Force evaluation
16 October 1957 - Completed stability and control testing
19 November 1957 - Completed entire flight test program.

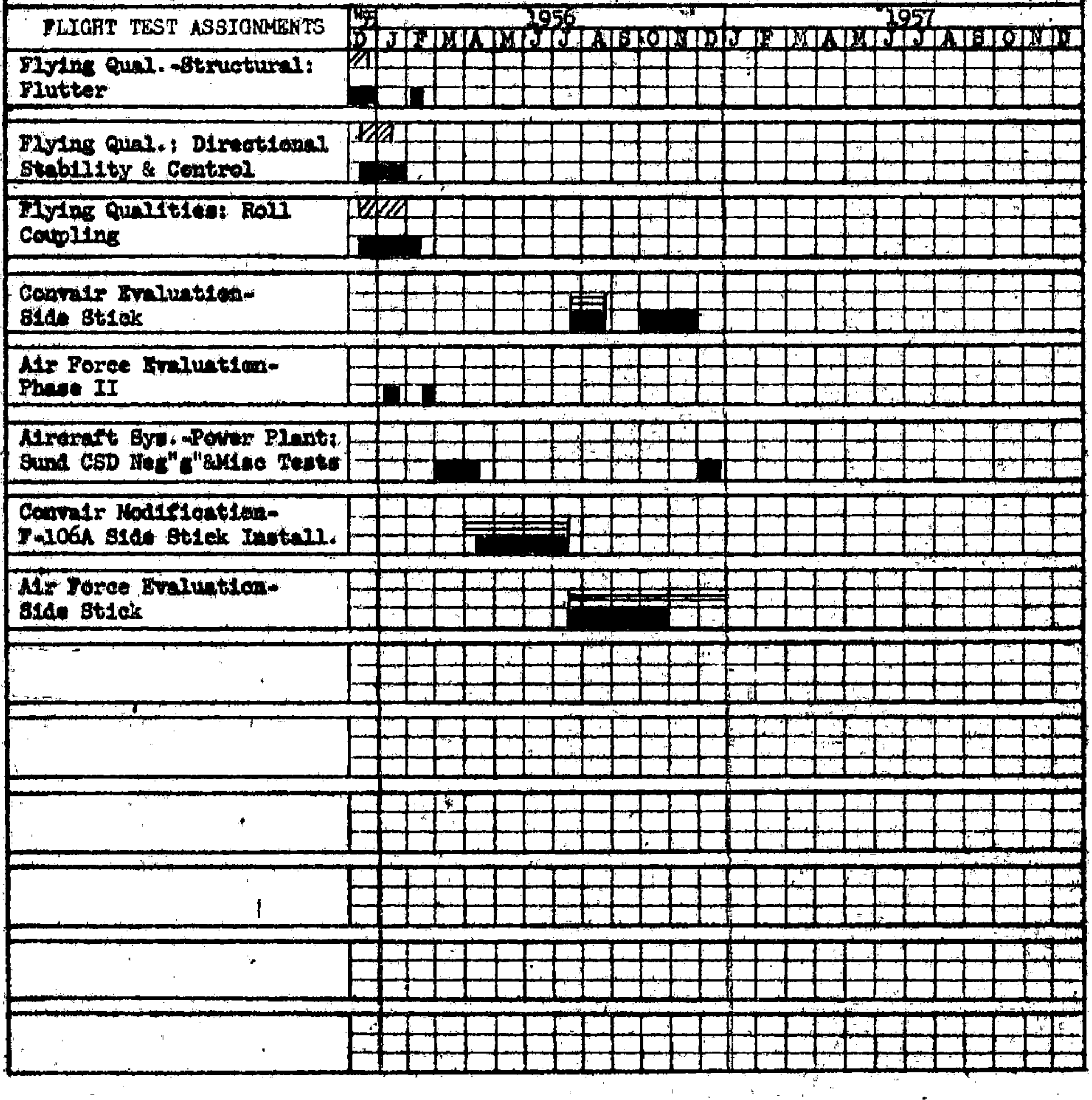
[redacted]

Convair Report No.:
ZC-8-171 (Large Fin)
ZC-8-197 (Side Stick) Flt. Test Prog.)

SUMMARY OF ACTIVITY

S/N 53-1813

Primary Assignment: 8-10 Large Fin (Gloved) - F-106A Side Stick



Original Planning ▨ Revised Planning ▤ Activity ■

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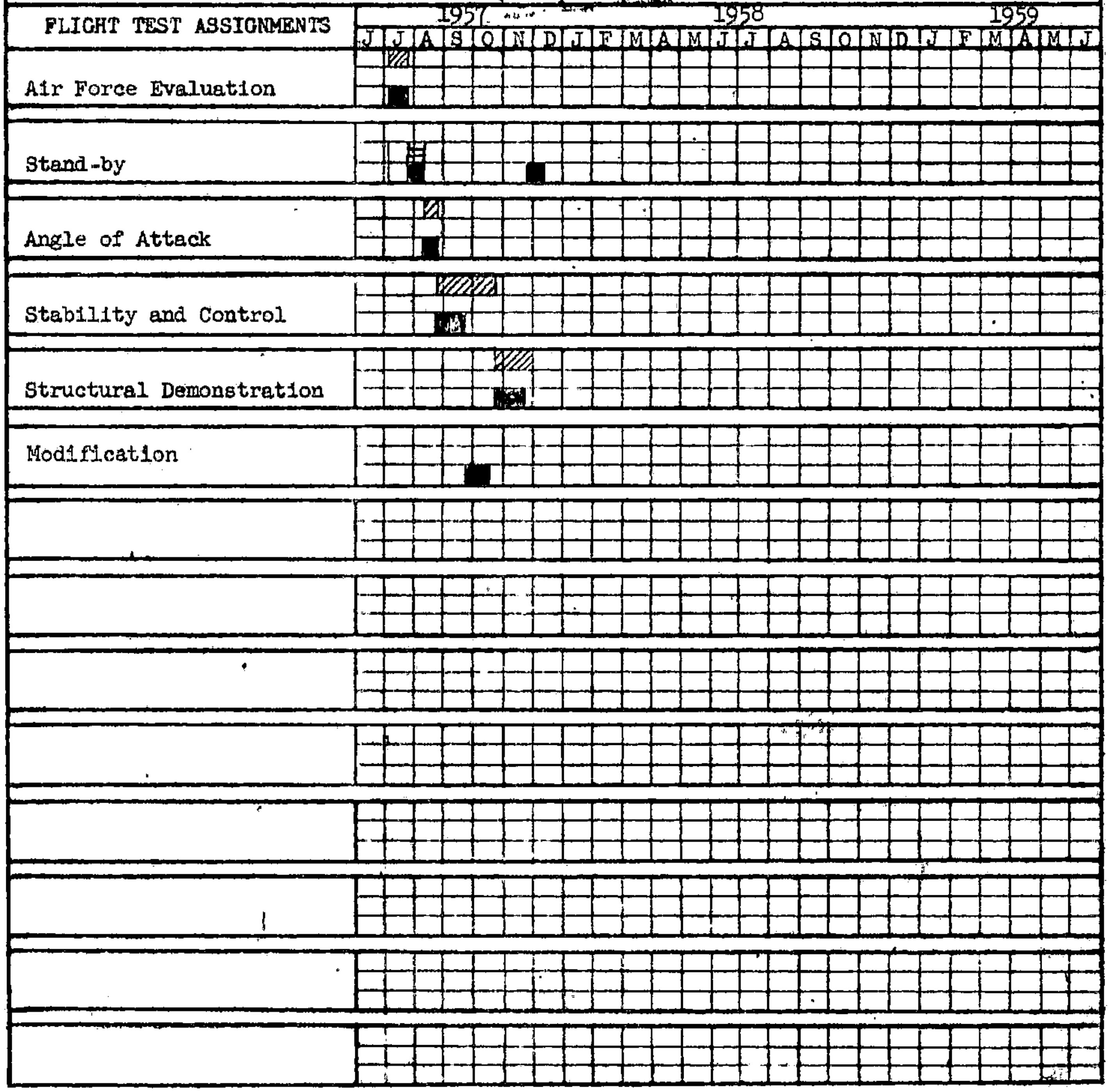
Convair Report No.:
ZC-8-276

SUMMARY OF ACTIVITY

S/N 54-1317

Primary Assignment: Case XX Wing Flight Demonstration Program



Original Planning ▨ Revised Planning ☰ Activity ■

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[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1351

Primary Assignment

The assignment for this aircraft as presented in Convair Sales Order No. 33-1-511, dated 26 July 1956, is quoted as follows: "To be used for investigation of buffet, performance evaluation, and Phase I, Phase II, and Phase III testing, including anti-icing, rain clearing and production system tests."

Summary of Activity

This aircraft was the first TF-102A aircraft to enter the flight test program. A buffet condition was experienced between approximately .85 and .90 Mach. Subsequent testing was devoted to investigation of the cause and correction of this condition. Various fixes were tried, including balsa wedges on the aircraft's exterior in the canopy area, boundary layer plow slats, and many configurations of vortex generators. With 1 1/2" alternating vortex generators on the canopy leading edge, in combination with boundary layer slats between the ducts and the fuselage, the airplane was considered to be satisfactory for conducting performance evaluation. As a result of additional buffet tests, buffet was further decreased by two rows of vortex generators. (One row aft and one forward of the transparent canopy area. These vortex generators, when properly sized and spaced over the canopy frame, reduced buffeting to an acceptable level.)

The Air Force conducted a comparative flight evaluation of the original canopy configuration of this aircraft against the cut-down canopy configuration of TF-102A S/N 54-1353. As a result of this evaluation, the original canopy configuration of this aircraft was selected.

A minimum stability and control investigation of the flight control damper system, positive quadrant roll coupling characteristics, emergency landing gear extension time with new restrictors, and ram air pressure and temperature tests were conducted, intermitently, since the aircraft entered the program. On 21 June 1956, the aircraft was placed on a ground work status for modification to incorporate specification conformance changes, which were completed at the end of October 1956.

Flights were then conducted to determine the cause and cure for aircraft buffet. Flights were also conducted for radome glycol anti-ice flow characteristics. The purpose of these tests was to demonstrate satisfactory distribution of the radome anti-icing fluid. The aircraft was ferried to San Diego the first part of December. Windshield anti-ice rain clearing, and windshield NESA glass tests were conducted. Part of the time at San Diego was spent correcting discrepancies discovered during rain clearing and anti-icing MPS checks. The aircraft was returned to Edwards AFB the last day of January and remained on a ground work status through February.

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[redacted]

S/N 54-1351 (Continued)

Testing conducted and completed during the March-April period includes Hydraulic system tests, relative wind transducer tests, airspeed calibration with production nose boom tests and Radome Anti-Ice tests.

Chronological History

17 October 1955 - Ferried from Convair-San Diego to Edwards Air Force Base
31 October 1955 - Initial flight test conducted
30 November 1955 - Finally accepted by the U. S. Air Force
11-19 June 1956 - Phase II testing by the U. S. Air Force
22 June 1956 to date - Modification at Edwards Air Force Base for S.C.C. installation and instrumentation
28 November 1956 - Completed buffet testing
4 December 1956 - Ferried from Edwards Air Force Base to Convair-San Diego
31 January 1957 - Ferried from Convair-San Diego to Edwards Air Force Base
18 April 1957 - Completed Hydraulic Tests
30 April 1957 - Completed Flight Test program.

[redacted]

[redacted]

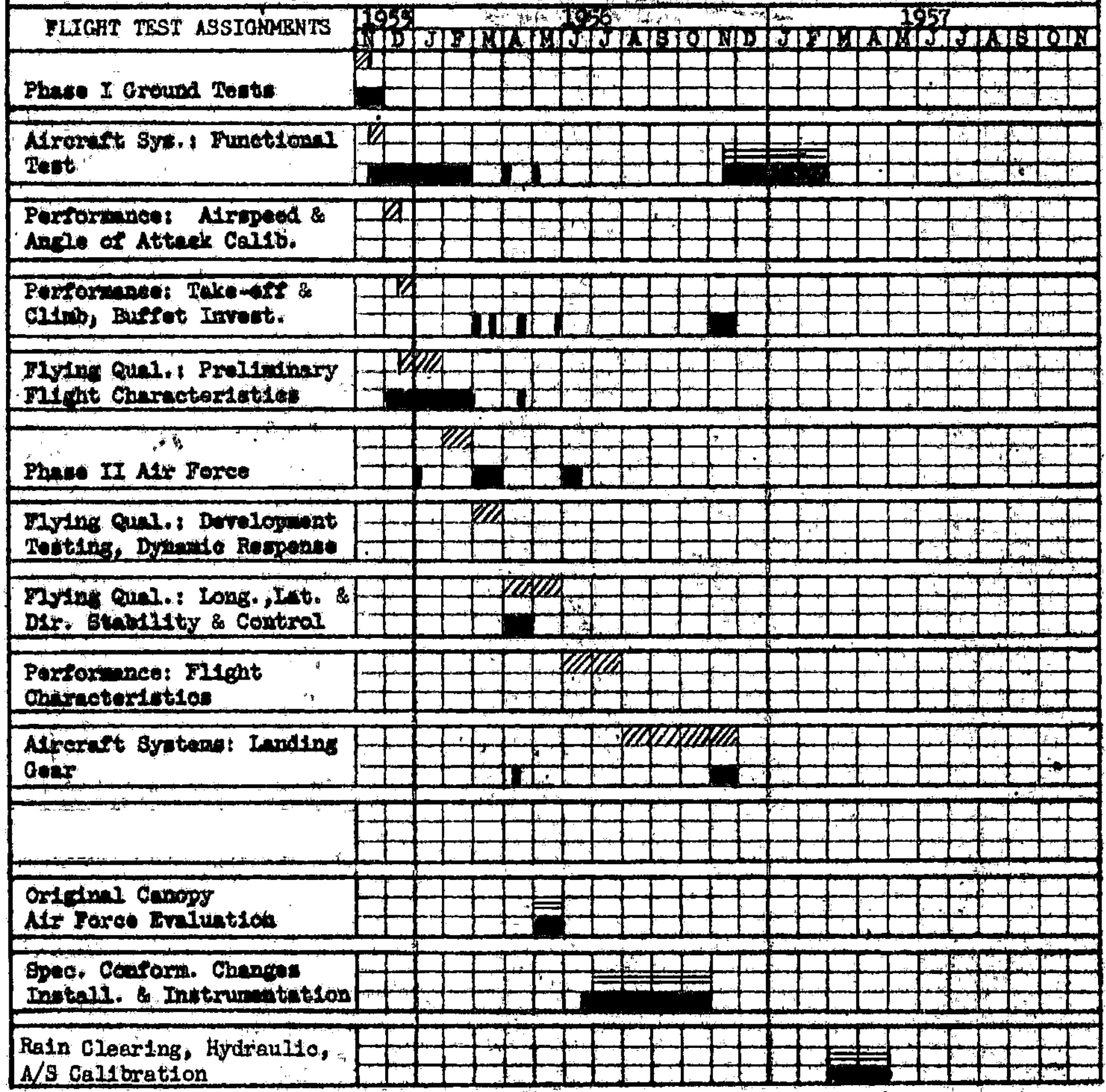
Page: 53A
Date: 1 May 1957
Report No.: ZC-8-142-
Model: TF-102A

Convair Report No.:
ZC-8-140 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 54-1351

Primary Assignment: Phase III



Original Planning ▨ Revised Planning ☰ Activity ■

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1353

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-511, dated 26 July 1956, is quoted as follows: "To be utilized for Phase I tests, Phase III roll and coupling test with original canopy and gloved tail, and external fuel tank tests."

Summary of Activity

This aircraft was primarily utilized to determine the effects of a cut-down canopy as compared to the original canopy configuration of TF 54-1351. During its initial flight, while being ferried to Edwards, a slight degree of buffeting was encountered, but this was a considerable improvement over S/N 53-1351's buffet characteristics. However, the pilot's visability from the cockpit, due to the cut-down canopy, was decreased appreciably. The slight buffeting characteristics of the aircraft was reduced by the addition of two rows of vortex generators placed fore and aft of the canopy transparent area, and the installation of boundary layer plow slats.

The U.S. Air Force conducted comparative flight evaluations between this aircraft (cut-down canopy), and S/N 54-1351 (original canopy), during May 1956. As a result of these evaluations the original canopy configuration was selected for production over the cut-down configuration. Other flight testing during this period included functional tests and pilot familiarization flights. Aircraft was ferried from Edwards Air Force Base to Convair-Palmdale for gloved fin and canopy modifications. These modifications were completed and the aircraft ferried from Convair-Palmdale to Edwards Air Force Base where testing for roll coupling was accomplished.

Flight testing during the November-December report period was limited to Air Force familiarization flights the beginning of November and a functional test flight at the end of December. The aircraft was primarily in a work status during this period for purposes of instrumentation calibration prior to Air Force Phase IV testing.

Chronological History

4 April 1956 - Initial test while being ferried from Convair-San Diego to Edwards
Air Force Base
29 May 1956 - Ferried from Edwards Air Force Base to Convair-Palmdale
31 August 1956- Ferried from Convair-Palmdale to Edwards Air Force Base.
25 October 1956 - Completed Roll Coupling testing.
31 October 1956 - Completed Convair portion of Flight Test Program.

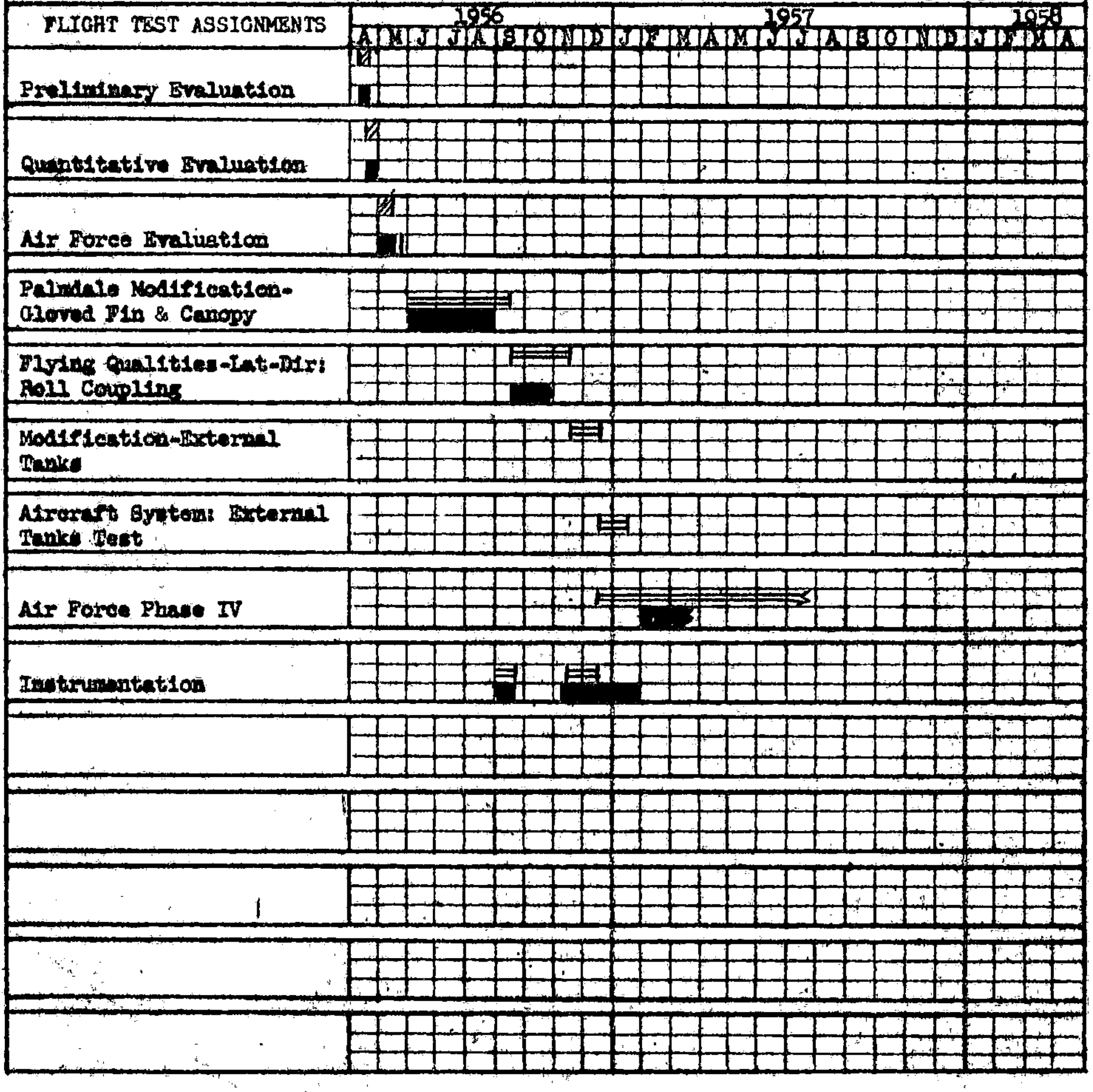
[redacted]

Convair Report No.:
ZC-8-183 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 54-1353

Primary Assignment: Flying Qualities-Lateral-Dir. Stab. & Cont.-Roll Coupling



Original Planning ▨ Revised Planning ☰ Activity ■

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1354

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-511, dated 26 July 1956, is quoted as follows: "To be utilized for armament testing, including MG-10T vibration environment tests and effect of armament on MG-10T system."

Summary of Activity

During its first month in the flight test program, this aircraft was primarily utilized for conducting pilot familiarization flights, in-flight armament hardware tests, and chase support.

Early in the test period, flights were conducted for engine response to missile exhaust gases. Flights were conducted for the purpose of a hi-speed missile launch to obtain data for verification of the firing capabilities and characteristics of the TF-102A (Design verification). Modification for Sundstrand Constant Speed Drive, initiated in October, was completed at the end of November.

A two-week period of missile separation and trajectory testing during negative "g" flights was concluded the middle of December. For the remainder of the month the aircraft was in modification for engine inlet duct anti-icing instrumentation installation. Instrumentation installations necessary for the anti-icing program were installed the first part of January. The anti-icing program was completed the middle of February. The 2.75-inch Rocket Program was initiated the last part of February 1957, and completed approximately one (1) month later after fifteen (15) flights. Test data were collected from 2.75-inch rocket jump angle firings, demonstration 24-rocket salvo firings, anti-icing tests, and angle of attack check points.

The aircraft was returned to San Diego the latter part of March for modification and MG-10T wiring provision installation. This task was completed and the aircraft was ferried to Palmdale the middle of August. MG-10T installation and checkout were completed and the aircraft was ferried to Holloman Air Force Base the latter part of September. Instrumentation was completed within a week and flights for 2.75-inch rocket jump angle data were then conducted. The 2.75-inch rocket program concluded with flight 74 the latter part of October. The aircraft was ferried to Fort Worth (test to tactical) 13 November 1957.

NOTE: The proposed "Effect of Armament Firing on MG-10T FCS" test period was cancelled in September 1957 by virtue of a Headquarters, USAF Directive to reduce expenditures. The 2.75" FFAR jump angle program had precedence at the time the test program was concluded.

[redacted]

[redacted]

Brief Summary of Aircraft Activity S/N 54-1354 - (Continued)

Chronological History

15 June 1956 - Aircraft ferried from Convair-San Diego to Convair-Palmdale
31 July 1956 - Final acceptance by the U.S. Air Force
7 August 1956 - Aircraft ferried from Convair-Palmdale to Holloman Air Force Base
7 August 1956 - Initial Flight test conducted
14 December 1956 - Completed negative missile firing test
12 February 1957 - Completed anti-icing program
1 May 1957 - Returned to San Diego for MG-10T provisions
13 August 1957 - Ferried to Palmdale for MG-10T installation and check out
20 September 1957 - Ferried to Holloman Air Force Base
23 October 1957 - Completed Flight test program.

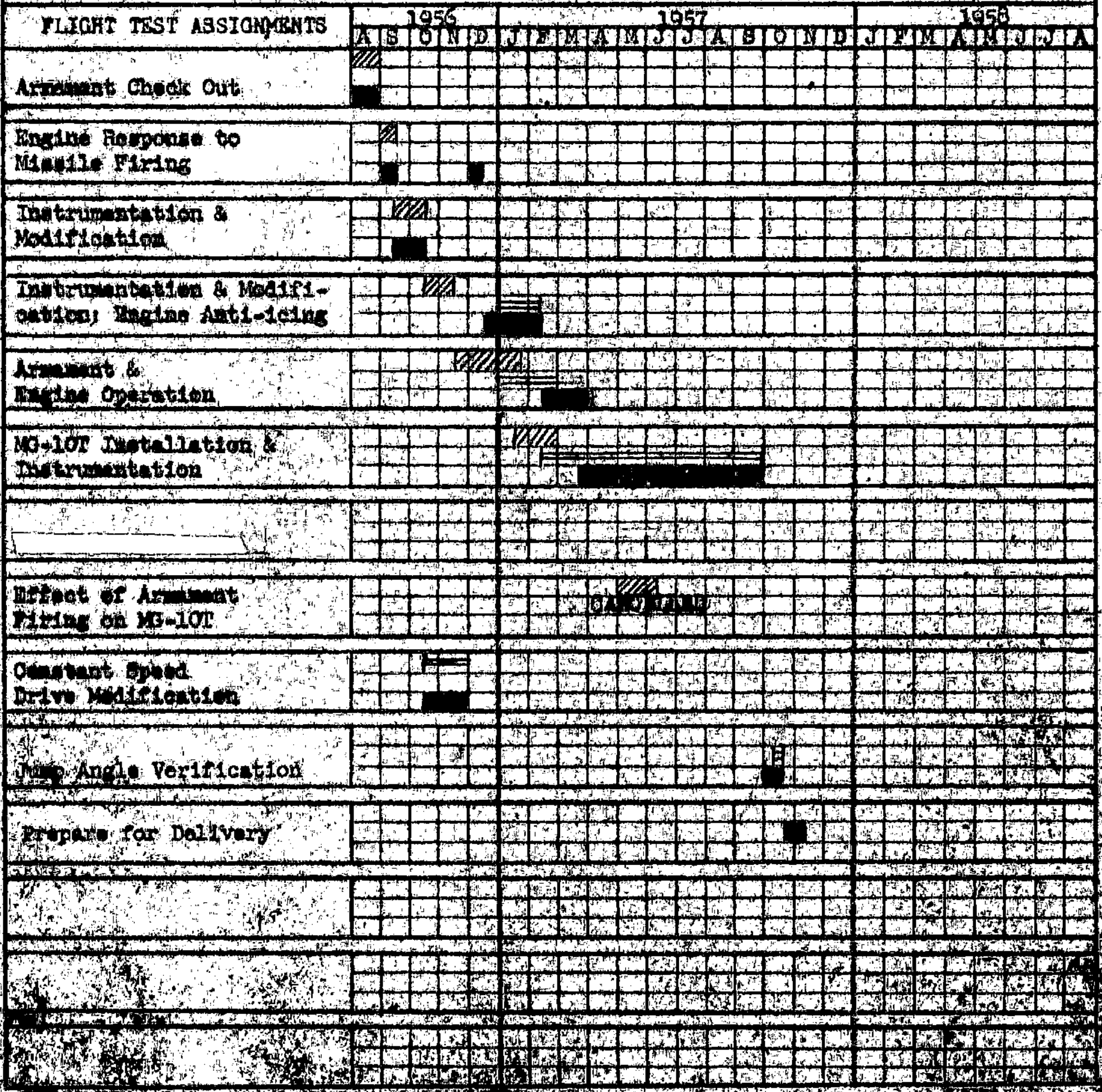
[redacted]

Convair Report No.:
ZC-8-141 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 54-1354

Primary Assignment: Auxiliary Systems Operation - Armament



Original Planning ▨ Revised Planning ☰ Actual ■

UNCLASSIFIED

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1361

Primary Assignment

As stated in Convair Sales Order No. 33-1-511, dated 26 July 1956, this aircraft is to be utilized for structural integrity demonstration. The testing and specifications to be complied with are as follows:

1. Structural demonstration
2. Cabin pressurization tests
3. Heat and ventilating (Specification AN-T-50A)
4. External Tank Tests (Specifications MIL-T-7478 and 1815B)

Summary of Activity

The aircraft arrived at Edwards Air Force Base the middle of March 1957 following five months lay-up at San Diego. The aircraft and instrumentation shakedown period lasted four weeks. Productive structural demonstration flights were initiated in April. During May and June thirteen (13) flights were conducted for symmetrical and asymmetrical structural demonstration. The symmetrical points were completed. The aircraft underwent thorough structural inspection early in the period to prove suitability for structural demonstration task. Late in the period the engine was replaced following indication of excessive engine temperature. Later, however, the discrepancy was isolated to instrumentation error.

Asymmetrical flight loads testing was resumed the first part of July. Two additional points remained to be flown for structural demonstration purposes. The aircraft was on a stand-by status during August and was turned over to AFFTC in September for Phase IV testing. USAF maintenance was used.

Following completion of Air Force Phase IV testing the latter part of December 1957, the aircraft was returned to Convair at Edwards AFB. Instrumentation for and flight demonstration of the additional points for structural demonstration were then conducted until mid-February 1958. A brief Air Force test period during which the missile bay doors were opened in flight (with external tanks installed) was conducted late February 1958. An inactive period preceded the large speed brake installation and instrumentation. Flight tests were subsequently conducted on the production and later, beefed-up speed brakes. The test program was completed mid-May 1958.

NOTE: As a result of a Headquarters, USAF directive to reduce expenditures, the Heat and Ventilating tests planned for execution on this aircraft were cancelled in October 1957.

[redacted]

[redacted]

Brief Summary of Aircraft Activity S/N 54-1361 - Continued)

Future Activity

Awaiting Air Force disposition.

Chronological History

14 March 1957 - Initial flight test
26 July 1957 - Completed flight loads program
27 December 1957 - Completed Air Force Phase IV tests
14 February 1958 - Completed structural demonstration tests
16 May 1958 - Completed speed brakes testing - end of program.

[redacted]

[redacted]

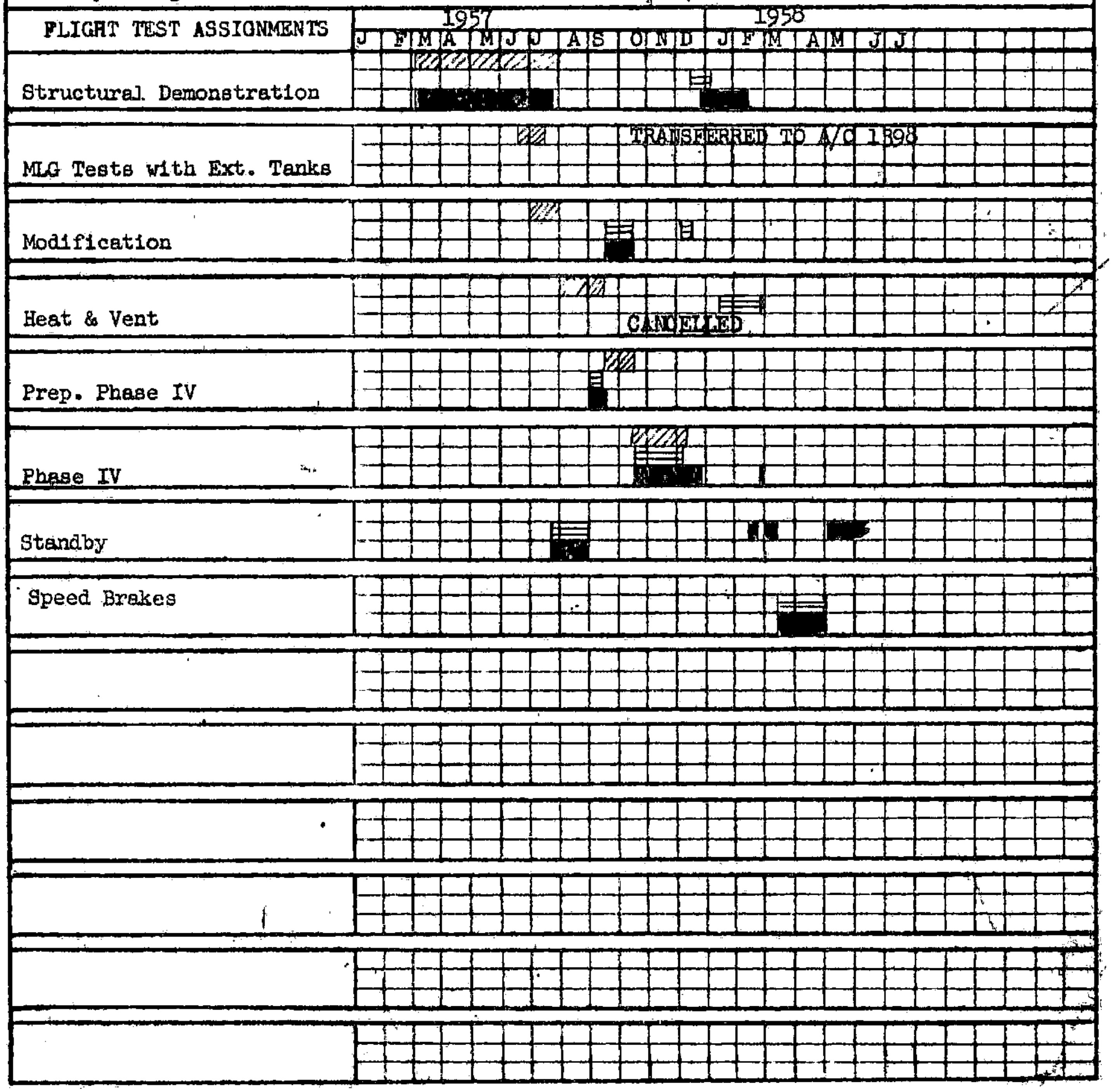
Page: 55AF
Date: 1 June 1958
Report No.: ZC-8-142-15
Model: TF-YF-F-102

Convair Report No.:
ZC-8-257

SUMMARY OF ACTIVITY

S/N 54-1361

Primary Assignment: Structural Demonstration, Air Force Phase IV, and Heat and Vent



Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1362

Primary Assignment

The assignment for this aircraft as presented in Convair Sales Order No. 33-1-511 dated 26 July 1956 is quoted as follows: "To be utilized for stability and control, spin and performance evaluation and Phase III systems tests of the production airplane configuration."

Summary of Activity

The aircraft was on a work status until the middle of May 1957, to accomplish the necessary instrumentation and aircraft items to conduct spin tests and also to test the spin chute deploy and jettison systems during taxi runs.

Spins were induced during several flights conducted through the middle of June. Recovery was effected by merely releasing controls. As a result of these successful tests the spin program has been completed. The aircraft was on a work status the latter part of June to remove spin equipment and to modify for stability and control tests.

The modification was completed and the stability and control testing began the latter part of August. Testing was conducted through September and included static and dynamic lateral-directional stability, dynamic longitudinal stability, low altitude stalls, and trim tests, etc. Completion of the stability and control portion of the program was curtailed as a result of a Headquarters USAF directive to reduce expenditures. The aircraft was ferried to Fort Worth (test to tactical) 8 November 1957.

Chronological History

29 March 1957 - Ferried from Convair-San Diego to Edwards Air Force Base.
19 June 1957 - Completed spin tests.
2 October 1957 - Completed flight test program.

[redacted]

UNCLASSIFI

[redacted]

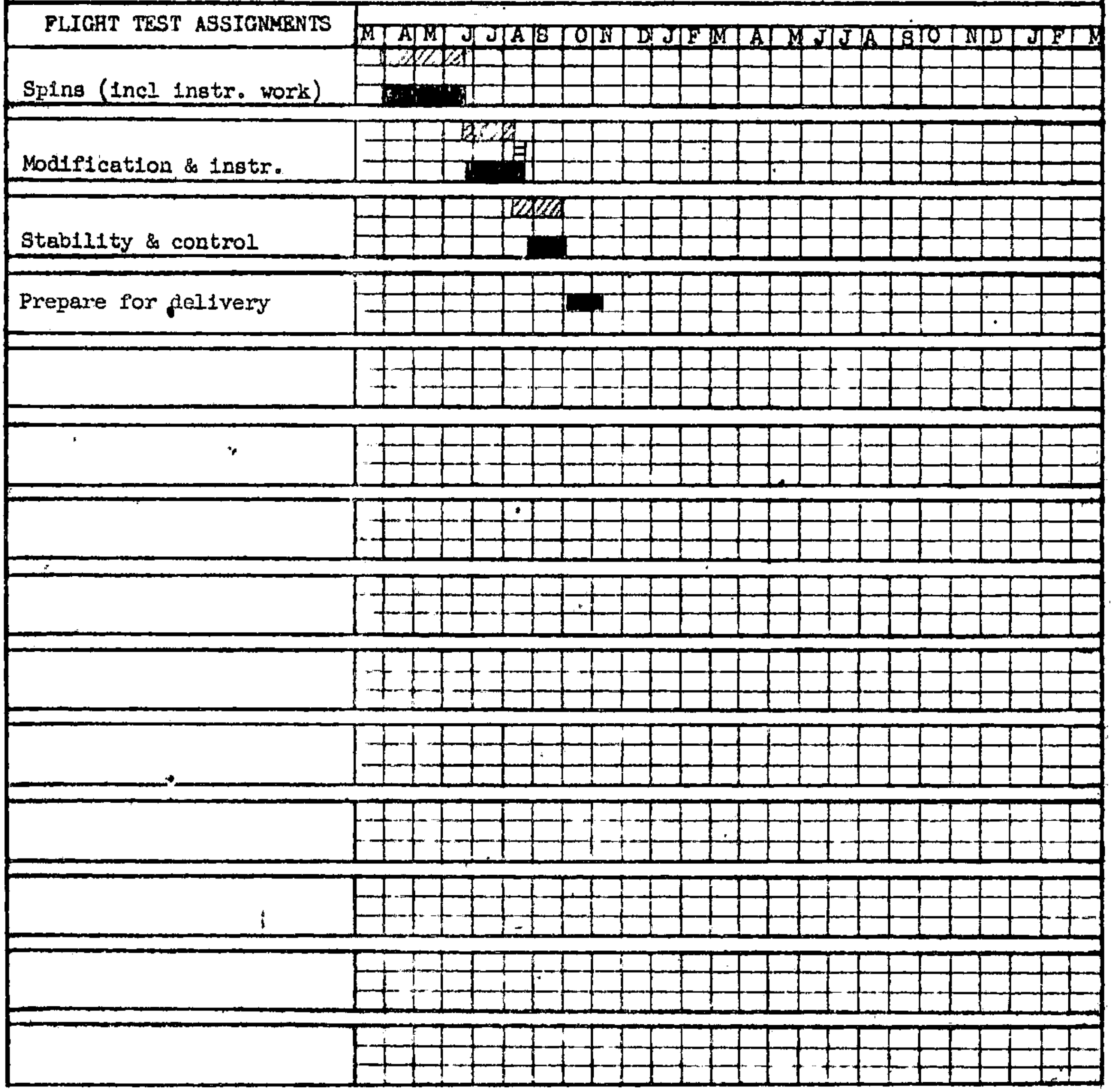
Page: 57A
Date: 1 June 1958
Report No.: ZC-8-142-
Model: TF-YF-F-102

Convair Report No.:
ZC-8-259

SUMMARY OF ACTIVITY

S/N 54-1362

Primary Assignment: Stability and Control, Spin and Performance



Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

CONFIDENTIAL

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1138

Primary Assignment

The assignment for this aircraft presented in Convair CCN No. 92, dated 17 April 1957 against Contract No. AF 33(600)-31174, states the program as follows: "Antenna test will be of qualitative nature with a minimum of instrumentation to determine the operational capabilities of the antenna".

Summary of Activity

The aircraft was ferried from Palmdale to Edwards Air Force Base 21 June 1957, during which a functional test flight of TACAN equipment was conducted. Only four TACAN check out flights were necessary to complete the flight test program. The TACAN equipment operated satisfactorily to meet all specification requirements.

Chronological History

21 June 1957 - Ferry flight from Palmdale to Edwards Air Force Base
26 June 1957 - Completed flight test program (ZC-8-288).

[redacted]

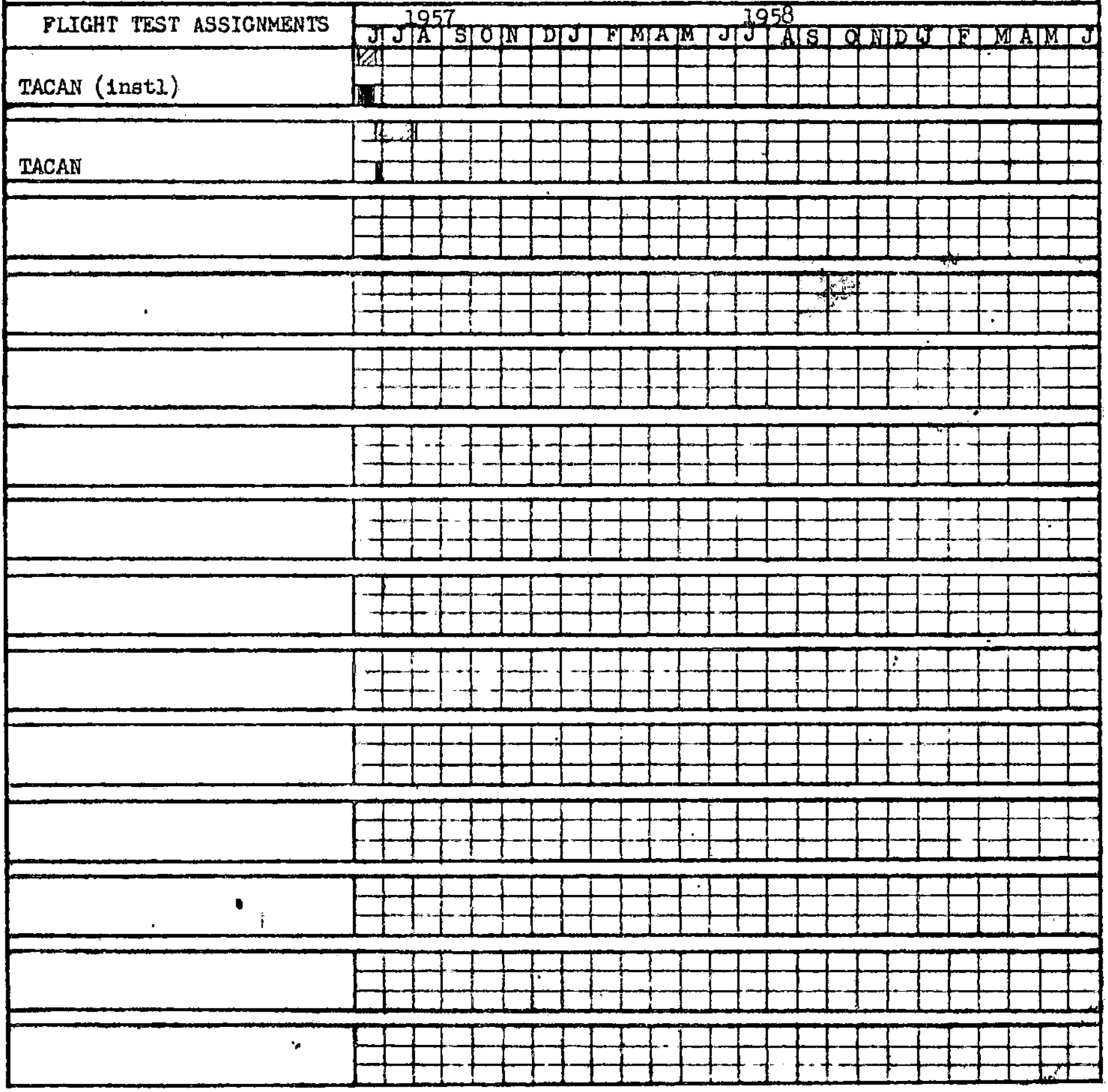
Page: 59A
Date: 1 July 1957
Report No.: YF-TF-F-102
Model: 2C-8-142-

Convair Report No.:

SUMMARY OF ACTIVITY

S/N 54-1138

Primary Assignment: TACAN



Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1380

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-586A, and dated 31 October 1956, is quoted as follows: "To be utilized for power plant development."

Summary of Activity

Since the initial flight test was conducted on 13 June 1956, the aircraft has been utilized for and has completed testing of the large area speed brakes, anti-surge bleed valve, afterburner shroud surface temperatures, engine flameout characteristics and constant speed drive development.

The Sundstrand constant speed drive development testing continued through November 1956. Tests regarding electrical system characteristics, afterburner relight with recirculating fuel igniters installed, emergency fuel control and cabin pressurization were conducted. Aircraft was in modification and instrumentation during December in preparation for components and systems development testing to be conducted in January.

Testing during January and February included:

Cabin pressure surge tests (to determine the cause for pressure surges and effect a cure.)
Carbon Monoxide tests
Edison fire detector test (to determine the ambient temperature in the vicinity of the fire detector cable.)
Afterburner relight tests (to determine the afterburner relight envelope with a recirculating ignitor)
Tests were also conducted to provide emergency fuel control data for the Link F-102A Simulator.
The aircraft was on a ground work status during part of February to complete aircraft modifications and instrumentation revisions.

Emergency landing gear extension tests, Sundstrand production parts tests, and temperature survey tests with shroud blankets removed were conducted through March and the initial part of April.

[redacted]

[redacted]

S/N 54-1380 (Continued)

Chronological History

28 February 1956 - Ferried from Convair-San Diego to Convair-Palmdale
26 April 1956 - Final acceptance by the U. S. Air Force
4 May 1956 - Ferried from Convair-Palmdale to Edwards Air Force Base
13 June 1956 - Initial flight test conducted
2 April 1957 - Completed flight test program at the termination of flight 68.

[redacted]

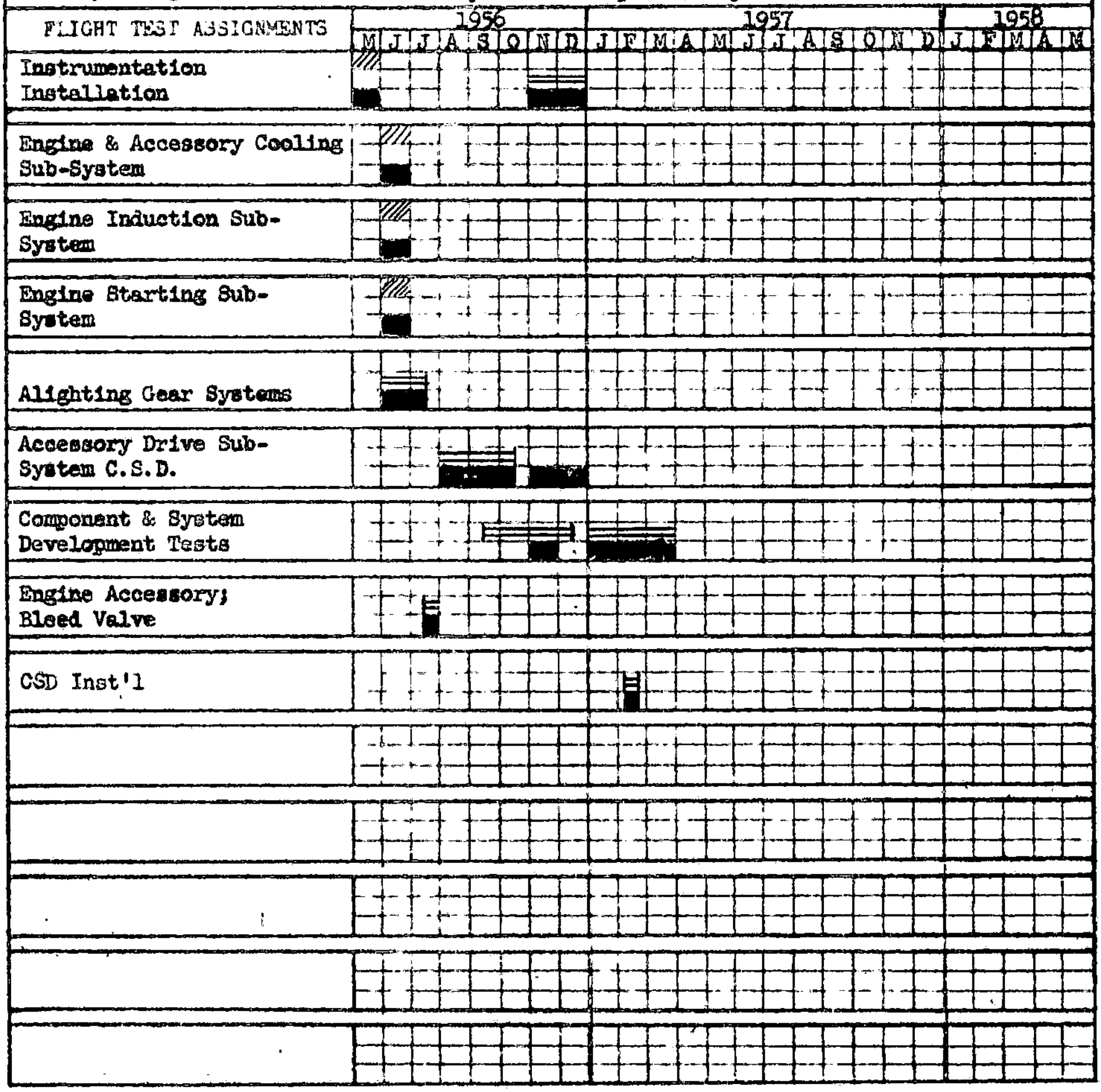
CONFIDENTIAL

Convair Report No..
ZC-8-240 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 54-1380

Primary Assignment: 8-10 Aircraft Systems - Propulsion System



Original Planning ▨ Revised Planning ☰ Activity ■

CONFIDENTIAL

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1388

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-586A, dated 31 October 1956, is quoted as follows: "To be utilized for specification compliance Phase III system testing and electronic environment testing with the production extended ramp, and MG-10 installations as follows:

(a) Heating and ventilating, including pilot comfort, electronic compartment cooling and vibration, windshield and canopy defog.

(b) All weather operation including radome, windshield and engine inlet scoop anti-ice and windshield rain clearing.".

Summary of Activity

Aircraft arrived at Edwards Air Force Base on 3 July 1956 from Convair-Palmdale to enter the F-102 Flight Test Program. During the middle of July 1956, it was utilized for flight testing of the engine inlet scoop anti-icing system , which were completed, and then placed on a ground work status in preparation for delivery to Convair-Palmdale, for MG-3 work. Instrumentation in preparation for evaluation of aft electronic area cooling was completed and the aircraft conducted testing for aft electronic cooling through October and heat and vent testing through November and December.

The testing conducted during January-February was primarily for determination of the adequacy of the heating and ventilating system. Stratos Unit, instrument panel vibrations, and air conditioning and pressurization tests were conducted. Rain clearing tests were completed.

Hamilton Standard Refrigeration Package testing was initiated and completed in March. Testing during the latter part of March and April included canopy tests, radome anti-icing tests, aft electronic cooling, and Stratos Refrigeration tests.

Following the above tests the aircraft was instrumented for hydraulic system tests. The hydraulic system tests were completed and the following by-product tests were accomplished during the report period: cabin pressure surge investigation, production trim servo tests, repeat dive for Stratos Unit tests, repeat dive (cabin pressure off) for pressurization tests.

The landing gear retraction test was not completed since the fabrication of parts could not be accomplished in time.

[redacted]

[redacted]

Brief Summary of Aircraft Activity S/N 54-1388 - (Continued)

Chronological History

4 June 1956 - Ferried from Convair-San Diego to Convair-Palmdale
30 June 1956 - Final acceptance by the U.S. Air Force
3 July 1956 - Ferried from Convair-Palmdale to Edwards Air Force Base
13 July 1956 - Initial flight test conducted
27 July 1956 - Ferried from Edwards Air Force Base to Convair-Palmdale for MG-3 installation
18 August 1956 - Ferried from Convair-Palmdale to Edwards Air Force Base
23 April 1957 - Completed all weather testing
18 June 1957 - Completed flight test program.

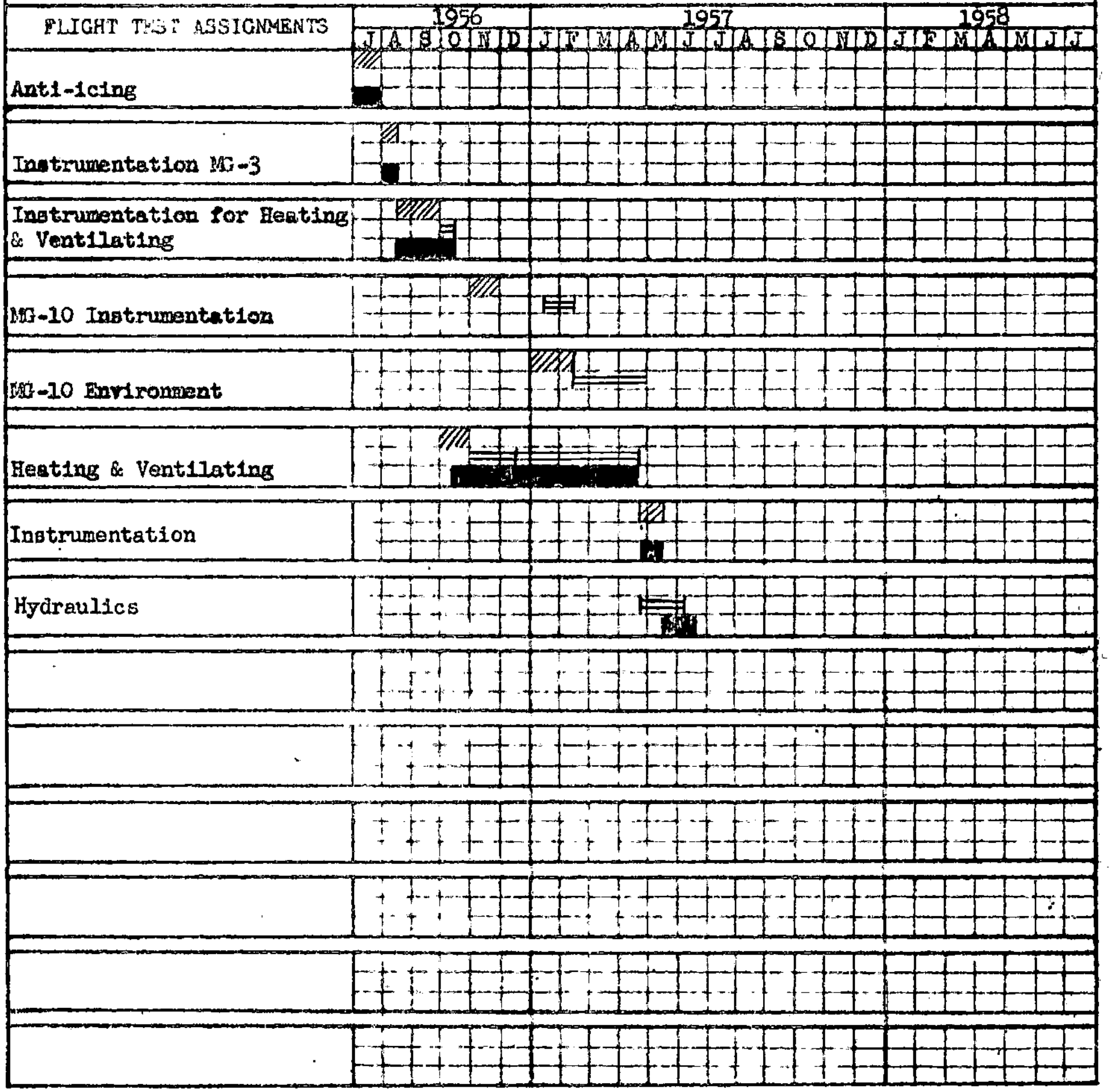
[redacted]

Convair Report No.:
ZC-8-183 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 54-1388

Primary Assignment: 8-10 All Weather - MG-10T Environment



Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1390

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-586A, dated 31 October 1956, is quoted as follows: "To be utilized for flutter stability, control and performance testing to evaluate the big tail installation and tactical improvement testing. The tactical improvement testing will be conducted to evaluate weight saving and such configuration changes as result from tactical operations by the United States Air Force.".

Summary of Activity

Initial testing on this aircraft consisted of production large fin in-flight flutter checks. The aircraft was placed on a ground work status in August 1956 for installation of: large speed brakes, oil cooling duct pressure instrumentation, control valve instrumentation, instrumentation required for inertia coupling and stability tests, and replacement of engine. Upon completion of ground work, testing was conducted to determine cause of roll-off during high "G" maneuvers, high "Q" pullouts to demonstrate lateral-directional stability characteristics, to evaluate the flutter characteristics of the F-102A with external wing tanks, and to determine aileron characteristics of the large tail configuration.

Stability and control testing continued through November. The aircraft was on work status during the first part of December for preparation of aircraft and instrumentation for landing gear tests.

Landing gear testing to determine the loads and deflections experienced by the skewed main landing gear were conducted through January and completed the latter part of February. Flights were also conducted for determination of drag chute deployment time for aircraft with large speed brakes and to determine the optimum bridle length for the drogue pilot chute. The aircraft was ferried to San Diego the latter part of February for MG-10 preparation prior to installation. Upon completion of these tasks the aircraft was ferried to Palmdale the middle of May.

The MG-10 was installed and production check out was accomplished prior to the aircraft being ferried to Holloman Air Force Base the latter part of July. Flights were conducted during the remainder of July for pilot familiarization with roll oscillation problem and flight check of the optical sight.

The Hughes package installation, which allows AFCS functions to be recorded, and instrumentation calibration was accomplished the first half of September.

[redacted]

[redacted]

Brief Summary of Aircraft Activity S/N 54-1390 (Continued)

Flights for investigation of roll oscillation in pilot assist mode were conducted the remainder of September and through October. The aircraft was placed on a stand-by status upon completion of these tests. While on stand-by status, the aircraft was utilized for target support of F-102A S/N 54-1401 until mid-January 1958. When the test program was completed, the aircraft was on stand-by until mid-February awaiting Air Force disposition. On 18 February the aircraft was ferried to another Air Force agency.

NOTE: The Specification Conformance Verification and Unsatisfactory Report test period was cancelled in September 1957 by virtue of a Headquarters, USAF Directive to reduce expenditures.

Chronological History

3 August 1956 - Ferried from Convair-San Diego to Edwards Air Force Base
3 August 1956 - Initial flight test conducted
13 August 1956 - Final acceptance by the United States Air Force
30 November 1956 - Completed stability and control tests
21 February 1957 - Completed landing gear tests
22 February 1957 - Ferried to Convair-San Diego
14 May 1957 - Ferried to Palmdale
20 July 1957 - Ferried to Holloman Air Force Base
23 October 1957 - Completed Convair flight test program
17 January 1958 - Completed target support role - on stand-by
18 February 1958 - Ferried to AFAC, Eglin AFB, Florida.

[redacted]

[redacted]

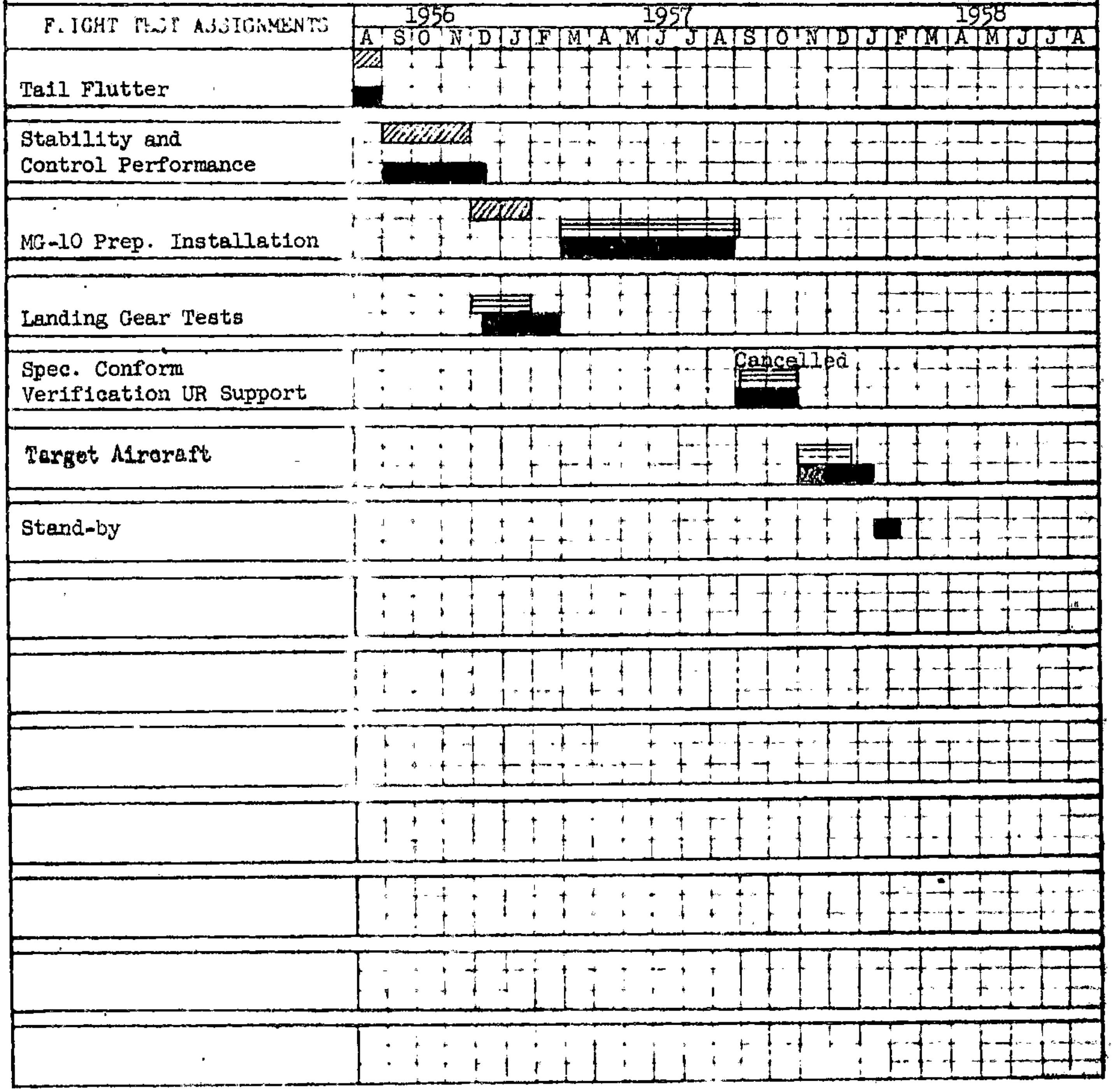
Page: 68A
Date: 1 June 1958
Report No.: ZC-8-142-19
Model: TF-YF-F-102

Convair Report No.
ZC-8-195

SUMMARY OF ACTIVITY

S/N 54-1390

Primary Assignment. 8-10 Stab.& Control, Tact. Improvement-Spec. Conform Verification, UR Support



Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

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BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1398

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-586A, dated 31 October 1956, is quoted as follows: "To be utilized for structural demonstration of the asymmetrical loading condition on the big tail and determination of Aerodynamic Stability Derivatives." (The latter test was transferred from S/N 53-1787 and later deleted from the program).

The test objectives of this program are:

1. To demonstrate the structural integrity of the airplane in the performance of asymmetric maneuvers determined to be critical for the aircraft.
2. To determine the magnitude and distribution of the structural loads imposed on the enlarged vertical fin by the air during the maneuvers determined to be critical by analysis and by review of the data obtained.
3. To provide flight loads data which will permit determination of structural margins of safety.

Summary of Activity

An extended shakedown period was necessitated due to unique instrumentation interference problems. The problems were resolved and data in furtherance of the flight test program were obtained during the latter part of December 1956.

The aircraft conducted asymmetric flight loads testing for almost the entire January, February, and March period. Testing in April included Telemetering Antenna Evaluation, Roll-off Investigation and Incremental Asymmetric Flight Loads tests which were completed the latter part of the month.

The aircraft underwent a twenty-five hour airframe and engine inspection following completion of the asymmetrical airloads program in April. Two flights were made to obtain needed lateral-directional response data. External tanks and reinforced landing gear fairing were installed for further demonstration testing and seven flights were conducted to complete the added program.

Upon assignment of further testing by WADC (landing loads on main landing gear) aircraft was withheld from scheduled delivery to Fort Worth for Test-to-Tactical conversion. Two instrumentation shakedown flights were conducted the first part of July.

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1398

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-586A, dated 31 October 1956, is quoted as follows: "To be utilized for structural demonstration of the asymmetrical loading condition on the big tail and determination of Aerodynamic Stability Derivatives." (The latter test was transferred from S/N 53-1787 and later deleted from the program).

The test objectives of this program are:

1. To demonstrate the structural integrity of the airplane in the performance of asymmetric maneuvers determined to be critical for the aircraft.
2. To determine the magnitude and distribution of the structural loads imposed on the enlarged vertical fin by the air during the maneuvers determined to be critical by analysis and by review of the data obtained.
3. To provide flight loads data which will permit determination of structural margins of safety.

Summary of Activity

An extended shakedown period was necessitated due to unique instrumentation interference problems. The problems were resolved and data in furtherance of the flight test program were obtained during the latter part of December 1956.

The aircraft conducted asymmetric flight loads testing for almost the entire January, February, and March period. Testing in April included Telemetering Antenna Evaluation, Roll-off Investigation and Incremental Asymmetric Flight Loads tests which were completed the latter part of the month.

The aircraft underwent a twenty-five hour airframe and engine inspection following completion of the asymmetrical airloads program in April. Two flights were made to obtain needed lateral-directional response data. External tanks and reinforced landing gear fairing were installed for further demonstration testing and seven flights were conducted to complete the added program.

Upon assignment of further testing by WADC (landing loads on main landing gear) aircraft was withheld from scheduled delivery to Fort Worth for Test-to-Tactical conversion. Two instrumentation shakedown flights were conducted the first part of July.

[redacted]

[redacted]

Brief Summary of Aircraft Activity S/N 54-1398 (Continued)

The aircraft was on an inactive status from July through September and on work status through October and the first week of November in preparation for the landing gear loads program. Instrumentation was installed and checked out with the exception of the instrumented main landing gear.

While waiting for the instrumented landing gear, flights were conducted during the November 1957 - January 1958 period to check out the aircraft's rate-of-sink instrumentation and evaluate the Navy mirror landing system which was installed for trials at Miramar NAS.

Upon receipt of the instrumented gear, the flight test program was initiated at EAFB and continued to completion approximately mid-April. The aircraft was then prepared for delivery to test-to-tactical modification at Fort Worth - leaving EAFB 23 April 1958.

Chronological History

21 September 1956 - Ferried from Convair-San Diego to Edwards AFB
26 September 1956 - Final acceptance by the United States Air Force
25 April 1957 - Completed incremental asymmetric flights loads testing
14 May 1957 - Completed external tank and main landing gear fairing demonstration flights
15 May through September 1957 - Inactive
9 April 1958 - Completed main landing gear loads program. (F-106A items).

[redacted]

[redacted]

Page: 71A
Date: [illegible] June 1958
Report No.: ZC-8-142-10
Model: F-102A

Convair Report No.:
ZC-8-243 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 54-1398

Primary Assignment: Structural Demonstration

FLIGHT TEST ASSIGNMENTS	1956								1957												1958				
	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M
Modification & Instrumentation Installation	▨ ■	▨ ■	▨ ■	▨ ■	▨ ■																				
Shakedown					▨ ■	■	■	■																	
Tail Load Asymmetrical Demonstration						▨	▨	▨ ☰ ■	▨ ☰ ■	☰ ■	☰ ■	☰ ■													
Instr. instl & checkout																		☰ ■	☰ ■	☰	☰				
Main landing gear loads																					☰	☰ ■	☰ ■		
Standby													☰ ■	☰ ■	☰ ■				☰ ■	☰ ■	■				
External tanks MLG fairing													☰ ■	☰ ■											
Inactive															☰ ■	☰ ■	☰ ■								

Original Planning ▨ Revised Planning ☰ Activity ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 54-1401

Primary Assignment

The assignment for this aircraft, as presented in Convair Sales Order No. 33-1-586A, dated 31 October 1956, is quoted as follows: "To be utilized for spin testing, to demonstrate specification compliance with the big tail and extended ramp configuration and to demonstrate the production airplane configuration operation to design limits including automatic flight control, data link and MG-10 demonstration and hydraulic and electrical systems." (The hydraulics portion has been transferred to and accomplished on aircraft S/N 54-1388.)

Summary of Activity

After the aircraft's arrival at Edwards AFB, it was placed on a ground work status for modification in preparation for spin tests. Upon completion, development and flight testing of an adequate spin recovery parachute was initiated.

Flights were conducted to determine spin recovery characteristics. Spins were successfully induced and recovery was effected by either releasing all controls or if necessary, applying "aileron with". The aircraft exhibited no pitch-up tendencies. The spin program was completed the latter part of April 1957.

The aircraft arrived at the San Diego Experimental Factory 1 May 1957 for aircraft and instrumentation modification and MG-10 FCS Provisions. The aircraft was ferried to Palmdale upon completion of the task at San Diego. The MG-10 FCS was installed at Palmdale during the latter part of August and first part of September. Instrumentation tie-in and checkout of the MG-10 was accomplished by the end of October at Holloman AFB. Snap up development flights were then conducted from November 1957 through mid-January. Attack courses of 0°, 70°, and 100° off the tail were accomplished with lock-on at 10,000 feet altitude differential.

Prior to completion of the Convair test program, the aircraft was assigned to the Armament Division of APGC (AFAC). The aircraft left Holloman AFB for AFAC on 18 February 1958.

Chronological History

21 September 1956 - Ferried from Convair-San Diego to Edwards AFB
26 September 1956 - Final acceptance by United States Air Force
26 April 1957 - Completed spin program
1 May 1957 - Ferried to Convair-San Diego
20 August 1957 - Ferried to Palmdale
16 September 1957 - Ferried to HAFB
17 January 1958 - Completed snap-up techniques test.

[redacted]

[redacted]

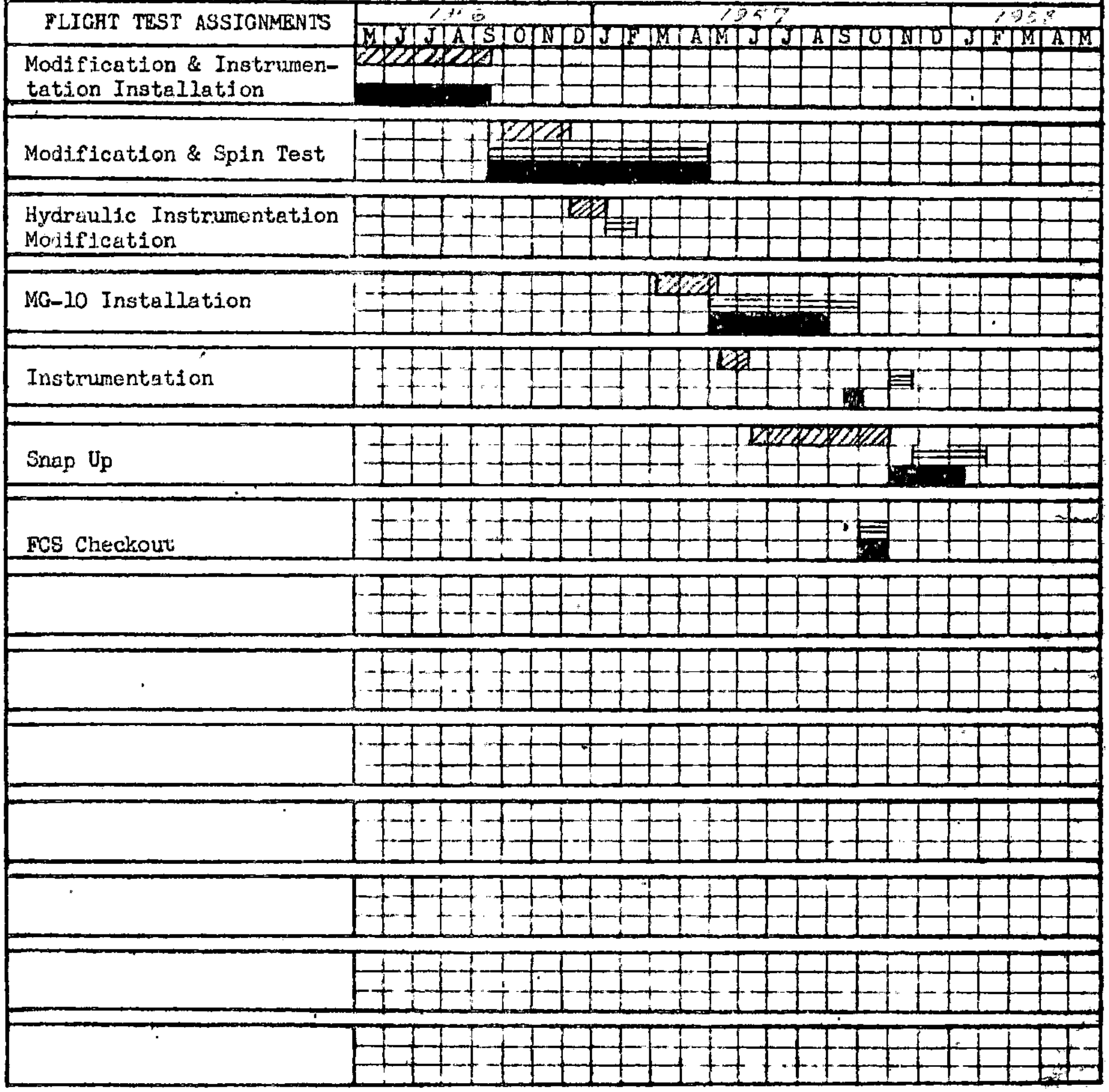
Page: 73A
Date: 1 June 1958
Report No.: ZC-8-142-10
Model: F-102A

Convair Report No.:
ZC-8-256 (Flt. Test Program)

SUMMARY OF ACTIVITY

S/N 54-1401

Primary Assignment: Spin Tests/Phase III Systems



Original Planning ▨ Revised Planning ☰ Activity ■

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Page: 75A
Date: 1 Sept 1957
Report No.: ZC-8-142-
Model: TF-YF-F-102

Convair Report No.:
ZC-8-153

SUMMARY OF ACTIVITY

S/N 55-4051

Primary Assignment: External Tanks

FLIGHT TEST ASSIGNMENTS	1957																								
	J	J	A	S	O	N	D	J	F	M															
External Tanks	▨ ■	▨ ■																							

Original Planning ▨ **Revised Planning** ☰ **Activity** ■

[redacted]

[redacted]

BRIEF SUMMARY OF AIRCRAFT ACTIVITY

S/N 55-4051

Primary Assignment

Convair Sales Order 33-1-713, dated 3 June 1957, indicates the assignment of the task of External Tank tests to aircraft S/N 55-4051 in lieu of S/N 54-1361.

Summary of Activity

The aircraft was ferried from Palmdale to Edwards Air Force Base on 17 June 1957. Flights were conducted for external tank evaluation tests for qualitative evaluation of stability, trim requirements, handling characteristics, and buffet characteristics of the TF-102A with external tanks. The program was concluded on completion of an external tank jettison test.

Chronological History

17 June 1957 - Ferry flight from Palmdale to Edwards Air Force Base
3 July 1957 - Completion of flight test program
5 July 1957 - Ferry to Palmdale.

[redacted]

[redacted]

APPENDIX C

COMPLETED FLIGHT TEST ITEMS AND REPORTS THEREON

[redacted]

[redacted]

Model YF-102 S/N 52-7994
Final Summary of Flight Test Program
ZC-8-101 Dated 7 August 1953

	Flight Test Items	Resolution of Flight Test Program
1.	Low Speed Taxi Tests	EF-8-84 - Low Speed Taxi Runs
2.	High Speed Taxi Tests	EF-8-84 - High Speed Taxi Runs
3.	Shakedown Flight Tests	*EF-8-84 - Shakedown Flight Tests *EF-8-89 - J57-P-11 Power Plant Characteristics (YF-102)
4.	a. Airspeed Calibration - Low Airspeed Range	Transferred to YF-102 USAF S/N 52-7995
	b. Airspeed Calibration - High Speed	Transferred to YF-102 USAF S/N 52-7995
5.	Angle of Attack & Angle of Yaw Pick-up Calibration	Transferred to YF-102 USAF S/N 52-7995
6.	Flutter and Vibration	*EF-8-84 - Flutter and Vibration *EF-8-84 - Results of YF-102 Flight Flutter Tests
7.	Preliminary Airplane Performance	*EF-8-84 - Preliminary Airplane Performance (YF-102)
8.	Dynamic Response	Transferred to YF-102 USAF S/N 52-7995
9.	Preliminary Flight Characteristics Tests	*EF-8-84 - Preliminary Flight Characteristics Tests (YF-102)
10.	Fuel Tank Pressurization System Tests	Transferred to YF-102 USAF S/N 52-7995
11.	Equipment Operation - Emergency Ram Air Hydraulic Power Unit	Transferred to YF-102 USAF S/N 52-7995
12.	Equipment Operation - Drogue Chute	Transferred to YF-102 USAF S/N 52-7995
13.	Fuel Consumption	Transferred to YF-102 USAF S/N 52-7995
14.	Emergency Fuel System Operation	Transferred to YF-102 USAF S/N 52-7995

[redacted]

UNCLASSIFIED

[redacted]

Model YF-102 S/N 52-7994
(Continued)

Flight Test Items	Resolution of Flight Test Program
15. Speed Brake Operation	*EF-8-84 - Speed Brake Operation
16. Stability and Control Tests	*EF-8-84 - Longitudinal Stability and Control *EF-8-84 - Directional and/or Lateral Stability and Control

*Indicates preliminary and/or qualitative

Entire program transferred to YF-102 USAF S/N 52-7995 because of YF-102 USAF S/N 52-7994 accident during take-off for Flight 7.

[redacted]

Model YF-102 S/N 52-7995
Final Summary of Flight Test Program
ZC-8-103 Dated 25 November 1953

[redacted]

Flight Test Items	Resolution of Flight Test Program	
1. Static Thrust Calibration	ZC-8-501-1	Thrust During Taxi Test
	ZC-8-501-3	Engine Operation During Take-off and Landing
	ZC-8-501-5	Engine Trim Characteristics
	ZC-8-501-6	Military Power Static Operation
	ZC-8-501-8	Engine Trim Characteristics
	ZC-8-530	Flight Test Evaluation of the J57-P-11 & P-41 Engine Installations in the YF-102 & YF-102A Airplanes
2. Brake Operation Evaluation	ZC-8-501-1	Operation During Taxi Tests
	ZC-8-501-5	Landing Without Drogue Chute
	ZC-8-501-6	Brake Drag-Drogue Chute Operation
	ZC-8-501-15	Wheel Brake System
	ZC-8-501-18	Drogue Chute Mechanism
3. Taxi Tests	ZC-8-501-1	Nose Wheel Lift-off - Elevator Forc [redacted]
	ZC-8-501-6	Nose Wheel Steering
	ZC-8-501-8	Nose Wheel Steering
	ZC-8-501-9	Touch Down Speed, Wind Effect, Ground Handling
	ZC-8-501-18	Nose Wheel Steering - Rudder System Modified
	ZC-8-501-19	Nose Wheel Steering - Rudder System Modified
	ZC-8-523	Take-off, Landing, and Ground Handling Characteristics
	ZC-8-512-7	Take-off, Landing and Ground or Water Handling Characteristics and Spray Characteristics Longitudinal Stick Force During Take-off
4. Engine Control System Operation	ZC-8-501-9	Engine Speeds
	ZC-8-501-10	Throttle Take-off Lock Control
	ZC-8-530	Flight Test Evaluation of the J57-P-11 & P-41 Engine Installations in the YF-102 & YF-102A Airplanes

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Model YF-102 S/N 52-7995
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Flight Test Items	Resolution of Flight Test Program
5. Shakedown Flights	ZC-8-501-1 - Operation During Taxi Tests - Low Pressure Pneumatic System - Oil Cooler Operation - Oil Cooler Regulators ZC-8-501-2 - Speed Brake Cycling - Cabin Temperature Regulator ZC-8-501-3 - Fin & Wing Buffet During Speed Brake Operation - Speed Brake Cycling - Cabin Pressure Regulator - Coc pit Fumes ZC-8-501-4 - In-flight Flutter (Damping) -Landing Gear Operation - In-flight Cycling Check - Low Pressure Pneumatic System ZC-8-501-6 - Oil System Temperatures - Brake Drag-Drogue Chute Operat - Landing Gear Operation ZC-8-501-7 - Aft Fuselage Skin & Base Press Survey - Cabin Pressure Surge ZC-8-501-8 - Tuft Study of Airflow Characteristics - Effect of Configuration Change on Oil Cooling - Oscillations of Secondary Hydraulic System - Landing Gear Operation - Better Cabin Heating, Modification for ZC-8-501-9 - Tuft Study of Airflow Characteristics - Fin & Wing Buffet During Maximum Power - Oscillations of Secondary Hydraulic System - Cabin Pressure Regulator

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Model YF-102 S/N 52-7995
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Flight Test Items	Resolution of Flight Test Program
5. Shakedown Flights - Continued	ZC-8-501-10 - Aft Fuselage Skin & Base Pressure Survey - Tuft Study of Airflow Characteristics - Control Buffet - In-flight Flutter (Supersonic) - Drogue Chute Operation - Main Landing Operation - Cabin Pressure Regulator - Cockpit Air Contamination ZC-8-501-11 - Tuft Study of Airflow Characteristics - Speed Brake Cycling ZC-8-501-12 - Landing Gear Operation - Grou ZC-8-527 - Low Pressure Pneumatic System - Cockpit
6. Equipment Operation - Speed Brakes	ZC-8-501-2 - Speed Brake Cycling ZC-8-501-3 - " " " ZC-8-501-11- " " "
7. Fuel Tank Pressurization System Tests	ZC-8-501-1 - Fuel System Pressures and Temperatures ZC-8-501-2 - Fuel Tank Pressures ZC-8-501-3 - Engine Fuel Flow Comparison ZC-8-501-4 - " " " " ZC-8-529 - Fuel & Lubricating Oil Systems Evaluation (YF-102)
8. Engine Equipment Operation	ZC-8-501-2 - Fuel Regulator - System Pressures ZC-8-501-4 - General Engine Operation ZC-8-501-15- " " " ZC-8-529 - Fuel and Lubricating Oil Systems Evaluation (YF-102)

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Flight Test Items	Resolution of Flight Test Program
9. Engine Equipment Operation - Engine Fuel Control	ZC-8-501-3 - Engine Fuel Flow Comparison
	ZC-8-529 - Fuel & Lubricating Oil Systems Evaluation (YF-102)
10. Engine Equipment Operation - Afterburner Operation	ZC-8-501-1 - General Operation
	ZC-8-501-9 - Afterburner Ignitor Control
	ZC-8-501-10- General Operation
	ZC-8-501-13- " "
	ZC-8-501-15- " "
	ZC-8-529 - Fuel and Lubricating Oil Systems Evaluation (YF-102)
11. Flutter and Vibration	ZC-8-501-2 -
	ZC-8-501-3 -
	ZC-8-501-4 - Effect of Mach Number
	ZC-8-501-6 -
	ZC-8-501-7 -
	ZC-8-501-9 -
	ZC-8-501-10-
	ZC-8-501-16-
	ZC-8-501-17
	ZC-8-501-20
	ZU-8-013 - Results of YF-102 Flight Flutter Tests
12. Airspeed Calibration - Low Speed	ZC-8-501-3 - Fly-by Method
	ZC-8-501-4 - " "
	ZC-8-501-8 - Radar, Fly-by, Radiosonde, and Photopanel Correlation
	ZC-8-501-12- Radar, Radiosonde, and Photopanel Correlation
	ZC-8-4540-Determination of Static Source Post. Error

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Model YF-102 S/N 52-7995
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Flight Test Items	Resolution of Flight Test Program
13. Airspeed Calibration - High Speed	ZC-8-501-4 - Pacer Method ZC-8-501-6 - Radar Method ZC-8-501-10- Radar, Radiosonde, & Photopanel Correlation ZC-8-501-12- Radar, Radiosonde, & Photopanel Correlation
4. Preliminary Flight Characteristics Test	ZC-8-501-1 - Nose Wheel Lift-off - Elevator Force ZC-8-501-2 - Shakedown Flight ZC-8-501-4 - Qualitative Analysis of Handling Characteristics ZC-8-501-6 - Qualitiative Analysis of Handling Characteristics ZC-8-501-10- Qualitative Analysis of Handling Characteristics ZC-8-501-11- Shakedown Flight ZC-8-501-15- Nose Wheel Lift-off - Effect of Configuration Changes ZC-8-501-20- Standard or Modified Cam in Shelley Unit ZC-8-501-21- Take-off Characteristics-Configuration Effect
15. Angle of Attack and Angle of Yaw Pick-up Calibration	Transferred to YF-102 USAF S/N 53-1780
16. Preliminary Airplane Performance, Climb	ZC-8-501-7 - Maximum Power Climb ZC-8-501-9 - ZC-8-501-10- Climb to Service Ceiling ZC-8-501-21- Effect of Configuration Changes ZC-8-534 - Climb Performance Evaluation of the YF-102 Airplane

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Model YF-102 S/N 52-7995
(Continued)

	Flight Test Items	Resolution of Flight Test Program
17.	Preliminary Airplane Performance, Speed Power, Fuel Consumption	ZC-8-501-1 - Climb-Fuel Consumption ZC-8-501-2 - Drag Evaluation ZC-8-501-4 - Speed Demonstration ZC-8-501-6 - " " ZC-8-501-SR3 - Drag Characteristics
18.	Cambered Wing L. E. and Reflex Wing Tips	ZC-8-518 - Wing Tip Air Loads ZC-8-501-24- Wing Tip Air Loads
19.	Phase II	USAF
20.	Effect of Missile Bay Door Operation on Flight Characteristics	Transferred to YF-102 USAF S/N 53-1780
21.	Airplane Performance, Determination of Optimum Leading Edges and Wing Tips	ZC-8-501-SR3 - Drag Characteristics ZC-8-512-27 - Drag Characteristics - 10° Reflexed and Fully Cambered Wing Tips
22.	Airplane Performance, Take-off Distance	ZC-8-501-2 - Take-off Distance ZC-8-501-3 - Take-off (Phototheodolite Camera Data) ZC-8-501-8 - Take-off (Phototheodolite Camera Data) ZC-8-501-12- Take-off (Phototheodolite Camera Data) ZC-8-523 - Take-off, Landing, and Ground Handling Characteristics
23.	Airplane Performance, Landing Distance	ZC-8-501-2 - Landing Distance ZC-8-501-3 - Landing (Phototheodolite Camera Data ZC-8-501-8 - " " " " ZC-8-501-12- " " " " ZC-8-523 - Take-off, Landing, and Ground Handling Characteristics

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Model YF-102 S/N 52-7995
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Flight Test Items	Resolution of Flight Test Program	
24. Engine and Structural Cooling	ZC-8-501-2 -	Accessory Compartment Cooling
	ZC-8-501-3 -	Effect of Configuration Changes on Cooling
	ZC-8-501-4 -	Effect of Configuration Changes on Cooling
	ZC-8-501-5 -	Accessory Compartment Cooling
	ZC-8-501-6 -	Airflow Study for Cooling Requirements
	ZC-8-501-7 -	Engine Accessory & Fuselage Structural Cooling
	ZC-8-501-8 -	Effect of Configuration Change on Cooling
	ZC-8-501-9 -	" "
	ZC-8-501-10-	Temperatures Aft of Firewall; Accessory Compartment Temp.
	ZC-8-501-11-	Effect of Configuration Changes on Cooling
	ZC-8-501-12-	A/B Nozzle and Aft Structure Temperatures; Accessory Compartment Temperatures
5. Dynamic Response	Transferred to YF-102 USAF S/N 53-1780	
6. Static Longitudinal Stability	ZC-8-501-23-	Elevator Trim (Effect of Load Factors, C.G., Mach No. etc.)
	ZC-8-501-24-	
	ZC-8-512-1 -	Elevator Trim (Effect of Load Factors, C.G., Mach No. etc.)
	ZC-8-526 -	Flight Test Evaluation of YF-102 Stability & Control Characteristics
27. Dynamic Longitudinal Stability	ZC-8-501-10-	"Pitch-up" Phenomenon
	ZC-8-501-15-	Porpoising
	ZC-8-501-16-	"
	ZC-8-501-21-	"
	ZC-8-501-24-	"Pitch-up" Phenomenon
	ZC-8-526 -	Flight Test Evaluation of YF-102 Stability & Control Characteristics

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Flight Test Items	Resolution of Flight Test Program	
28. Longitudinal Control Force	ZC-8-501-3	- Spring Replacement for Improved Nose Lift-off
	ZC-8-501-8	- Longitudinal Control Force
29. Longitudinal Control Effectiveness	ZC-8-501-14	- Effect of Modified Shelley Unit
		- Longitudinal Control During Landing - Cambered Wing
	ZC-8-526	- Flight Test Evaluation of YF-10 [redacted] Stability and Control Characteristics
[redacted] 30. Longitudinal Trim Changes	ZC-8-501-3	- Elevator Angle Required for Tri
	ZC-8-501-8	- Longitudinal Trim Rate
	ZC-8-526	- Flight Test Evaluation of YF-10 Stability and Control Characteristics
31. High Speed Requirements	ZC-8-501-16	
	ZC-8-526	- Flight Test Evaluation of YF-10 Stability and Control Characteristics
32. Static Directional Stability	ZC-8-501-6	- Directional Oscillations During Speed Brake Operation
	ZC-8-501-7	- Directional Oscillations
	ZC-8-501-9	- " "
	ZC-8-501-20	- Transonic Directional Trim Chang
	ZC-8-526	- Flight Test Evaluation of YF-10 Stability and Control Characteristics
33. Dynamic Directional Stability	ZC-8-501-10-	
	ZC-8-501-15-	Rudder Kicks (Effect of Yaw Damper)
	ZC-8-501-16-	" " " " " "
	ZC-8-501-17-	Rudder Kicks (Effect of Mach No.)
	ZC-8-512-8	- Aileron Rolls (Effect of Yaw Damper)
	ZC-8-512-16-	Aileron Rolls (Normal & Lateral Acceleration)
	ZC-8-526	- Flight Test Evaluation of YF-102 Stability and Control Characteristics

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Model YF-102 S/N 52-7995
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	Flight Test Items	Resolution of Flight Test Program
34.	Spiral Stability	Cancelled
35.	Rudder Control Effectiveness	Cancelled
36.	Directional Trim Control Effectiveness	Cancelled
37.	Static Lateral Stability	Cancelled
38.	Dynamic Lateral Stability	ZC-8-512-8 - Aileron Rolls (Effect of Yaw Damper) ZC-8-512-16- Aileron Rolls (Normal & Lateral Acceleration) ZC-8-526 - Flight Test Evaluation of YF-102 Stability and Control Characteristics
39.	Dihedral Effect	Cancelled
40.	Aileron Control Effectiveness	ZC-8-512-9 - Aileron Effectiveness (Pb/2V vs Aileron Angle) ZC-8-512-21- Aileron Effectiveness (Pb/2V vs Aileron Angle) ZC-8-526 - Flight Test Evaluation of YF-102 Stability and Control Characteristics
41.	Stalling Characteristics	ZC-8-501-10- Stalling Characteristics
42.	Pilot Assist Development	Transferred to YF-102 USAF S/N 53-1780
43.	Engine Operation (Air Starts)	Cancelled
44.	Drogue Chute Tests	ZC-8-512-10- ZC-8-512-20-
45.	Equipment Operation (Emergency Fuel Control System)	Cancelled

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Model YF-102 S/N 52-7995
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	Flight Test Items	Resolution of Flight Test Program
46.	Equipment Operation - Emergency Ram Air Hydraulic Power Unit	Transferred to YF-102 USAF S/N 53-1780 Reinstated to S/N 52-7995 by Addendum 12
47.	Flight Control System Operation	ZC-8-501-8 - Rudder Forces - Lateral Control Response ZC-8-501-9 - Rudder Feel System ZC-8-501-12- Yaw Damper System ZC-8-501-16- Effect of Modified Cam in Shelley Unit ZC-8-501-17- Longitudinal Stick Forces ZC-8-501-19- Elevator Control Feel System ZC-8-501-21- " " " " - Elevator Ratio Changer - Evaluation of Double-headed Control Stick ZC-8-501-22- Elevator Ratio Changer ZC-8-501-23- Bellophragm "Q" Pot Evaluation Elevator Control Feel System ZC-8-512-2 - Force Link Evaluation ZC-8-512-6 - Control System Hysteresis Check With Force Link Installed ZC-8-512-26- Non-linear Pitch Rate Network - Pitch 'g' Limiter - Shelley Unit Static Pressure Source Evaluation ZC-8-519 - Direct Manual Control System - Developmental Flight Tests

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Model YF-102 S/N 52-7995
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	Flight Test Items	Resolution of Flight Test Program	
	Addenda to Flight Test Program		
1.	Landing Gear Hydraulic Actuating Pressures	ZC-8-501-12-	Landing Gear Operation - Ground
2.	Inlet Duct Performance	Superseded by Addendum 16A	
3.	Dummy Fuel Tank Vent Pressures, L/H Reflexed Wing	Cancelled	
4.	Right Hand Engine Inlet Duct Pressures and Strains	ZC-8-501-14-	Engine Air Inlet Duct Pressures and Stresses
		ZC-8-501-15-	Static and Dynamic Engine Air Inlet Duct Stresses
		ZC-8-530 -	Flight Test Evaluation of the J57-P-11 & P-41 Engine Installations in YF-102 & YF-102A A/C
5.	Dynamic Directional Stability	ZC-8-501-15-	Rudder Kicks (Effect of Yaw Damper)
		ZC-8-501-16-	Rudder Kicks (Effect of Yaw Damper)
		ZC-8-501-17-	Rudder Kicks (Effect of Mach No.
		ZC-8-526 -	Flight Test Evaluation of YF-102 Stability and Control Characteristics
6.	Airspeed Calibration - Transonic Speed	Superseded by Addendum 20	
7A.	Primary Hydraulic System Oil Temperatures	ZC-8-501-22-	Hydraulic Pump Temperature
		ZC-8-516 -	Hydraulic System Oil Temperature, YF-102 and YF-102A
8.	Pressure Data on the Elongated Nose	ZC-8-512-2 -	Pressure Distribution on Elongated Nose
		ZC-8-528 -	Fuselage Pressure Distribution Studies from Flight Test

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Flight Test Items	Resolution of Flight Test Program
Addenda to Flight Test Program - Continued	
9B. Afterbody Fillet (Yellow Canary), Afterburner Nozzle Actuator	ZC-8-501-18- Cockpit Noise Level - Pressure Distribution on Large Fuselage - Elevon Fairing ZC-8-501-20- Pressure Distribution on Large Fuselage - Elevon Fairing ZC-8-528 - Fuselage Pressure Distribution Studies from Flight Test
[redacted]A. Engine Performance Tests at Two Conditions of Bleed Air Consumption	ZC-8-501-15- Effect on Thrust of Compressor Bleed Airflow for Auxiliary Power ZC-8-501-16- Effect of Thrust on Compressor Bleed Airflow ZC-8-530 - Flight Test Evaluation of the J57-P-11 & P-41 Engine Installations in the YF-102 & YF-102A A/
11. Drag Data Test	ZC-8-501-16- Drag Polars-Level Flight ZC-8-501-18- Drag Polars ZC-8-501-19- Drag Evaluation ZC-8-501-SR3 - Drag Characteristics
12. Preliminary Ram Air Turbine Tests	ZC-8-512-10- Ram Air Turbine Operation ZC-8-514 - Ram Air Turbine Emergency Hydraulic Power System Evaluation, F-102 Series 8-82
13. Simulated Landings with Ram Air Turbine Emergency Hydraulic Power	Superseded by Addendum 12B

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Model YF-102 S/N 52-7995
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	Flight Test Items	Resolution of Flight Test Program	
	Addenda to Flight Test Program - Continued		
14A.	Longitudinal Sensitivity	ZC-8-501-21	
		ZC-8-519	
[redacted] 15.	Temperature of Engine Outer Wall and Nozzle Actuators	ZC-8-501-2 -	Accessory Compartment Cooling
		ZC-8-501-13-	Afterburner Nozzle and Aft Fuselage Structure Temperatures
		ZC-8-501-14-	A/B Nozzle Actuator Temperature [redacted]
		ZC-8-501-15-	" " " "
		ZC-8-501-18-	" " " "
		ZC-8-501-20-	Afterburner Nozzle Actuator and Aft Fuselage Fairing Temperatur
		ZC-8-522 -	Engine and Structural Cooling System Evaluation (YF-102)
6A.	J57-P-11 Compressor Stall Test	ZC-8-501-20-	Anti-Surge Bleed Valve Optimiza- tion Tests Compressor Stall
		ZC-8-501-21-	Provoked Compressor Stalls
		ZC-8-501-22-	Compressor Stalls - Acceleratio and Deceleration
		ZC-8-530 -	Flight Test Evaluation of the J57-P-11 and P-41 Engine Installations in the YF-102 and YF-102A Airplanes
17.	Engine Lubricating Oil Temperatures With and Without Air-Oil Cooler	ZC-8-501-18-	Hydraulic Cooling (Fuel Cooled-Oil Cooler Removed)
		ZC-8-501-19-	Hydraulic Cooling (Fuel Cooled-Oil Cooler Removed)
		ZC-8-501-22-	Engine Oil Temperature Comparison, Hydraulic Pump Temperature

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Model YF-102 S/N 52-7995
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Flight Test Items		Resolution of Flight Test Program
Addenda to Flight Test Program - Continued		
17. Engine Lubricating Oil Temperatures With and Without Air-Oil Cooler - Continued	ZC-8-512-3 -	Hydraulic Pump Temperature
	ZC-8-512-4 -	Engine Oil Cooling (Air-Lube Oil Cooler By-passed)
	ZC-8-512-12-	Engine Oil Cooling (Air-Lube Oil Cooler By-passed)
	ZC-8-529 -	Fuel and Lubricating Oil System Evaluation (YF-102)
18. Aft Section Cooling Test	ZC-8-501-13-	Afterburner Nozzle and Aft Fuse lage Structure Temperatures
	ZC-8-501-19-	Direction of Cooling Airflow
	ZC-8-501-20-	Afterburner Nozzle Actuator and Aft Fuselage Fairing Temperatur
	ZC-8-501-12-	Afterburner Nozzle and Aft Stru ture Temperatures; Accessory Compartment Temperatures
	ZC-8-522 -	Engine and Structural Cooling System Evaluation (YF-102)
19. Instrument Panel Vibration Test	ZC-8-501-21-	Pilot's Instrument Panel Vibration - Vibrator Installed
	ZC-8-501-22-	Pilot's Instrument Panel Vibration - Vibrator On and Off
	ZC-8-501-23-	Pilot's Instrument Panel Vibration - Vibrator On and Off
	ZC-8-501-24-	Pilot's Instrument Panel Vibration - Vibrator Installed
20. Airspeed Calibration	ZC-8-501-18-	Pacer, Radar, and Fly-by Methods
	ZC-8-501-20-	Fly-by Method

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Model YF-102 S/N 52-7995
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Flight Test Items	Resolution of Flight Test Program
Addenda to Flight Test Program - Continued	
21. Drag Characteristics	ZC-8-501-16- Drag Polars-Level Flight ZC-8-501-18- Drag Polars ZC-8-501-19- Drag Evaluation ZC-8-501-20- " " ZC-8-501-21- " " ZC-8-501-SR3 - Drag Characteristics
22. Vibration and Acceleration Characteristics	ZC-8-501-22- Fuselage Vibrations
23. Cable Tension Values	ZC-8-512-5 - Control Cable Tensions - Turn Coordinator Operation ZC-8-519 - Direct Manual Control System - Developmental Flight Tests
24. Buffet Characteristics and Wing Fence Evaluation	ZC-8-501-21- Drag Evaluation - Fin Buffet ZC-8-501-22- Fin and Wing Buffet ZC-8-501-23- " " " " EFT-A-8-26 - Wing Fence and Wing Tip Evaluation (YF-102)
25A. Adverse Yaw and Directional Characteristics	ZC-8-512-4 - Aileron Rolls (Effect of Yaw Damper) ZC-8-526 - Flight Test Evaluation of YF-102 Stability and Control Characteristics

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Model YF-102 S/N 53-1779
Final Summary of Flight Test Program
ZC-8-108 Dated 6 April 1954

	Flight Test Items	Resolution of Flight Test Program
1.	Airspeed Calibration - Subsonic Speed Range	ZC-8-502-1 - Airspeed Calibration - Pacer Method ZC-8-540 - Determination of Static Source Post. Error
2.	Airspeed Calibration - Transonic and Supersonic Flight Range	Cancelled
3.	Angle of Attack and Angle of Yaw Pick-up Calibration	Transferred to YF-102 USAF S/N 53-1780
4.	Dynamic Lateral - Directional Stability Without Damping	ZC-8-502-2 - Dynamic Directional and/or Lateral Stability Dynamic Directional Stability - Lateral Directional Oscillations (Time to Damp) ZC-8-502-3 - Dynamic Directional and/or Lateral Stability Dynamic Directional Stability ZC-8-502-4 - Dynamic Directional and/or Lateral Stability Dynamic Directional Stability ZC-8-502-6 - Dynamic Directional and/or Lateral Stability Dynamic Directional Stability ZC-8-502-7 - Dynamic Directional and/or Lateral Stability Dynamic Directional Stability ZC-8-502-8 - Dynamic Directional and/or Lateral Stability Dynamic Directional Stability ZC-8-502-9
5.	Dynamic Lateral - Directional Stability With Damping	ZC-8-502-2 - Dynamic Directional and/or Lateral Stability Dynamic Directional Stability - Lateral Directional Oscillations (Time to Damp)

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Flight Test Items	Resolution of Flight Test Program	
5. Dynamic Lateral - Directional Stability With Damping - Continued	ZC-8-502-3	- Dynamic Directional and/or Lateral Stability Dynamic Directional Stability
	ZC-8-502-4	- Dynamic Directional and/or Lateral Stability Dynamic Directional Stability
		- Yaw Damper System
	ZC-8-502-6	- Dynamic Directional and/or Lateral Stability Dynamic Directional Stability
		- Yaw Damper System
	ZC-8-502-7	- Dynamic Directional and/or Lateral Stability Dynamic Directional Stability
		- Pitch Damper System
		- Yaw Damper System
	ZC-8-502-8	- Dynamic Directional and/or Lateral Stability Dynamic Directional Stability
		- Pitch Damper System
		- Yaw Damper System
	ZC-8-502-9	- Pitch Damper System
	ZC-8-521	- Automatic Flight Control System (Model 8-82) Flight Test Development
	ZC-8-526	- Flight Test Evaluation of YF-102 Stability and Control Characteristics
6. Dynamic Longitudinal Stability Without Damping	ZC-8-502-7	- Dynamic Longitudinal Stability
	ZC-8-502-8	- " " "
	ZC-8-502-9	- " " "
	ZC-8-526	- Flight Test Evaluation of YF-102 Stability and Control Characteristics

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Model YF-102 S/N 53-1779
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Flight Test Items		Resolution of Flight Test Program
7. Dynamic Longitudinal Stability With Damping	ZC-8-502-5	- Pitch Damper System
	ZC-8-502-7	- Dynamic Longitudinal Stability - Pitch Damper System - Yaw Damper System
	ZC-8-502-8	- Dynamic Longitudinal Stability - Pitch Damper System - Yaw Damper System
	ZC-8-502-9	- Dynamic Longitudinal Stability - Pitch Damper System
	ZC-8-521	- Automatic Flight Control Systems (Model 8-82) Flight Test Development
	ZC-8-526	- Flight Test Evaluation of YF-102 Stability & Control Characterist
8. Dynamic Response to Speed Brake Operation	Cancelled	
9. Dynamic Response to Armament Bay Door Operation	ZC-8-502-3	- Missile Bay Door Operation
	ZC-8-526	- Flight Test Evaluation of YF-10 Stability & Control Characteri

Addendum To Flight Test Program

1. Control System Ground Test	ZC-8-502-4	- Yaw Damper System
	ZC-8-521	- Automatic Flight Control Systems (Model 8-82) Flight Test Development

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Model YF-102 S/N 53-1780
Final Summary of Flight Test Program
ZC-8-111 Dated 17 July 1954

Flight Test Items	Resolution of Flight Test Program	
1. Dynamic Lateral - Directional Stability	ZC-8-503-2	- Aileron Rolls (Effect of Yaw Damper)
	ZC-8-503-5	- Adverse Yaw Due to Lateral Control
	ZC-8-526	- Flight Test Evaluation of YF-102 Stability & Control Characteristi [redacted]
[redacted] 2. Directional Dynamic Response	ZC-8-503-5	- Pitch Damper System - Pitch "g" Limiter - Yaw Damper System
	ZC-8-503-6	- Control System Friction - Pitch Damper System - Yaw Damper System - Yaw Damper-Turn Coordinator Function
	ZC-8-504-7	- Pilot Assist System - Pitch "g" Limiter - Turn Coordinator Operation - Yaw Damper System
	ZC-8-503-8	- Pilot Assist System - Pitch Damper System - Pitch "g" Limiter - Yaw Damper System - Yaw Damper-Turn Coordinator Function
	ZC-8-503-9	- Control System Hysteresis Check - Pilot Assist System - Pitch Damper System - Pitch "g" Limiter - Turn Coordinator Operation - Yaw Damper System
	ZC-8-503-10	- Full Authority Pilot Assist System - Pitch "g" Limiter - Yaw Damper System
	ZC-8-521	- Automatic Flight Control Systems (Model 8-82) Flight Test Development

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Model YF-102 S/N 53-1780
(Continued)

Flight Test Items	Resolution of Flight Test Program	
3. Dynamic Longitudinal Stability	ZC-8-503-2	- Effect of Pitch Damper on Programmer Induced Oscillations
	ZC-8-503-6	- Pitch Damper Operation
	ZC-8-526	- Flight Test Evaluation of YF-102 Stability & Control Characteristics
4. Longitudinal Dynamic Response	ZC-8-503-3	- Pitch Damper System
	ZC-8-503-4	- " " "
		- Pitch "g" Limiter
	ZC-8-503-5	- Pitch Damper System
		- Pitch "g" Limiter
		- Yaw Damper System
	ZC-8-503-6	- Control System Friction
		- Pitch Damper System
		- Yaw Damper System
		- Yaw Damper-Turn Coordinator Function
	ZC-8-503-7	- Pilot Assist System
		- Pitch "g" Limiter
		- Turn Coordinator Operation
		- Yaw Damper System
	ZC-8-503-8	- Pilot Assist System
		- Pitch Damper System
		- Pitch "g" Limiter
		- Yaw Damper System
		- Yaw Damper-Turn Coordinator Function
	ZC-8-503-9	- Control System Hysteresis Check
		- Pilot Assist System
		- Pitch Damper System
		- Pitch "g" Limiter
		- Turn Coordinator Operation
		- Yaw Damper System
	ZC-8-521	- Automatic Flight Control Systems (Model 8-82) Flight Test Development

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Model YF-102 S/N 53-1780
(Continued)

Flight Test Items	Resolution of Flight Test Program
5. Full Authority Elevon Control	ZC-8-512-14- Linear Elevon Feedback Potentiometers ZC-8-512-15- Pilot Assist System ZC-8-512-26- Non-linear Pitch Rate Network - Pitch "g" Limiter - Shelley Unit Static Pressure Source Evaluation ZC-8-503-7 - Pilot Assist System - Pitch "g" Limiter - Turn Coordinator Operation - Yaw Damper System ZC-8-50. -8 - Pilot Assist System - Pitch Damper System - Pitch "g" Limiter - Yaw Damper System - Yaw Damper-Turn Coordinator Function ZC-8-503-9 - Control System Hysteresis Check - Pilot Assist System - Pitch Damper System - Pitch "g" Limiter - Turn Coordinator Operation - Yaw Damper System ZC-8-503-10- Full Authority Pilot Assist Sys. - Pitch "g" Limiter - Yaw Damper System ZC-S-503-11- Pilot Assist System - Pitch "g" Limiter ZC-8-503-12- Pilot Assist System - Pitch "g" Limiter - Turn Coordinator Operation ZC-8-521 - Automatic Flight Control Systems (Model 8-82) Flight Test Development

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Model YF-102 S/N 53-1780
(Continued)

Flight Test Items	Resolution of Flight Test Program
6. Turn Coordination Effectiveness	ZC-8-512-5 - Control Cable Tensions
	- Turn Coordinator Operation
	ZC-8-512-7 - " " "
	ZC-8-512-17- " " "
	ZC-8-512-18- Automatic Instrument Landing (AILS) Approach
	ZC-8-512-29- Turn Coordinator Operation
	ZC-8-503-6 - Control System Friction
	- Pitch Damper System
	- Yaw Damper System
	- Yaw Damper-Turn Coordinator Function
	ZC-8-503-7 - Pilot Assist System
	- Pitch "g" Limiter
	- Turn Coordinator Operation
	- Yaw Damper System
	ZC-8-503-8 - Pilot Assist System
	- Pitch Damper System
	- Pitch "g" Limiter
	- Yaw Damper System
	- Yaw Damper-Turn Coordinator Function
	ZC-8-503-9 - Control System Hysteresis Check
	- Pilot Assist System
	- Pitch Damper System
	- Pitch "g" Limiter
	- Turn Coordinator Operation
	- Yaw Damper System
	ZC-8-503-12- Pilot Assist System
	- Pitch "g" Limiter
	- Turn Coordinator Operation
	ZC-8-521 - Automatic Flight Control Systems (Model 8-82) Flight Test Development

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Model YF-102 S/N 53-1780
(Continued)

	Flight Test Items	Resolution of Flight Test Program	
7.	Static Longitudinal Stability	ZC-8-512-15-	
		ZC-8-503-4 -	Static Longitudinal Stability
		ZC-8-503-5 -	Maneuvering Flight Characteristics (Stick Force Per g, $\delta e/g$, etc.)
		-	Programmed Elevator Pulses (C_m Aerodynamic Center)
		ZC-8-503-6 -	Re-evaluated Longitudinal Trim Data
		ZC-8-503-7 -	Elevator Trim (Effect of Load Factors, C.G., Mach No., etc.)
		ZC-8-503-10-	
		ZC-8-526 -	Flight Test Evaluation of YF-102 Stability and Control Characteristics
[redacted]	Longitudinal Control Force	ZC-8-503-5 -	
		ZC-8-526- -	Flight Test Evaluation of YF-102 Stability & Control Characterist
[redacted]	Static Lateral - Directional Stability	Cancelled	
[redacted]	Angle of Attack and Angle of Yaw Pick-up Calibration	ZC-8-512-3 -	Angle of Attack Indicator
		ZC-8-512-4 -	" " " "
		ZC-8-503-3 -	Outside Air Temperature Probes
		-	Relative Wind Transducers
		ZC-8-503-4 -	Angle of Attack Indicators
		-	Outside Air Temperature Source for HAC
		ZC-8-513 -	Angle of Attack Indicators - In-flight Calibration
11.	Airspeed Calibration	Cancelled	

Model YF-102 S/N 53-1780
(Continued)

Flight Test Items	Resolution of Flight Test Program
12. Production O.A.T. Pick-up Survey	ZC-8-503-3 - Outside Air Temperature Probes - Relative Wind Transducers ZC-8-503-4 - Angle of Attack Indicators - Outside Air Temperature Source for HAC Fire Control Computer
[redacted] 13. ATM Power Capability	ZC-8-503-11- Cabin Pressure-Buffeting - Low Pressure Pneumatic System Air Turbine Motor ZC-8-503-12- Cabin Pressure-Buffeting - Low Pressure Pneumatic System Air Turbine Motor [redacted]
14. Throttle Idle Stop Adjustment Check	Cancelled by Addendum 5
15. Cooling Capability of ATM System	ZC-8-503-4 - Cabin Pressure Schedule - Cabin Temperatures ZC-8-503-5 - Cabin Pressure Regulator ZC-8-503-6 - Cabin Pressure Regulator ZC-8-503-7 - " " " ZC-8-503-8 - " " " ZC-8-503-9 - " " "
16. ATM System Stability	ZC-8-512-6 - Low Pressure Pneumatic System - Simulated MG-3 Fire Control System & Air Turbine Motor ZC-8-512-13- Low Pressure Pneumatic System - Simulated MG-3 Fire Control System & Air Turbine Motor ZC-8-503-11- Cabin Pressure - Buffeting - Low Pressure Pneumatic System Air Turbine Motor ZC-8-517 - Low Pressure Pneumatic System Air Turbine Motor

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Model YF-102 S/N 53-1780
(Continued)

	Flight Test Items	Resolution of Flight Test Program
17.	Ground Operation of the ATM	ZC-8-512-14 - Low Pressure Pneumatic System Air Turbine Motor
18.	Watertightness Test	Cancelled
19.	Power Control Tests	Cancelled
20.	Engine Oil System Tests	Cancelled
21.	Oil Cooler Tests	Cancelled
22.	Altitude and Fuel Cell Pressure Tests	Cancelled
23.	Fuel Availability	Cancelled
24.	Emergency Operation of Landing Gear	Cancelled
25.	Normal Operation of Landing Gear	Cancelled
26.	Electrical System	Cancelled
	Addenda To Flight Test Program	
1.	Control System Ground Test	ZC-8-503-9 - Cancelled
2.	Boundary Layer Inlet Duct Recovery	Cancelled
3.	Temperature of Engine Outer Wall and Nozzle Actuators	Transferred to YF-102 USAF S/N 53-1783
4.	Flight Test of Pitch "g" Limiter	ZC-8-512-1 - Pitch "g" Limiter ZC-8-512-26- Non-linear Pitch Rate Network - Pitch "g" Limiter - Shelley Unit Static Pressure Source Evaluation

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Model YF-102 S/N 53-1780
(Continued)

Flight Test Items	Resolution of Flight Test Program
Addenda To Flight Test Program - Continued	
4. Flight Test of Pitch "g" Limiter — Continued	ZC-8-503-4 - Pitch Damper System - Pitch "g" Limiter
	ZC-8-503-5 - Pitch Damper System - Pitch "g" Limiter - Yaw Damper System
	ZC-8-503-7 - Pilot Assist System - Pitch "g" Limiter - Turn Coordinator Operation - Yaw Damper System
	ZC-8-503-8 - Pilot Assist System - Pitch Damper System - Pitch "g" Limiter - Yaw Damper System - Yaw Damper-Turn Coordinator Function
	ZC-8-503-9 - Control System Hysteresis Check - Pilot Assist System - Pitch Damper System - Pitch "g" Limiter - Turn Coordinator Operation - Yaw Damper System
	ZC-8-503-10- Full Authority Pilot Assist Sys - Pitch "g" Limiter - Yaw Damper System
	ZC-8-503-11- Pilot Assist System - Pitch "g" Limiter
	ZC-8-503-12- Pilot Assist System - Pitch "g" Limiter - Turn Coordinator Operation
	ZC-8-531 - Flight Test Development of the Pitch "g" Limiter
5. Throttle Idle Stop Adjustment	Cancellation of Item #14

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Model YF-102 S/N 53-1780
(Continued)

Flight Test Items	Resolution of Flight Test Program
Addenda To Flight Test Program	
6. Nose Gear Lift Off Speed	Cancelled due to design change
7. Engine Exhaust Nozzle Manual Override	ZC-8-512-14- Nozzle Manual Over-ride
	ZC-8-517 - Low Pressure Pneumatic System Air Turbine Motor
8. Roll Rate Limiter Development	ZC-8-512-24

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Model YF-102 S/N 53-1781
Final Summary of Flight Test Program
ZC-8-112 Dated 12 May 1954

Flight Test Items	Resolution of Flight Test Program
1. Armament Bay Door Operation	ZC-8-504-1 - Effect of Missile Bay Door Operation - Missile Bay Door Operation
	ZC-8-504-2 - Effect of Simulated Missile Firings - Cycling of Missile Launching System
	ZC-8-504-3 - Effect of Ram Air Turbine Extension
	ZC-8-504-4 - Effect of Missile Launching - First Missile Launching From F-102
	ZC-8-504-6 - Effect of Missile Launching - Missile Launching Operations - Rocket Exhaust Contamination (Aircraft and Power Plant)
	ZC-8-504-7 - Effect of Rocket Package Extension and Rocket Firing - Missile Exhaust Contamination - Rocket Armament Gear, Functional Check of
2. Armament Launcher Extension Arm Operation	ZC-8-504-2 - Effect of Simulated Missile Firings - Cycling of Missile Launching Sys [redacted]
	ZC-8-504-4 - Effect of Missile Launching - First Missile Launching From F-102
	ZC-8-504-6 - Effect of Missile Launching - Missile Launching Operations - Rocket Exhaust Contamination (Aircraft and Power Plant)
	ZC-8-504-7 - Effect of Rocket Package Extension and Rocket Firing - Missile Exhaust Contamination - Rocket Armament Gear, Functional Check of
	ZC-8-504-8 - Effect of Extended Forward Missiles - Missile Aerodynamic Loads, Intervalometer Operation

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Model YF-102 S/N 53-1781
(Continued)

Flight Test Items	Resolution of Flight Test Program
2. Armament Launcher Extension Arm Operation - Continued	ZC-8-504-9 - Effect of Missile Launching - Missile Exhaust Contamination (Aircraft) - Missile Launcher Operation, Cycle Time, Separation and Aerodynamic Loads
	ZC-8-512-5 - Missile Bay Door Rod End Loads ZC-8-512-11- " " " " " "
3. Missile Bay and Forward Fuse Airloads	Cancelled
4. Captive Missile Loads	ZC-8-504-8 - Effect of Extended Forward Missiles - Missile Aerodynamic Loads, Inter valometer Operation
	ZC-8-504-9 - Effect of Missile Launching - Missile Exhaust Contamination (Aircraft) - Missile Launcher Operation, Cycl Time, Separation and Aerodynamic Loads
5. Missile Separation	ZC-8-504-4 - First Missile Launching From F-102
	ZC-8-504-6 - Missile Launching Operations - Rocket Exhaust Contamination (Aircraft and Power Plant)
	ZC-8-504-8 - Missile Aerodynamic Loads, Intervalometer Operation
	ZC-8-504-9 - Missile Exhaust Contamination (Aircraft) - Missile Launcher Operation, Cycle Time, Separation & Aerodynamic Loads

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Model YF-102 S/N 53-1781
(Continued)

Flight Test Items	Resolution of Flight Test Program
5. Missile Separation - Continued	ZC-8-512-1 - Missile Firing Under Positive "g" Conditions ZC-8-512-19- Missile Separation & Trajectory
6. Missile Trajectory	ZC-8-504-4 - First Missile Launching From F-102 ZC-8-504-6 - Missile Launching Operations - Rocket Exhaust Contamination (Aircraft and Power Plant) ZC-8-504-9 - Missile Exhaust Contamination (Aircraft) - Missile Launcher Operation, Cycl Time, Separation and Aerodynamic Loads ZC-8-512-2 - Missile Flight Path Corrected for Wind Velocity & Direction
7. Rocket Separation and Dispersion	ZC-8-504-7 - Effect of Rocket Package Extension and Rocket Firing - Missile Exhaust Contamination - Rocket Armament Gear, Functional Check of ZC-8-512-23- Missile Firing Summary ZC-8-512-25- Rocket Dispersion
8. Airspeed System Calibration	Cancelled
9. Dynamic Response to Armament Operation	ZC-8-504-1 - Effect of Missile Bay Door Operation - Missile Bay Door Operation ZC-8-504-2 - Effect of Simulated Missile Firings - Cycling of Missile Launching System ZC-8-504-3 - Effect of Ram Air Turbine Extension ZC-8-504-4 - Effect of Missile Launching - First Missile Launching from F-102

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Model YF-102 S/N 53-1781
(Continued)

Flight Test Items	Resolution of Flight Test Program
9. Dynamic Response to Armament Operation - Continued	ZC-8-504-6 - Effect of Missile Launching - Missile Launching Operations - Rocket Exhaust Contamination (Aircraft and Power Plant) ZC-8-504-7 - Effect of Rocket Package Extension and Rocket Firing - Missile Exhaust Contamination - Rocket Armament Gear, Functional Check of ZC-8-504-8 - Effect of Extended Forward Missiles - Missile Aerodynamic Loads, Intervalometer Operation ZC-8-504-9 - Effect of Missile Launching - Missile Exhaust Contamination (Aircraft) - Missile Launcher Operation, Cycle Time, Separation and Aerodynamic Loads
10. Rocket and Missile Exhaust Gas Effects	ZC-8-504-2 - Missile Exhaust Contamination of Cockpit Air - Cycling of Missile Launching Sys. ZC-8-504-4 - Power Plant Operation - Effect of Missile Exhaust Gases - Sealed Cockpit During Missile Launching - First Missile Launching From F-1 ZC-8-504-6 - Power Plant Operation - Effect of Missile Exhaust Gases - Sealed Cockpit During Missile Launching - Missile Launching Operations - Rocket Exhaust Contamination (Aircraft and Power Plant) ZC-8-504-7 - Rocket Contamination of Cockpit Air - Missile Exhaust Contamination (Aircraft and Power Plant)

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Model YF-102 S/N 53-1781
(Continued)

Flight Test Items	Resolution of Flight Test Program
10. Rocket and Missile Exhaust Gas Effects - Continued	ZC-8-504-7 - Rocket Armament Gear, Functional Check of
	ZC-8-504-9 - Missile Exhaust Contamination of Cockpit Air
	- Missile Exhaust Contamination (Aircraft)
	- Missile Launcher Operation, Cycle Time, Separation and Aerodynamic Loads
	ZC-8-512-17- Missile Exhaust Gas Summary
11. Fuel Venting Pressure Measurements	ZC-8-504-6 - Fuel Venting During Missile Launching Operation
	ZC-8-504-7 - Fuel Venting During Negative "g" Maneuvers
	ZC-8-504-8 - Fuel Venting During Negative "g" Maneuvers
	ZC-8-512-1 - Modifications to Reduce Fuel Venting
12. Preliminary Ram Air Turbine Tests	ZC-8-504-1 - Effect of Missile Bay Door Operation
	- Ram Air Turbine Operation
	ZC-8-504-2 - Effect of Simulated Missile Firings
	- Ram Air Turbine Operation
	ZC-8-504-3 - Effect of Ram Air Turbine Extension
	- Ram Air Turbine Operation

Addenda To Flight Test Program

1. Armament Launcher Operation	See Flight Test Item #2

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Model YF-102 S/N 53-1781
(Continued)

Flight Test Item	Resolution of Flight Test Program
Addenda to Flight Test Program - Continued	
2. Missile Separation and Trajectory at V_{max} and Hp_{max}	See Flight Test Items 5 and 6
3. Two-inch Rocket Package	See Flight Test Item 7

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Model YF-102 S/N 53-1781
Final Summary of Flight Test Program
ZC-8-131 Dated 1 September 1955

Flight Test Items	Resolution of Flight Test Program	
1. Tether Test Firing	ZC-8-512-64 -	Tether Test Firing
	ZC-8-536 -	MB-1 Rocket Launch, Pylon Configuration
2. Jettison Inert Rocket	ZC-8-512-48 -	Inert Ding Dong Jettison
	ZC-8-536 -	MB-1 Rocket Launch Pylon Configuration
3. Separation and Trajectory	ZC-8-512-64 -	Ignition Time Delay
	ZC-8-512-68 -	Rocket Ignition and First Movement Sequence
	-	Rocket Separation and Trajectory
	ZC-8-536 -	MB-1 Rocket Launch Pylon Configuration, (Separation and Trajectory)
4. Airplane Response to Rocket Firing	ZC-8-512-68 -	Airplane Response
	ZC-8-536 -	MB-1 Rocket Launch Pylon Configuration
	ZC-8-612-64 -	Engine Operation During Firing
	ZC-8-512-68 -	Engine Operation During Firing

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Model YF-102 S/N 53-1781
(Continued)

Flight Test Items	Resolution of Flight Program
5. Blast Effects of Firing Ding Dong	ZC-8-512-64 - Panel Strains
	ZC-8-512-68 - Panel Strains - Fuselage Static Pressures
	ZC-8-512-51 - Sound Pressure Levels During MB-1 Rocket Firing
	ZC-8-536 - MB-1 Rocket Launch Pylon Configuration
6. Engine Snuffing	Transferred to S/N 53-1797
7. Exhaust Dust Effects	ZC-8-512-64 - Rocket Exhaust Dust Deposit
	ZC-8-512-68 - Exhaust Products Concentration
	ZC-8-536- MB-1 Rocket Launch Pylon Configuration
Addenda To Flight Test Program	
Addendum #1 - Determination of Temperature Envelope Permitting the Air Firing of the Type MB-1 Rocket (MB-1 Motor Skin Temp.)	ZC-8-512-52 - Rocket Temperature Environment Study
	ZC-8-512-55 - Rocket Temperature Environment Study
	ZC-8-512-64 - Boundary Layer and Skin Temperatures
	ZC-8-512-68 - Boundary Layer and Skin Temperatures
	ZC-8-536 - MB-1 Rocket Launch, Pylon Configuration

Model YF-102 S/N 53-1782
Final Summary of Flight Test Program
ZC-8-113 Dated 19 May 1954

Flight Test Items	Resolution of Flight Test Program
1. Antenna - Functional Characteristics - AN/ARN-18 Glide Path	ZC-8-505-3 - Radar Antenna ZN-152-05 - Antenna Flight Testing for YF-102 Aircraft
2. Antenna - Functional Characteristics - AN/ARN-14 OMNI	ZC-8-505-1 - Radar Antenna ZC-8-505-2 - Electrical Power Controller - Radio Equipment ZN-152-05 - Antenna Flight Testing for YF-102 Aircraft
3. Antenna - Functional Characteristics - AN/ARN-25 UHF-DF	ZC-8-505-3 - Radar Antenna - Radio Equipment ZN-152-05 - Antenna Flight Testing for YF-102 Aircraft
4. Antenna - Functional Characteristics - AN/ARC-34	ZC-8-505-2 - Electrical Power Controller - Radio Equipment ZC-8-505-3 - Radar Antenna - Radio Equipment ZN-152-05 - Antenna Flight Testing for YF-102 Aircraft
5. Antenna - Functional Characteristics - AN/ARN-12	ZC-8-505-2 - Electrical Power Controller - Radio Equipment ZN-152-05 - Antenna Flight Testing for YF-102 Aircraft

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Model YF-102 S/N 53-1782
(Continued)

Flight Test Items	Resolution of Flight Test Program
Addenda To Flight Test Program	
1. Data Link Antenna - Isolation Study	Transferred to YF-102 USAF S/N 53-1783
2. Glide Path Antenna - Location Study	ZC-8-505-3 - Radar Antenna - Radio Equipment
	ZN-152-05 - Antenna Flight Testing for YF-102 Aircraft

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Model YF-102 S/N 53-1783
Final Summary of Flight Test Program
ZC-8-114 Dated 28 June 1954

Flight Test Items	Resolution of Flight Test Program
1. Angle of Attack and Angle of Yaw Pick-up Calibration	Cancelled
2. Airspeed Calibration - Subsonic Speed Angle ZC-8-540 - Determination of Static Source Post. Error	ZC-8-506-9 - Re-calibration after Nose Cone Change - Pacer Method
3. Outside Air Temperature Pick-up Location Survey	ZC-8-506-7 - Nordon Flush Type Temperature Probe Completed on YF-102 USAF S/N 53-1780
4. Ground Test of Wing, Tail, and Engine Inlet Anti-icing System	ZC-8-506-6 - Engine Inlet Surface Anti-icing System - Windshield (NESA) De-fog System
5. Wing, Tail, and Engine Inlet Anti-icing System and Windshield (NESA) Anti-icing System (Climb and Descent Tests)	ZC-8-506-6 - Engine Inlet Surface Anti-icing System - Windshield (NESA) De-fog System ZC-8-506-7 - Engine Inlet Scoop Anti-icing System - Hot Air De-fog System - Windshield (NESA) De-fog System ZC-8-506-8 - Engine Inlet Scoop Anti-icing System - Radome Anti-icing System
6. Wing, Tail, and Engine Inlet Anti-icing System and Windshield (NESA) Anti-icing System (Level Flight Tests)	ZC-8-506-6 - Engine Inlet Surface Anti-icing System - Windshield (NESA) De-fog System

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Model YF-102 S/N 53-1783
(Continued)

Flight Test Items	Resolution of Flight Test Program	
Addenda To Flight Test Program - Continued		
2. Windshield Canopy De-fog System Test	ZC-8-506-6	- Engine Inlet Surface Anti-icing System - Windshield (NESA) De-fog System
	ZC-8-527	- Low Pressure Pneumatic System - Cockpit Air Conditioning and Pressurization
[redacted]A. Cabin Air Temperatures	ZC-8-506-4	- Distribution of Inlet Air Throughout Cockpit
	ZC-8-506-5	- Cabin Air Conditioning System Check
	ZC-8-506-6	- Cabin Air Conditioning System Check
	ZC-8-506-7	- Cabin Temperature Regulator
	ZC-8-527	- Low Pressure Pneumatic System - Cockpit Air Conditioning and Pressurization
[redacted]. Modifications Required on Model 8 Aircraft S/N 53-1783 Due to Reassignment of Certain Antenna Flight Tests	ZC-8-506-2	- Radar Antenna
	ZC-8-506-3	- " "
	ZC-8-506-5	- Radio Antenna - Landing Approach [redacted]
	ZN-152-05	- Antenna Flight Testing for YF-102 Aircraft
5. Performance of Ram Air Turbine Emergency Hydraulic Power System	ZC-8-506-1	- Ram Air Turbine Operation - General System Operation
	ZC-8-506-2	- Ram Air Turbine Operation
	ZC-8-514	- Ram Air Turbine Emergency Hydraulic Power System Evaluation, F-102 Series 8-82
	ZC-8-516	- Hydraulic System Oil Temperature, YF-102 and YF-102A

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Model YF-102 S/N 53-1783
(Continued)

Flight Test Items	Resolution of Flight Test Program
6. Wing, Tail, and Engine Inlet Anti-icing System and Windshield (NESA) Anti-icing System (Level Flight Tests)	ZC-8-506-7 - Engine Inlet Scoop Anti-icing System - Hot Air De-fog System - Windshield (NESA) De-fog System ZC-8-506-8 - Engine Inlet Scoop Anti-icing System - Radome Anti-icing System
7. Radome Anti-icing	ZC-8-506-8 - Engine Inlet Scoop Anti-icing System - Radome Anti-icing System ZC-8-506-9 - " " " "
9. Heating and Ventilating Equipment Specification Checkout	Cancelled
10. Cabin Pressure Specification Checkout	ZC-8-506-4 - Distribution of Inlet Air Throughout Cockpit
11. Power Plant and Structural Cooling Specification Compliance	Cancelled
12. Dynamic Response	Not Required
13. Pilot Assist System	Not Required
Addenda To Flight Test Program	
1A. Temperature of Engine Outer Wall and Nozzle Actuators	ZC-8-506-2 - Effectiveness of Afterburner Blanket in Reducing Afterburner Nozzle Actuator Temperatures ZC-8-506-3 - Effectiveness of Afterburner Blanket in Reducing Afterburner Nozzle Actuator Temperatures ZC-8-522 - Engine & Structural Cooling System Evaluation (YF-102)

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Model YF-102 S/N 53-1783
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Flight Test Items	Resolution of Flight Test Program
Addenda To Flight Test Program - Continued	
5A. Performance of Modified Ram Air Turbine Emergency Hydraulic Power System	ZC-8-506-8 - Ram Air Turbine Operation ZC-8-514 - Ram Air Turbine Emergency Hydraulic Power System Evaluation, F-102 Series 8-82 ZC-8-516 - Hydraulic System Oil Temperature YF-102 and YF-102A
6. Fin Ram Air Inlet Pressure and Temperature Measurements	Cancelled

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Model YF-102A S/N 53-1787
Final Summary of Flight Test Program
ZC-8-110 Dated 2 December 1954

Flight Test Items	Resolution of Flight Test Program	
A. Phase I Flight Tests		
Stage I Ground Tests		
1.01 Engine Operation	ZC-8-530	Flight Test Evaluation of the J57-P-11 and P-41 Engine Installations
1.03 Static Thrust Calibration		
Stage III Airspeed Calibration		
3.01 Airspeed Calibration (Pacer Method)	ZC-8-512-86	Five-Foot Test Nose Boom Assembly
3.02 Airspeed Calibration (Fly-by Method)	ZC-8-537	Flight Test Evaluation of Total and Static Pressure Sources on the Fuselage
State IV Airplane Performance		
4.02 Level Flight	EFT-A-8-220	Evaluation of the Lewis Flush-Mounted Outside Air Temperature Probe
	ZC-8-512-75	Outside Air Temperature Probe Evaluation (Giannini)
Phase VI Systems Development		
6.04 Ram Air Turbine Tests	ZC-8-532	Ram Air Turbine Performance Flight Test Evaluation
6.05 Pilot Assist System	ZC-8-531	Flight Test Development of the Pitch-G-Limiter
Addendum Items		
Addendum No. 1 High Speed Vibration (Buzz) Investigation	ZC-8-542	Flight Test Investigation Inlet Duct Instability and Cockpit Sound Level
Addendum No. 2 Duct Rumble Investigation		

External Fuel Tank Tests
ZC-8-173 Dated 19 January 1956

Flight Test Items	Resolution of Flight Test Program	
Group 2.00 Flight Characteristics		
2.03 Fuel Venting and Fuel Transfer Tests	ZC-8-512-69	External Fuel Tank Vent Pressure Study

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Model YF-102A S/N 53-1787
Summary of Flight Test Program
ZC-8-118 Dated 2 December 1954

Flight Test Items	Resolution of Flight Test Program	
Stage II Aircraft System Shakedown		
2.01 Shakedown Flights	ZC-8-511-1	Summary of Results
	ZC-8-516	Hydraulic System Oil Temperatures
2.02 Flight Control System Operation	ZC-8-511-1	Flight Control System
2.03 Emergency Fuel Control	Cancelled	
2.04 Engine and Structural Cooling	EFT-A-8-476	Flight Test Evaluation of the Engine Cooling System
2.05 Brake Parachute Operation	Accomplished on F-102A USAF S/N 53-1791	
2.06 Automatic Flight System	ZC-8-511-1	Summary of Results
2.07 Control Sensitivity	Transferred to F-102A USAF S/N 53-1791	
Stage IV Airplane Performance		
4.01 Take-off	EFT-A-8-17	Summary of Performance Test Completed During Phase I Flight Tests
4.02 Climb	EFT-A-8-328	Wing Fence Evaluation
4.03 Level Flight	EFT-A-8-17	Summary of Performance Test Completed During Phase I Flight Tests
	EFT-A-8-328	Wing Fence Evaluation
4.04 Dive	Transferred to F-102A USAF S/N 53-1791	
4.05 Landing	EFT-A-8-17	Summary of Performance Test Completed During Phase I Flight Tests
Stage V		
5.01 General Handling Characteristics	ZC-8-511-1	Flight and Handling Characteristics
5.02 Dynamic Longitudinal Stability	ZC-8-512-6	Dynamic Longitudinal Stability
5.03 Static Longitudinal Stability	ZC-8-512-3	Static Longitudinal Stability
5.04 Maneuvering Flight Characteristics	ZC-8-512-5	Maneuvering Flight Characteristics
	ZC-8-512-33	Maneuvering Flight Characteristics
	ZC-8-512-34	Buffet Investigation
5.05 Dynamic Directional Stability	ZC-8-512-20	Dynamic Directional Stability
5.06 Static Directional Stability	ZC-8-512-27	Static Directional Stability
5.07 Lateral Control	ZC-8-512-21	Lateral Control Effectiveness

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Model YF-102A S/N 53-1787
(Continued)

Flight Test Items	Resolution of Flight Test Program
Stage VI Systems Development	
6.01 Boundary Layer Bleed Air	Cancelled
6.03 Low Pressure Pneumatic System	Cancelled
6.05 Pilot Assist System	ZC-8-512-8 Turn Coordinator Operation ZC-8-512-28 Turn Coordinator Operation ZC-8-512-28 Trim Servo Operation
Stage VII Dynamic Response	ZU-8-055 (Rev. B) F-102 Aircraft Aileron Roll Maneuver Study Final Report
Stage VIII Stability and Control	
8.01 Dynamic Longitudinal Stability	Transferred to F-102A USAF S/N 53-1791
8.02 Static Longitudinal Stability	"
8.03 Elevator Control Effectiveness	"
8.04 Maximum C_L Landing	"
8.05 Take-off	"
8.06 Maximum Load Factors	"
8.07 High Speed Dive Requirements	"
8.08 High Speed Flight Requirements	"
8.10 Maneuvering Flight Characteristics	"
8.11 Dynamic Lateral-Directional Stability	"
8.12 Static Directional Stability	"
8.13 Adverse Yaw	"
8.14 Rudder Control Effectiveness	"
8.15 Aileron Characteristics	"
8.16 Stall	"
8.17 Trimming Devices	"
Addendum Item	
Addendum No. 3 Engine Compressor Stall Testing	Transferred to F-102A S/N 54-1380

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Model YF-102A S/N 53-1787
Final Summary of Tailpipe Ejector Flight Test Program
ZC-8-177 Dated 23 January 1956

Flight Test Items	Resolution of Flight Test Program
A. Subsonic Performance	ZC-8-512-70 Effect of Revised Cooling Shroud on Subsonic Performance
B. Maximum Afterburner Performance	ZC-8-512-71 Effect of Revised Cooling Shroud on Performance During Afterburner Operation
	ZC-8-512-81 Base Pressure Coefficient During Tailpipe Ejector Tests
	ZC-8-512-83 Effect of Tailpipe Ejector on Cooling Air Flow

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Model YF-102A S/N 53-1788
Final Summary of Flight Test Program
ZC-8-119 Dated 11 February 1955

Flight Test Program Items	Resolution of Flight Test Program
1. Armament Hardware	ZC-8-512-8 - Armament Launcher Operation -11 - Missile Bay Door Rod End Loads -21 - Armament Launcher Operation - Fast Acting -36 - Armament Pneumatic System Pressures -4,-5, -10 - Fast Acting Launcher Operation
2. Missile Separation and Trajectory	ZC-8-512-54 - Missile Trajectory ZC-8-512-13 - GAR-1 Firing EFT-A-8-414 - Armament Firing Summary EFT-A-8-330 - GAR-1 Falcon Missile Separation and Trajectory
3. Rocket Separation (2.00 Inch)	ZC-8-512-4 - Rocket Firing Circuit - Rocket Ground Test Firing -10 - 2.00 Inch Rocket Ground Firing -12,-24 - 2.00 Inch Rocket Firing EFT-A-8-414 - Armament Firing Summary
4. Rocket Trajectory and Dispersion	ZC-8-512-12 - 2.00 Inch Rocket Firing -23 - 2.00 Inch Rocket Firing In-Flight -25 - Rocket Dispersion -40 - Rocket Firing Summary -45 - 2.00 Inch Rocket Firing -48 - 2.00 Inch Rocket Dispersion
5. Armament Bay Loads	ZC-8-512-11 - Missile Bay Door Rod End Loads
6. Airplane Dynamic Response to Armament Operation	ZC-8-512-57, -75, EFT-A-8-219 - Angle of Attack Calibration

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Model YF-102A S/N 53-1788
(Continued)

Flight Test Program Items	Resolution of Flight Test Program
7. Armament Exhaust Dust Effects	ZC-8-512-17 - Missile Exhaust Gas Summary
Addenda to Flight Test Program	
Addendum #1 Visual Evaluation of Windshield Rain Clearing System	Transferred
Addendum #2 Drag Chute Deployment Time	Transferred to S/N 53-1787
Addendum #3 Captive Missile	Superseded by 3.A
Addendum #3.A Captive Missile	ZC-8-512-24 - Captive Missile Loads Flight Investigation
Addendum #4 Evaluation of 8-90 Emergency Altenator During Engine Starting	ZC-8-512-36 - Engine Ground Starts - No External Electrical Power
Addendum #5 Missile Trajectory Program	ZC-8-512-54 - Missile Trajectory EFT-A-8-414 - Armament Firing Summary EFT-A-8-330 - GAR - 1 Falcon Missile Separation and Trajectory
Addendum #6 2.75 Inch Rocket Program	EFT-A-8-331 - 2.75 Inch FFAR Ballistic Information EFT-A-8-414 - Armament Firing Summary ZC-8-512-90 - Jump Angle Comparison - 2.00 Inch and 2.75 Inch Rockets (FFAR)

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Model F-102A S/N 53-1791
Final Summary of Flight Test Program
ZC-8-120 Dated 2, May 1955

Flight Test Items	Resolution of Flight Test Program
A. Phase I Flight Tests	
Test Item 2.00 Aircraft Systems Functional Tests	
2.01 Functional Test Flights	ZC-8-542 Flight Test Investigation of Inlet Duct Instability and Cockpit Sound Level ZC-8-512-60 1.5 "g" Pull-up ZC-8-512-66 Main Landing Gear Side Brace Loads ZC-8-512-95 Main Wheel Well Pressure Survey LETTER #FKE-353 (dated 3-12-56) Natural Vibration Frequency of the Landing Gear
2.04 Engine and Structural Cooling	ZC-8-512-55 Afterburner Actuator Temperatures EFT-A-8-445 Flight Test Evaluation of Effect of Insulating Blankets on Afterburner Nozzle Actuator Temperatures
2.05 Brake Parachute Operation	ZC-8-512-94 Drag Chute System Deployment Time
Test Item 4.00 Flight Flutter Tests	
4.01 Determination and Investigation of Critical Airplane Vibration Modes	ZU-8-034 Results of Flight Flutter Tests **ZC-8-512-30 & 31 Low Speed Vibration Investigation**
C. Phase III Flight Tests	
Test Item 7.00 Stability and Control	EFT-A-8-219 Angle of Attack Calibration - True vs. RWT ZC-8-539 Evaluation of Automatic Control System
Test Item 8.00 Noise Level (Cockpit)	ZC-8-542 Flight Test Investigation of Inlet Duct Instability and Cockpit Sound Level

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Model F-102A S/N 53-1791
(Continued)

Flight Test Items	Resolution of Flight Test Programs
Test Item 1.00 Ground Tests	
1.01 Engine Operation	)EFT-A-8-110 Powerplant Characteristics of P-23
1.02 Engine Equipment Operation	) Engines in the F-102A and TF-102A
1.03 Static Thrust Calibration	) From Flight Tests
1.04 Lowspeed Taxi & Lift-off Characteristics	Satisfactory Performance
1.05 Highspeed Taxi & Lift-off Characteristics	Satisfactory Performance
Test Item 2.00 Aircraft Systems Functional Tests	
2.02 Flight Control System Operation	Check-out Only
2.03 Emergency Fuel Control	Cancelled
2.06 Pilot Assist System	Cancelled
2.07 Control Sensitivity	Cancelled
Test Item 3.00 Airspeed Calibration	
3.01 Airspeed Calibration (Pacer Method)	))Cancelled to Permit Initiation of Phase IV
3.02 Airspeed Calibration (Fly-by Method)	)Performance Test Which Accomplished These Items)
Test Item 5.00 Airplane Performance	
5.01 Take-Off	)Cancelled to Permit Initiation of Phase IV
5.02 Climb	)Performance Test Which Accomplished These Items
5.03 Level Flight	)
5.04 Dive	ZC-8-512-127 Aircraft Performance at Supersonic Speeds
5.05 Landing	Cancelled to Permit Initiation of Phase IV Performance Test Which Accomplished This Item

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Model F-102A S/N 53-1791
(Continued)

Flight Test Items	Resolution of Flight Test Programs
Test Item 6.00 Preliminary Flight Characteristics	
6.01 General Handling Characteristics	Transferred to F-102A USAF S/N 53-1798
6.02 Dynamic Longitudinal Stability	"
6.03 Static Longitudinal Stability	"
6.04 Maneuvering Flight Characteristics	"
6.05 Dynamic Directional Stability	"
6.06 Static Directional Stability	"
6.07 Roll Coupling	"
B. Phase II	USAF
C. Phase III Flight Tests	
Test Item 7.00 Stability and Control	
7.01 Dynamic Longitudinal Stability	Transferred to F-102A USAF S/N 53-1798
7.02 Static Longitudinal Stability	"
7.03 Elevator Control Effectiveness	"
7.04 Maximum C_L Landings	"
7.05 Take-Off	"
7.06 Maximum Load Factors	"
7.07 High Speed Dive Requirements	"
7.08 High Speed Flight Requirements	"
7.09 Longitudinal Trim Change	"
7.10 Maneuvering Flight Characteristics	"
7.11 Dynamic Lateral-Directional Stability	"
7.12 Static Directional Stability	"
7.13 Adverse Yaw	"
7.14 Rudder Control Forces and Effectiveness	"
7.15 Aileron Characteristics	"
7.16 Stall Characteristics	"
7.17 Trimming Devices	"

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Model F-102A S/N 53-1791
(Continued)

Flight Test Items	Resolution of Flight Test Programs
Test Item 9.00 Carbon Monoxide (Cockpit)	Transferred to S/N 54-1380
Test Item 10.00 Spin Test	Transferred to S/N 54-1401
Test Item 11.00 Cockpit and Inclosure Water Tightness	Cancelled
Test Item 12.00 Cabin Pressurization	Transferred to S/N 54-1380
Addendum No. I Control System Ground Tests	Transferred to S/N 53-1798

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Model F-102A S/N 53-1791
Final Summary of Flight Test Program
(External Fuel Tank Tests)
ZC-8-173 Dated 19 January 1956

Flight Test Items	Resolution of Flight Test Program
Group 1.00 Ground Tests	
1.01 Ground Vibration Test with External Fuel Tanks Attached	ZU-8-034 ADD. B Results of Flight Flutter Tests
Group 2.00 Flight Characteristics	
2.01 Stability	ZC-8-512-124 General Handling Characteristics
2.02 Jettison	EF-8-009 External Fuel Tank Jettison Tests (film) EFT-A-8-394 External Fuel Tank Jettison Tests
2.03 Fuel Venting and Fuel Transfer Tests	EFT-A-8-395 Effect of External Fuel Tanks on Fuel Vent Pressure and External Tank Transfer Evaluation
2.04 Flight Flutter Test	ZU-8-034 ADD B Results of Flight Flutter Tests
2.05 Flight Performance Test	Cancelled to permit initiation of Phase IV performance tests which accomplished this item

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Model F-102A S/N 53-1791
Final Summary of Flight Test Program
ZC-8-286 Dated 3 May 1957

Flight Test Program Items	Resolution of Flight Test Program
1.0 Verification of WADC Advanced Instrument Display	EFT-A-8-474 - Flight Test Results of ARDC Advanced Cockpit Instrument Display
1.01 Calibration and Alignment	EFT-A-8-474 - Flight Test Results of ARDC Advanced Cockpit Instrument Display
1.02 Utility and Operation of Instrument Display	EFT-A-8-474 - Flight Test Results of ARDC Advanced Cockpit Instrument Display
2.0 Determine Accuracy of Recorded Intelligence Supplied by Air Data Computer	Cancelled

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Model F-102A S/N 53-1793
Final Summary of Flight Test Program
ZC-8-242 Dated 14 May 1956 (Phase III)

Flight Test Program Items		Resolution of Flight Test Program	
1.0	Armament Hardware		
1.01	Structural Qualification of Armament Displacement Hardware (preliminary)	EFT-A-8-391, ZC-8-512-82	- Armament Bay Door Flutter Tests With Debris Deflectors
		ZC-8-512-38	- Armament Gear Operation - In Flight
		ZC-8-512-35,44	Armament Gear Loads
		ZC-8-512-[illegible]	- Armament Gear Cycling Time
1.02	24-Rocket Door Configuration	EFT-A-8-391	- Armament Bay Door Flutter Tests With Debris Deflectors
1.03	High Pressure Pneumatic System	ZC-8-512-117	- Pneumatic Pressure Requirements For Armament Operation
2.0	Armament Firing	EFT-A-8-414	- Armament Firing Summary
2.01	Missile Separation	EFT-A-8-406B	- GAR-1 Missile Separation and Trajectory
2.02	Rocket Separation and Airplane Launching Stability	EFT-A-8-383B	- 2.75 Inch FFAR Separation, Dispersion and Firing
		EFT-A-8-443	- 2.75 Inch FFAR Jump Angle Determination
2.03	Missile and Rocket Blast Effects	EFT-A-8-383B	- 2.75 Inch FFAR Separation Dispersion and Firing
		ZC-8-512-79	- Missile Bay Pressures
2.04	Armament Bay Air Loads	EFT-A-8-363	- Missile Launcher Air Loads
2.05	Temperature Rise of Aft Hardware	Cancelled	
3.0	Relative Wind Transducer (RWT)		
3.01	To Obtain Inflight Calibration of RWT vs α_t		Transferred to S/N 53-1791

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Model F-102A S/N 53-1793
Final Summary of Flight Test Program
ZC-8-242 Dated 14 May 1956 (Phase III)

Flight Test Program Items (Cont'd)	Resolution of Flight Test Program
3.02 Qualitative Evaluation of RWT Calibration from Photopanel and Oscillograph Records	Transferred to S/N 53-1791
Addendum to Flight Test Program	
Addendum I	
Full Salvo 2.75 Inch FFAR Demonstration (Motion Picture)	MP-57-16-8 - F-102A Armament 2.75 Inch Rocket

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Model F-102A S/N 53-17[illegible]
Final Summary of Flight Test Program
ZC-8-122 Rev. 3 Dated 11 February 1957

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Flight Test Items	Resolution of Flight Test Program	
1. Wing Symmetrical Flight Loads	ZU-8-017	Results of Flight Air Loads Measurements on the F-102A Airplane Volume I Symmetrical Wing Loads (Clean Configuration)
	ZC-8-525-6, 7, 8, 9, 10, 11, 12, 13	Progress Report F-102A Flight Loads Program
2. Wing Asymmetrical Flight Loads	ZC-8-525-6, 7, 8, 9, 10, 11, 12, 13	Progress Report F-102A Flight Loads Program
3. High Drag Conditions (with external fuel tanks)	ZC-8-525-14, 15, 16	Progress Report F-102A Flight Loads Program
4. Supplemental Flight Loads, Wing Symmetrical	ZU-8-017	Results of Flight Air Loads Measurements on the F-102A Airplane Volume I Symmetrical Wing Loads (Clean Configuration)
	ZC-8-525-6, 7, 8, 9, 10, 11, 12, 13	Progress Report F-102A Flight Loads Program
Rudder Induced Flight Loads	ZC-8-525-13	Progress Report F-102A Flight Loads Program
Supplemental Flight Loads - Wing Asymmetrical	ZC-8-525-12, 13, 14	Progress Report F-102A Flight Loads Program

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NOTE: All flight test items are covered by Report ZU-8-017, Volumes I, II, III, and IV.

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Model YF-102C S/N 53-1797
Final Summary of Flight Test Program
ZC-8-166 Dated 1 February 1956

Flight Test Items	Resolution of Flight Test Program
1.00 Armament Hardware	
1.01 Armament Hardware Check	
Ding Dong Extension Gear Operation	ZC-8-535A - Flutter Characteristics of Armament Bay Door
Bay Door Loads and Accelerations	ZC-8-535A - Flutter Characteristics of Armament Bay Door
1.02 Rocket Fly Around Check	Cancelled
1.03 Rocket Jettison Check	Cancelled
2.00 Ding Dong Firing	
2.01 Separation and Trajectory	EF-8-004 - Film report: MB-1 Rocket Firings (Launcher Rail) ZC-8-533 - Rocket Ignition and Movement Sequences - Rocket Separation and Trajectory EFT-A-8-296 MB-1 Rocket Jump Angle Determination and Ballistic Characteristics EFT-A-8-457 MB-1 Rocket Ballistic Characteristics
2.02 Blast Effects	ZC-8-533 - Fuselage Boundary Layer and Skin Temperatures - Fuselage Boundary Layer Total Pressures - Fuselage Skin Panel Strain
2.03 Airplane Response to Ding Dong Firing	ZC-8-533 - Aircraft Dynamic Response EFT-A-8-457 Aircraft Dynamic Response
2.04 Engine Response to Firing	ZC-8-533 - Engine Operation During Rocket Launch EFT-A-8-457 Engine Response to Rocket Firing
2.05 Exhaust Dust Effects	ZC-8-533 - Rocket Motor By-Products
3.00 XMG-14 System	
3.01 XMG-14 Checkout	Cancelled

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Model YF-102C S/N 53-1797
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4.00 Ding Dong Firing With XMG-14 4.01 Ding Dong Firing at Targets	Cancelled

Addenda to Flight Test Program

Addendum #1 Main Landing Gear Wing Fairing Link Loads	ZC-8-512-89 - Main Landing Gear Operation EFT-A-8-307A- Landing Gear Operation Time and Main Landing Gear Wing Fairing Link Loads

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Model F-102A S/N 53-1798
Final Summary of Flight Test Program
ZC-8-157 Dated 11 October 1955

Flight Test Items		Resolution of Flight Test Program	
PHASE I			
1.01	Hysteresis & Breakout Forces	10035	Aircraft Flight Control System Determination of Dynamic Characteristics
1.02	Elevon System Operation	10035	Aircraft Flight Control System Determination of Dynamic Characteristics
1.03	Elevon System Operation	10035	Aircraft Flight Control System Determination of Dynamic Characteristics
2.01	Pilot Assist System	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
2.02	Turn Coordinator	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
2.03	Manual Mode Yaw-Damper	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
2.04	Manual Mode Pitch-Damper	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
PHASE III			
3.01	Pilot Assist Development	Cancelled - Decision Made to Use System Developed by Hughes	
3.02	Turn Coordinator Development	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
3.03	Pitch Damper Development	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
3.04	Yaw Damper Development	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
3.06	Components Development	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
		ZC-8-531	Flight Test Development of the Pitch g Limiter
4.01	Dynamic Longitudinal Stability	ZC-8-539	Evaluation of the F-102A Automatic Flight Control System
4.02	Static Longitudinal Stability	EFT-A-8-304A	AFT CG Flight Test

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Model F-102A S/N 53-1798
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Flight Test Items		Resolution of Flight Test Program
4.03	Elevator Control Effectiveness	* No buzz-fix on this airplane, intent of this item was satisfied by F-102A flight loads program
4.04	Maximum Load Factors	
4.05	Maximum C_L Landings	* ZC-8-512-139 Landing and Take-Off Performance with Skewed Axis Main Landing Gear
4.06	Take Off	
4.07	High Speed Flight Requirements	* No buzz-fix on this airplane, intent of this item was satisfied by F-102A flight loads program
4.08	Longitudinal Trim Characteristics	* Not Demonstrated
4.09	Maneuvering Flight Characteristics	* ZC-8-552, Stability and Control Demonstration of the F-102A With the Case XX Wing
4.10	Dynamic Lateral Directional Stability	Transferred to AF 54-1390
4.11	Static Directional Stability	Transferred to AF 54-1390
4.12	Adverse Yaw	Transferred to AF 54-1390
4.13	Rudder Control & Effectiveness	Transferred to AF 54-1390
4.14	Aileron Characteristics	Transferred to AF 54-1390
4.15	Stall Characteristics	Transferred to AF 54-1401
Add #1	Production Airspeed Calibration	EFT-A-217 Position Error Calibration for the Production Nose Boom Airspeed System
Add #3	O.A.T. Probe Calibration	EFT-A-220 Evaluation of the Lewis Flush-Mounted Outside Air Temperature Probe
Add #4	Ram Air Turbine Check-out & Performance	ZC-8-532 Ram Air Turbine Flight Test Performance
Add #2	Special Engine Fuel Control (Evaluated on S/N 54-1380)	EFT-A-8-270A Linear Thrust Fuel Control Evaluation
Add #5	Drag Chute Deployment Times (Test accomplished on S/N 54-1791)	ZC-8-512-84 Drag Chute System Deployment Time
Add #6	Reduced Ground Steering Forces	ZC-8-512-130 Tests of Reduced Ground Steering Forces
Add #7	Landing Gear Extension & Loads Tests	EFT-A-8-305A Landing Gear Extension & Loads Test EFT-A-8-307A Landing Gear Operating Time & Main Landing Gear Fairing Link Loads

* Tests were not completed on this airplane before reassignment, and were not rescheduled on another airplane.

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Model F-102A S/N 53-1778
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Flight Test Items	Resolution of Flight Test Program	
4.10 Dynamic Lateral Directional Stability	Transferred to AF 54-1390	
4.11 Static Directional Stability	Transferred to AF 54-1390	
4.12 Adverse Yaw	Transferred to AF 54-1390	
4.13 Rudder Control & Effectiveness	Transferred to AF 54-1390	
4.14 Aileron Characteristics	Transferred to AF 54-1390	
4.15 Stall Characteristics	Transferred to AF 54-1401	
Add #1 Production Airspeed Calibration	EFT-A-219	Position Error Calibration for the Production Nose Boom Airspeed System
Add #3 O.A.T. Probe Calibration	EFT-A-220	Evaluation of the Lewis Flush-Mounted Outside Air Temperature Probe
Add #4 Ram Air Turbine Check-out & Performance	ZC-8-532	Ram Air Turbine Flight Test Performance
Add #2 Special Engine Fuel Control (Evaluated on S/N 54-1380)	EFT-A-8-270A	Linear Thrust Fuel Control Evaluation
Add #5 Drag Chute Deployment Times (Test Accomplished on S/N 54-1791)	ZC-8-512-84	Drag Chute System Deployment Time
Add #6 Reduced Ground Steering Forces	ZC-8-512-130	Tests of Reduced Ground Steering Forces
Add #7 Landing Gear Extension & Loads Tests	EFT-A-8-305A	Landing Gear Extension & Loads Test
	EFT-A-8-307A	Landing Gear Operating Time & Main Landing Gear Fairing Link Loads

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Model F-102A S/N 53-1798
Final Summary of Flight Test Program
ZC-[illegible]-124 Dated 1 September 1955

Flight Test Items	Resolution of Flight Test Program
Antenna Tests	ZN-[illegible]-011 - Antenna Flight Testing on F-102A Aircraft

External Fuel Tank Tests
Z[illegible]-175 Dated 1[illegible] January 1956

Group 3.00 Antenna Radiation 3.11 Antenna Radiation Patterns	Z[illegible]-011 Antenna Flight Testing on F-102A Aircraft

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Model F-102A S/N 53-1799
Final Summary of Flight Test Program
ZC-8-143 Dated 15, March 1956

Flight Test Items	Resolution of Flight Test Program
Part I at Edwards Air Force Base	
1. Calibration Checkout of the Fire Control System and Missiles	ZC-8-512-54, 60, 73 - Airspeed Calibration-Production Nose Boom Assembly ZC-8-512-57 - Snap Up Performance
2. Snap Up Performance Tests	ZC-8-512-57 - Snap Up Performance OR-8-39 - Snap Up Tests on F-102A
Part II Weapons System Demonstration Testing at Holloman Air Force Base	ZC-8-512-69 - Snap Up Performance ZC-8-512-60 - Armament Gear Operation ZC-8-530 - Weapons System Evaluation

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Model YF-102C S/N 53-1806
Final Summary of Flight Test Program
ZC-8-168 Dated 20 January 1956

Flight Test Program Items	Resolution of Flight Test Program	
1.0 Inert Rocket Firing	EFT-A-8-414A	- Armament Firing Summary
	ZC-8-553	- Final F-102 Armament Firing Summary
1.01 Rocket Ejection Tests	EF-8-003	- MB-1 Ejection System Development (Film Report)
	ZC-8-512-58	- Ground Tests of Rocket Ejection System
	ZC-8-512-78	- MB-1 Rocket Ejection System Tests
	ZC-8-512-103	- Ejection System (EJECT-10)
	EFT-A-8-320	- Evaluation of MB-1 Rocket Ejection System
	EFT-A-8-364	- Ejection System (EJECT-10) Qualifying Tests
	EFT-A-8-373A	- Ejection System Development to First Live MB- Rocket Ejection
	EFT-A-8-408B	- Ejection System (EJECT-10) Development Tests
	ZC-8-543-3,4	- F-102 Airplane/MB-1 Rocket Development Progre Report
2.0 Rocket Firing Less XMG-14	EFT-A-8-414A	- Armament Firing Summary
	ZC-8-553	- Final F-102 Armament Firing Summary
	ZC-8-512-91	- MB-1 Live Motor Ejections - Ground Stand
2.01 Separation and Trajectory		
(1) One g Level Flight	ZC-8-512-97	- Live Motor MB-1 Rocket Ejection
	ZC-8-512-110	- MB-1 Rocket Trajectory and Dispersion
	ZC-8-512-116, -120	- MB-1 Rocket Vertical Dispersion
	ZC-8-512-123	- MB-1 Relative Jump Angle
	ZC-8-543-1,2, 3,4	- F-102 Airplane/MB-1 Rocket Development Progress Report
	EF-8-1806-157	- Film Report of MB-1 Ejections, Flights 62, 63, 64, 65, and 66
	EF-8-1806-159	- Film Report of MB-1 Ejections, Flights 56, 57, 59, and 61
	EF-8-1806-187	- Film Report of MB-1 Ejection, Flight 60
	EFT-A-8-448	- Film Report of MB-1 Ejections, Flights 69, 70, 73, and 75
	EFT-A-8-455	- Film Report of MB-1 Ejection Flights 78, 79, and 80

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Model YF-102C S/N 53-1806
(Continued)

Flight Test Program Items		Resolution of Flight Test Program
2.0 Continued	EFT-A-8-456	- Film Report of MB-1 Ejections, Flights 81, 82, and 83
	EFT-A-8-464	- Film Report of MB-1 Ejection, Flight 88
	EFT-A-8-465	- Film Report of MB-1 Ejection, Flight 89
(2) Off g Flight	EFT-A-8-466	- Film Report of MB-1 Ejection, Flight 90
	EFT-A-8-495	- Film Report of MB-1 Ejections, Flights 92, 97, 98, and 99
2.02 Blast Effects	ZC-8-512-97	- Live Motor MB-1 Ejections
2.03 Airplane Response to Rocket Firing	ZC-8-512-97	- Live Motor MB-1 Ejections
	ZC-8-543-5	- F-102 Airplane/MB-1 Rocket Development Progress Report
2.04 Engine Response to Rocket Firing	ZC-8-512-97	- Live Motor MB-1 Ejections
2.05 Exhaust Dust Effects	ZC-8-512-97	- Live Motor MB-1 Ejections
3.0 Rocket Firing With XMG-14	Cancelled	
Addenda to Flight Test Program		
Addendum I		
1. Inert Rocket Ejection	See 1.0	
2. Live Rocket Ejection	See 2.0	
3. Fire Control System Operation	See 3.0 (Cancelled)	
4. Extended Rocket Trajectory Program	See 2.01	
Addendum IA		
1. Inert Rocket Ejection	See 1.0	
2. Live Rocket Ejection	See 2.0	
3. Snap-Up Mode Evaluation	Cancelled	

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Model F-102A S/N 53-1808
Final Summary of Flight Test Program
ZC-8-286 Dated 3 May 1957

Flight Test Program Items	Resolution of Flight Test Program
1.0 Verification of WADC Advanced Instrument Display	EFT-A-8-474 - Flight Test Results of ARDC Advanced Cockpit Instrument Display
1.01 Calibration and Alignment	EFT-A-8-474 - Flight Test Results of ARDC Advanced Cockpit Instrument Display
1.02 Utility and Operation of Instrument Display	EFT-A-8-474 - Flight Test Results of ARDC Advanced Cockpit Instrument Display
2.0 Determine Accuracy of Recorded Intelligence Supplied by Air Data Computer	Cancelled

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Model F-102A S/N 53-1813
Final Summary of Flight Test Program
ZC-8-171 Dated 15 November 1955

Flight Test Items	Resolution of Flight Test Program
1.01 Flutter Investigation	ZU-[illegible]-034 Addendum A Results of Flight Flutter Testing of Large Tail F-102A Aircraft
2.01 Static Directional Stability	ZC-[illegible]-512-45 Static Directional Stability
2.02 Dynamic Lateral-Directional Stability	ZC-[illegible]-512-43 Dynamic Lateral-Directional Stability
2.03 Roll Coupling	ZU-[illegible]-055 Rev. B F-102A Abrupt Aileron Roll Maneuver Study Final Report ZC-[illegible]-512-56 Lateral Control Effectiveness
Addendum #1 Landing Light Evaluation	ZC-[illegible]-512-[illegible]1 Landing Light Evaluation ZC-[illegible]-512-[illegible]2 Landing Light Evaluation
Sundstrand Constant Speed Drive Oil System Recirculating Tests (Negative "g")	EFT-A-3-142A Sundstrand Constant Speed Drive Evaluation

(Also see S/N 54-1300 for Additional Sundstrand Tests)

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Model F-102A S/N 53-1813
Final Summary of Flight Test Program
ZC-8-137 Dated 24 June 1956

Flight Test Items	Resolution of Flight Test Program
Side Stick Control	EFT-A-[illegible]-403A – Quantitative Evaluation of Side Stick Control System

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Model F-102A S/N 56-1317
Final Summary of Flight Test Program
(Case XX Wing Flight Demonstration)
ZC-8-276 Rev. B, Dated 24 July 1957

Flight Test Items			Resolution of Flight Test Program	
Group	1.00	Performance	EFT-A-8-503	Flight Test Performance Evaluation of the F-102A Equipped with the Case XX Wing Design.
	2.00	Angle of Attack Calibration	EFT-A-8-406	Angle of Attack Calibration
	3.00	Stability and Control	ZC-8-552	Stability and Control Demonstration of the F-102A with the Case XX Wing Design.
	4.00	Structural Demonstration	ZC-8-551	Final Report Case XX Wing Structural Demonstration

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Model TF-102A S/N 54-1351
Final Summary of Flight Test Program
ZC-8-140 Dated 1 October 1955

Flight Test Items		Resolution of Flight Test Program	
Phase I Flight and Ground Tests			
Test Item 1.00 Ground Tests			
1.01	Engine Operation	EFT-A-8-231A	Engine Inlet Duct Pressure Recovery Flight Test Results
1.02	Engine Equipment Operation	EFT-A-8-231A	Engine Inlet Duct Pressure Recovery Flight Test Results
1.03	Static Thrust Calibration	EFT-A-8-231A	Engine Inlet Duct Pressure Recovery Flight Test Results
1.04	Low Speed Taxi Tests		Operation Satisfactory
1.05	High Speed Taxi Tests and Lift-Off Characteristics		Operation Satisfactory
Test Item 2.00 Airplane System Functional Tests			
2.01	Emergency Landing Gear Extension	ZC-8-512-61	Emergency Landing Gear Extension Tim
2.01	Aircraft Systems Test	ZC-8-512-62	Functional Test Flight
2.02	Flight Control System	ZC-8-512-58	Turn Coordinator and Damper Operatio
2.02	Damper and Turn Coordinator	ZC-8-512-60	Damper Optimization
2.03	Emergency Fuel Control	ZC-8-512-36	Functional Test Flights
2.04	Power Plant Operation	EFT-A-8-110A	Power Plant Characteristics of the P-23 Engine in the F-102A &TF-102A from Flight Tests
		EFT-A-8-213A	Engine Inlet Duct Pressure Recovery Flight Test Results
2.05	Engine & Structural Cooling	ZC-8-512-36 ZC-8-512-60	Functional Test Flights
2.06	Brake Parachute Operation		System evaluated on S/N 54-1390

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Model TF-102A S/N 1351
(Continued)

Flight Test Items		Resolution of Flight Test Program	
2.07 Control Sensitivity Check		ZC-8-512-36 ZC-8-512-60	Functional Test Flights
Test Item 3.00 Calibrations			
3.01	Airspeed Calibration	ZC-8-512-60	Airspeed Calibration, Instrumented Test Nose Boom Assembly
Test Item 4.00 Airplane Performance			
4.01	Climb	ZC-8-512-62	Performance - Climb to Service Ceiling
4.02	Level Flight	ZC-8-512-67	Performance - Level Flight
Test Item 5.00 Preliminary Flight Characteristics			
5.01	Damper System Check	ZC-8-512-60	Damper Optimization
5.02	Roll Coupling	EFT-A-8-478	Flt. Test Response of Small Tail TF S/N 54-1351 During Abrupt Aileron Rolls
5.03	Buffet Investigation	EFT-A-8-417	Buffet of the TF-102A Airplane
		EFT-A-8-415A	Instrument Panel Vibration
Test Item 6.00 All Weather Equipment			
6.01	Windshield Anti-Icing System	EFT-A-8-420B	Flight Test Evaluation of the Wind-Shield, Rain-Clearing, Anti-Icing & Defogging Systems
6.02	Rain Clearing System	EFT-A-8-420B	Flight Test Evaluation of the Wind-Shield, Rain-Clearing, Anti-Icing & Defogging Systems
6.03	Radome Anti-Icing	EFT-A-8-449	Flight Test Evaluation of the Radome Anti-Icing System
Test Item 7.00 Hydraulics System			
7.01	Hydraulic System Flight Tests	EFT-A-8-424	Flight Test Evaluation of the Hydraulic System

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Model TF-102A S/N 54-1353
Final Summary of Flight Test Program
ZC-8-180 Dated 24 February 1956

Flight Test Items	Resolution of Flight Test Program	
Phase I Flight Tests		
Test Item 1.00 Preliminary Evaluation		
1.01 Qualitative Evalution of Buffet	ZC-8-512-63 EFT-A-8-417	Buffet Intensities Buffet on the TF Airplanes
Test Item 2.00 Quantitative Evaluation		
2.01 Cut Down Canopy	ZC-8-512-63	Buffet Intensities
2.02 Cut Down Canopy plus minor Modifications	EFT-A-8-417	Buffet on the TF Airplanes
Test Item 3.00 Stability and Control		
3.01 Static and Dynamic Directional Stability	ZC-8-512-88	Aileron Step Functions
3.02 Roll Coupling	EFT-A-8-479	Flight Test Response to Abrupt Aileron Rolls

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Model TF-102A S/N 54-1354
Final Summary of Flight Test Program
ZC-8-141 Dated 25 June 1956

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Flight Test Program Items		Resolution of Flight Test Program	
2.0	Airborne Non-Firing Tests		
2.01	Armament Hardware Functional Checkout	ZC-8-512-83 ZC-8-512-79	Armament Hardware Armament Gear Cycling Time
3.0	Firings Without Fire Control System	ZC-8-553	Final Summary of Armament Expenditures from F-102 Aircraft
3.01	Separation and Trajectory		
	(1) Rockets	ZC-8-512-111	2.75-Inch FFAR Separation, Dispersion and Firing Performance Demonstration
		EFT-A-8-399A	2.75-Inch FFAR Separation, Dispersion and Firing Performance Demonstration
		EFT-A-8-459	Film Report of 2.75-Inch FFAR Firings, Flights 35, 36, 44, 45, 47, 48, & 49
		ZC-8-512-140	2.75-Inch FFAR Jump Angle
		EFT-A-8-496	TF-102A 2.75-Inch FFAR Jump Angle Determination
	(2) Missiles	EFT-A-8-482	GAR-1 Falcon Missile Separation, Dispersion and Firing Performance Demonstration
3.02	Aircraft Response to Firing	EFT-A-8-399A EFT-A-8-482	See 3.01.
3.03	Engine Response to Rocket and Missile Exhaust	ZC-8-512-82 EFT-A-8-435A	Engine Response to Missile Exhaust Engine Response to GAR-1 Missile Firings

Addenda to Flight Test Program

Addendum 1

Engine Inlet Scoop Anti-Icing Equipment	EFT-A-8-470	Flight Test Evaluation of the Inlet Duct Anti-Icing System

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Model TF-102A S/N 54-1361
Final Summary of Flight Test Program
ZC-8-257 Dated 23 November 1956

Flight Test Items	Resolution of Flight Test Program
1.00 Structural Demonstration	
1.01 Wing Symmetrical Structural Loads Demonstration	ZC-8-412-121 Flight Operations Report F-102, F-102 Development Program
1.02 Bank to Bank Rolls with Load Factor - Flight Loads Demonstration	
1.03 Level Flight Rolls 360° - Flight Loads Demonstration	
1.04 Rudder Kicks - Flight Loads Demonstration	
1.05 Zero Load Factor Rolls (180°) - Flight Loads Demonstration	
1.06 360° Rolls with Load Factor - Flight Loads Demonstration	

NOTE: All flight test items are covered by Report ZC-8-555, "Final Report Structural Demonstration".

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Model TF-102A S/N 54-1362
Summary of Flight Test Program
ZC-8-259 Rev. B Dated 30 April 1957

	Flight Test Item	Resolution of Flight Test Program	
2.01	Equipment Check	EFT-A-8-480A	Flight Test Results of Stall & Spin Tests
2.02	Preliminary Investigation	EFT-A-8-480A	Flight Test Results of Stall & Spin Tests
2.03	Specification Demonstration of Spins & Stalls	EFT-A-8-480A	Flight Test Results of Stall & Spin Tests

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Model F-102A S/N 56-1138
Final Summary of Flight Test Program
ZC-8-258 Dated 24, May 1957

Flight Test Items		Resolution of Flight Test Program
1.00 ARN-21 (TACAN), ARN-31 Evaluation		
1.01 ARN-21 And Antenna Evaluation	ZN-8-021	Test Flight ARN-21 and ARN-31
1.02 ARN-21 Antenna and Distance Measuring Equipment Evaluation	ZN-8-021	Test Flight ARN-21 and ARN-31
1.03 ARN-31 Evaluation	ZN-8-021	Test Flight ARN-21 and ARN-31

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Model F-102A S/N 54-1380
Final Summary of Flight Test Program
ZC-8-340 Dated 2 July 1956

		Resolution of Flight Test Program
2.00 Aircraft Systems Functional Tests		
2.01 Functional Test Flights	EFT-A-8-362	Emergency Landing Gear System Operation and Evaluation of Main Landing Gear Fairing Link Loads
	EFT-A-8-193	Emergency Landing Gear System Extension Times
2.02 Emergency Fuel Control System	EFT-A-8-340	Emergency Fuel Control System Tests
	EFT-A-8-366	Emergency Control System Operation
2.03 Engine and Structural Cooling	EFT-A-8-476	Flight Test Evaluation of the F-102A Engine Cooling System
3.00 Fuel System		
3.01 Flow Joiners (Hydra Electric, others)	EFT-A-8-392	Flight Test Evaluation of Hydra-Electric Fuel Flow Joiner
3.02 Twin Boost Pump	EFT-A-8-411	Light Weight Boost Pump Tests
3.03 Modified Engine Fuel Control	EFT-A-8-270	Linear Thrust Fuel Control Evaluation
5.00 Afterburner		
5.01 J-57-P-23 Engine Afterburner Flameout	EFT-A-8-472	Flight Test Evaluation of the Pratt and Whitney J57-P-23 Engine Afterburner Flame-out and Relight Characteristics
5.02 A/B Igniter Valve Continuous Fuel Flow		
6.00 Exhaust Nozzle		
6.01 Nozzle Actuating Temperatures	EFT-A-8-445	Flight Test Evaluation of Effect of Insulating Blankets on Afterburner Nozzle Actuator Temperatures
7.00 Sundstrand Unit		
7.01 Negative "g" Check	EFT-A-8-333	Flight Test Evaluation of Sundstrand Negative g Valve Operation
	EFT-A-8-142	Sundstrand Constant Speed Drive Evaluation

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Model F-102A S/N 54-1380
(Continued)

Resolution of Flight Test Program

8.00	General Airframe Improvements		
8.03	Relocated Oil Breather Outlet	ZC-8-512-127	Engine Oil Breather Vent Relocation
9.00	Combustion Starter Changes	EFT-A-8-397	Investigation of Combustion Starter Operation
10.00	Fire Detector		
10.01	Evaluation of Existing System	EFT-A-8-367	Edison Fire and Overheat Detector System Temperatures
11.00	Cabin Pressurization and Ventilating System		
11.02	Carbon Monoxide Testing	EFT-A-8-380	Carbon Monoxide Concentration in Cockpit Enclosure
12.00	Engine and Afterburner Relight Envelope	EFT-A-8-472	Flight Test Evaluation of the Pratt and Whitney J57-P-23 Engine Afterburner Flame-out and Relight Characteristics
13.00	Engine Development Changes	EFT-A-8-472	Flight Test Evaluation of the Pratt and Whitney J57-P-23 Engine Afterburner Flame-out and Relight Characteristics

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Model F-102A S/N 54-1380
(Continued)

		Resolution of Flight Test Program	
1.00	Ground Tests	Qualitative Checkout Satisfactory.	
1.01	Engine Operation		
1.02	Engine Equipment Operation		
1.03	Static Thrust Calibration		
1.04	Low Speed Taxi Tests		
2.00	Aircraft Systems Functional Tests		
2.04	Removal of Engine Diffuser Heat Shield	EFT-A-8-491	Flight Test Survey of Engine Compartment Temperatures with Heat Shielding removed.
3.00			
3.04	Fuel Tank Negative "g" Vent Valve	Test Not Completed.	
4.00	Anti-surge Bleed Valve Optimization	ZC-8-512-69	Anti-surge Bleed Valve Optimization Tests.
		ZC-8-512-77	Anti-surge Bleed Valve Optimization Tests.
		ZC-8-512-90	Intercompressor Bleed Valve Operation with the Convair proposed Governor Setting.
7.00	Sundstrand Unit		
7.02	Development Changes	**Tests Not Completed.**	
7.03	Checkout Revised Engine Oil Breather Valve Setting.		
7.04	Independent Oil System		

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Model F-102A S/N 54-1380
(Continued)

		Resolution of Flight Test Program
8.00	General Airframe Improvements	
8.01	N Bleed Ducts	Test Not Completed.
8.02	Plastic Ducts	Test Not Completed.
8.04	External Fuel Tanks - Development Tests	Test Not Completed.
10.00	Fire Detector	
10.02	Evaluation of Walter Kidde Detector	Test Not Performed.
11.00	Cabin Pressurization and Ventilating System	
11.01	Specification Conformance (dive and climb)	Test Not Completed.
11.03	Hamilton Standard Package	Test Not Completed.
14.00	Fuselage Skin Drain Adequacy	Data Inconclusive.

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Model F-102A S/N 54-1388
Final Summary of Flight Test Program
ZC-8-183 Dated 21 May 1956

Phase III Systems Tests			
Group I			
1.01		ZC-8-548	Engine Inlet Duct Anti-Icing
Group II	Low Pressure Pneumatics		
2.01	Air Conditioning and Pressurization	EFT-A-8-471A	Flight Test Evaluation of the Air Conditioning Pressurization and Adverse Weather Systems
2.02	Hot Day Ground Support Tests	EFT-A-8-356	Hot Day Ground Support Tests
		EFT-A-8-255	Evaluation of Jet Ejector Ground Cooling System for the MG-10 Electronic Equipment
Group III	Anti-Icing & Rain Clearing		
3.01	Windshield & Canopy Defogging & Anti-Icing	EFT-A-8-471A	Flight Test Evaluation of the Air Conditioning Pressurization and Adverse Weather Systems
3.02	Windshield Rain Clearing	EFT-A-8-471A	Flight Test Evaluation of the Air Conditioning Pressurization and Adverse Weather Systems
Group III			
3.03	Radome Anti-Icing	EFT-A-8-427	Flight Test Evaluation of the Radome Anti-Icing System
Group IV			
4.01	Vibration Survey	EFT-A-8-428A	Instrument Panel Vibration
		EFT-A-8-481	Vibrational Loads & Frequencies of the Stratos Refrigeration Unit From Flight Test

ZC-8-183 Revision I, Addendum II
12 April 1957

Group I			
1.00	Hydraulic System Functional Demonstration	ZC-8-546	Flight Test Evaluation of the F-102A & TF-102A Hydraulic Systems

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Model F-102A S/N 54-1390
Final Summary of Flight Test Program
ZC-8-195 Dated 2 July 1956

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	Flight Test Items		Resolution of Flight Test Program
1.01	In Flight Flutter	ZU-8-034	(Addendum A) Results of Flight Flutter Testing of Large Tail F-102A Aircraft
		EFT-A-8-390A	Resonant Rudder Oscillation Study
2.00	Stability and Control	EFT-A-8-387A	Determination of Hinge Moments and Investigation of Pull-out Characteristics
2.01	Directional Stability & Yaw Damper Optimization	ZC-8-512-87	Lateral Directional Stability Characteristics
2.02	Turn Coordinator Optimization	ZC-8-539	Evaluation of the F-102A Automatic Control System
2.03	Adverse Yaw	ZC-8-552	Stability & Control Demonstration of the F-102A with the Case XX Wing
2.04	Rudder Control		
2.05	Aileron Characteristics	EFT-A-8-353	Roll Coupling Flight Test in Direct Manual Mode
4.00	Tactical Improvement Tests	ZC-8-512-102	Drag Chute Riser Service Tests
		ZC-8-512-104	Drag Chute Deployment Times for Airplane With Large Speed Brakes
		EFT-A-8-361A	Landing Gear Loads Tests
		EFT-A-8-398A	Evaluation of Loads in the Modified Landing Gear
3.01	Level Flight Performance	Cancelled - Deemed Unnecessary.	
3.02	Take off and Landing	ZC-8-512-139	Take off and Landing Performance With Skewed Axis MLG
3.03	Speed Brake Effectiveness	Cancelled - Deemed Unnecessary.	
	Supplement I		
	Production Support and Specification Verification for Tactical F-102A Aircraft	EFT-A-8-493	Investigation of Roll Oscillation During Pilot Assist Operation of the Automatic Fire Control System

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Model F-102A S/N 54-1398
Final Summary of Flight Test Program
ZC-8-243 Dated 24 September 1956
*ZC-8-122 Rev. 3 Dated 11 February 1957

Flight Test Items	Resolution of Flight Test Program
1. Bank to Bank Rolls with Load Factor	ZC-8-525-11, 12, 13, 14 Progress Report F-102A Flight Loads Program
2. Level Flight Rolls (360°)	ZC-8-525-10, 11, 12, 13, 14 Progress Report F-102A Flight Loads Program
3. Level Flight Rudder Kicks	ZC-8-525-12, 13, 14 Progress Report F-102A Flight Loads Program
4. Zero Load Factor Rolls (180°)	ZC-8-525-10, 11, 12, 13, 14 Progress Report F-102 Flight Loads Program
5. 360° Rolls with Load Factor	ZC-8-525-10, 11, 12, 13, 14 Progress Report F-102 Flight Loads Program
6. *Asymmetric Loads Structural Demonstration	ZC-8-525-15, 16 Progress Report F-102A Flight Loads Program
7. *Flight Loads Structural Demonstration of Landing Gear	ZC-8-525-15, 16 Progress Report F-102A Flight Loads Program
8. *Structural Demonstration of Ram Air Turbine	ZC-8-525-15, 16 Progress Report F-102A Flight Loads Program

NOTE: All flight test items are covered by Report ZU-8-017, Volumes II, III, and IV.

*This portion of S/N 54-1398 Program transferred from Flight Test Program for F-102A S/N 53-1794, ZC-8-122 Rev. 3 11 February 1957

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Model F-102A S/N 54-1401
Final Summary of Flight Test Program
ZC-8-256 Dated 1 October 1956

Flight Test Items		Resolution of Flight Test Program	
1.01	Special Test Equipment Checkout	EFT-A-8-348A	Ground Deployment Tests of Spin Recovery Chute
		EFT-A-8-403A	In-Flight Evaluation of Spin Recovery Equipment
1.02	Preliminary Investigation	EFT-A-8-423A	Evaluation of Flight Data Obtained From Preliminary Spin Maneuvers
1.03	Specification Demonstration, Spins	EFT-A-8-451	Spin Tests of an F-102A 16 MM Movie Film
		EFT-A-8-430A	Flight Test Results of Stall and Spin Tests
1.04	Specification Demonstration Stalls	EFT-A-8-430A	Flight Test Results of Stall and Spin Tests
2.00	Hydraulic Systems Test	Transferred to 54-1388 per EFT-P-8-2086	
Addendum I			
1.00	Snap-Up Development	EFT-A-8-504	Flight Test Evaluation of the Snap-Up Attack
2.00	Electrical Loads	Deleted	

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Model TF-102A S/N 55-4051
Final Summary of Flight Test Program
(External Fuel Tank Tests)
ZC-8-173 Dated 19 January 1956

Flight Test Items		Resolution of Flight Test Program	
Group 5.00	TF Demonstration		
5.01	Stability Tests	EFT-A-8-484	Qualitative Evaluation of the TF-102A with external fuel tanks.
5.02	Flight Performance Tests	EFT-A-8-484	

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F-102 ACCELERATED DEVELOPMENT
TEST PROGRAM
STATUS
OF
FLIGHT TEST ACTIVITIES

FEBRUARY 1956

CLASSIFIC
UNCL
AUTHORIZED BY: DD
CONTRACT NO. F
RECLASSIFIED BY:
DEPT. 11

CLASSIFICATION CHANGED TO:
UNCLASSIFIED
AUTHORIZED BY: DD 254 DATE 12/12/66
CONTRACT NO F106/36546/30169/
RECLASSIFIED BY RCB
DEPT. 1162 DATE 6/24/66
L. G. AVRITT, SECURITY OFFICER
CONVAIR DIVISION OF GENERAL DYNAMICS

CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

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SAN DIEGO

ZC-8-142-6

1 March 1956

F-102

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F-102 ACCELERATED DEVELOPMENT TEST PROGRAM STATUS OF FLIGHT TEST ACTIVITIES FEBRUARY 1956

CLASSIFICATION CHANGED TO:
UNCLASSIFIED
AUTHORIZED BY: DD 254 DATE 12/12/66
CONTRACT NO. F106/3654/30167/etc
RECLASSIFIED BY:
DEPT. 1107-2 DATE 6/24/08
~~L. G. AVRITT,~~ SECURITY OFFICER
CONVAIR DIVISION OF GENERAL DYNAMICS

PREPARED BY Lloyd Child
L. Child

GROUP Engineering Flight Test

REFERENCE

CHECKED BY T. L. Maloy
T. L. Maloy

APPROVED BY P. M. Prophett
P. M. Prophett
Chief of Engineering
Flight Test

NO.				PAGES AFFECTED

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S/N 53-1787 (Stability and Control, Phase I, II, and III) 13 and 14

S/N 53-1788 (Armament, 8-90 Model) 15 and 16

S/N 53-1791 (Phase I, II, III) 17 and 18

S/N 53-1797 (Rocket Type MB-1 Program) 19 and 20

S/N 53-1798 (Antenna and Control Tests) 21 and 22

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** First portion of Flight Test Program (Completed).

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INTRODUCTION

This summary report is to maintain the current presentation of the general overall progress of CONVAIR Model F-102 aircraft in the Accelerated Development Test Program. Commitments on Contract AF AF 33(600)-5942 are compared with actual accomplishments on these airplanes currently being tested by Convair.

SUMMARY OF FLIGHT TEST ACTIVITY

During the 29 months since the initial flight of an F-102 aircraft, Convair test airplanes have flown a total of 655 flight hours during 945 flights in 178 maintenance months of the program. During these flights favorable results have been achieved, particularly on the following test items:

Flutter program completed to 655 knots and Mach Number 1.5
Buzz problem eliminated by extended air intake ramp.
Armament firing up to maximum altitude and Mach Number 1.2.
Pilot assist system tested.
Original configuration evaluated.
Extended nose and area rule waisted fuselage evaluated.
Large vertical tail evaluated.

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SUMMARY OF FLIGHT TEST ACTIVITIES FOR FEBRUARY 1956

Ten Convair test airplanes flew 49.9 hours in 65 flights during February 1956. Of this total, 44 hours in 54 flights were flown toward the accomplishment of the Convair Development Test Program.

Flight test items of special interest:

a. Airplane S/N 53-1791: Air Force Phase IV performance flight tests are underway.

b. Airplane S/N 53-1799: Miss evaluation testing of the fire control system has started. On the two passes which have been completed to date, HAC reports miss evaluation of 1.5° calculated root mean squared angle off course at end of missile boost which is well within the 2.8° permissible for 12 consecutive passes.

c. Airplane S/N 53-1813: The roll coupling program with the gloved vertical tail was completed. With 5° aileron displacement the airplane is unrestricted in roll and has satisfactory roll rates.

d. Airplane S/N 54-1351: Alternating vortex generators 1-1/2 inches high over the canopy have improved the buffet condition to a point where it is now marginally acceptable.

At Edwards Air Force Base flight testing was conducted on lateral acceleration limit warning device, pilot assist, pitch "g" limiter, pitch damper, turn coordinator, yaw damper, beep trim check, trim servo check, up elevator testing, subsonic and transonic buffet, buzz investigation, airspeed, outside air temperature and air data computer calibration, miss evaluation, snap-up performance, fire detector and overheat systems checks, flutter and roll investigation and Air Force Phase IV evaluation.

At Holloman Air Force Base flight tests were accomplished on yaw damper checkout, pylon airflow with and without the MB-1 rocket installed, rocket environmental tests, angle of attack measurement, negative pitch rate determination, squib firing and missile launch tests.

The current aircraft in the flight test program are the following:

Serial Number	Location	Assignment
53-1781	Holloman Air Force Base	Rocket Type MB-1 Armament
53-1787	Edwards Air Force Base	Stab. and Control, PhaseI,II,& III
53-1788	Holloman Air Force Base	Armament, 8-90 Model
53-1791	Edwards Air Force Base	Phase I, II, and III
53-1793	San Diego Exp. Factory	Armament, 8-10 Model(under repair)
53-1794	San Diego Exp. Factory	Flight loads(under modification)
53-1797	Holloman Air Force Base	Rocket Type MB-1 Armament
53-1798	Edwards Air Force Base	Antenna tests, flight controls
53-1799	Edwards Air Force Base	Weapons Syst. Demonstration,PhaseIII
53-1806	Edwards Air Force Base	Rocket Type MB-1 Armament, Holleman
53-1813	Edwards Air Force Base	Large tail
54-1351	Edwards Air Force Base	Phase I & II (Trainer).

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SUMMARY OF FLIGHT TEST ACTIVITIES FOR FEBRUARY 1956 - Continued

The summary of Convair flight operations during February is as follows:

Serial Number	Total Hours	Prod. Hours	Number of Flights	Days Flown	Major Ground Work Items
53-1781	4.0	4.0	5	5	Tethered Fire Instrumentation
53-1787	4.5	4.5	5	5	Pilot Assist, Auto Flt. Cont.
53-1788	6.5	6.5	8	6	—
53-1791	9.1	8.7	12	10	—
53-1794	1.3	0.8	2	2	Instrumentation, modification
53-1797	2.8	—	4	4	Shakedown, instrumentation
53-1798	6.4	5.9	7	6	Pilot assist
53-1799	4.4	4.4	5	4	Hughes Aircraft Fire Control
53-1806*	1.2	—	2	2	Shakedown
53-1813	5.6	5.1	8	7	—
54-1351	4.1	4.1	7	6	—
Totals	49.9	44.0	65	57	

* Not yet on Engineering Flight Test status.

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FLIGHT TEST HOURS FOR CONVAIR ASSIGNED AIRCRAFT (TOTAL ACCUMULATED)

Airplane S/N	Total	Convair Test	Other Agency	Ferry, Shakedown, Chase, Familiarization, etc.
* 52-7994	1.8	1.8	0.0	0.0
** 52-7995	105.0	67.2	36.1	1.7
** 52-1799	12.6	9.9	0.8	1.9
** 53-1780	52.6	46.0	5.3	1.3
53-1781	62.6	53.8	—	8.8
53-1782	21.1	18.4	0.3	2.4
** 53-1783	84.0	23.5	59.8	0.7
53-1787	105.4	66.4	35.3	3.7
53-1788	74.7	67.4	0.9	6.4
53-1791	32.4	20.7	10.5	1.2
*** 53-1793	11.4	9.0	0.0	2.4
**** 53-1794	6.2	4.6	0.0	1.3
53-1797	4.0	—	—	4.0
53-1798	33.9	26.4	—	7.0
53-1799	9.0	8.1	—	0.9
53-1806	1.2	—	—	1.2
53-1813	19.7	16.4	2.5	0.8
54-1351	16.9	15.1	1.3	0.5
Totals	654.5	454.7	152.8	46.2

NUMBER OF FLIGHTS FOR CONVAIR ASSIGNED AIRCRAFT (TOTAL ACCUMULATED)

52-7994	6	6	0	0
52-7995	163	113	46	4
53-1779	24	19	1	4
53-1780	76	67	6	3
53-1781	103	89	0	14
53-1782	29	24	1	4
53-1783	94	35	57	2
53-1787	148	102	41	5
53-1788	112	102	1	9
53-1791	58	41	15	2
53-1793	16	13	0	3
53-1794	10	7	0	3
53-1797	6	0	0	6
53-1798	34	25	0	9
53-1799	11	10	0	1
53-1806	2	—	—	2
53-1813	26	23	3	0
54-1351	27	24	2	1
Totals	945	700	173	72

* Crashed (Tests added to 7995)
** Indicates Convair [redacted]
Operations complete
*** Being repaired at San Diego
**** Being modified at San Diego

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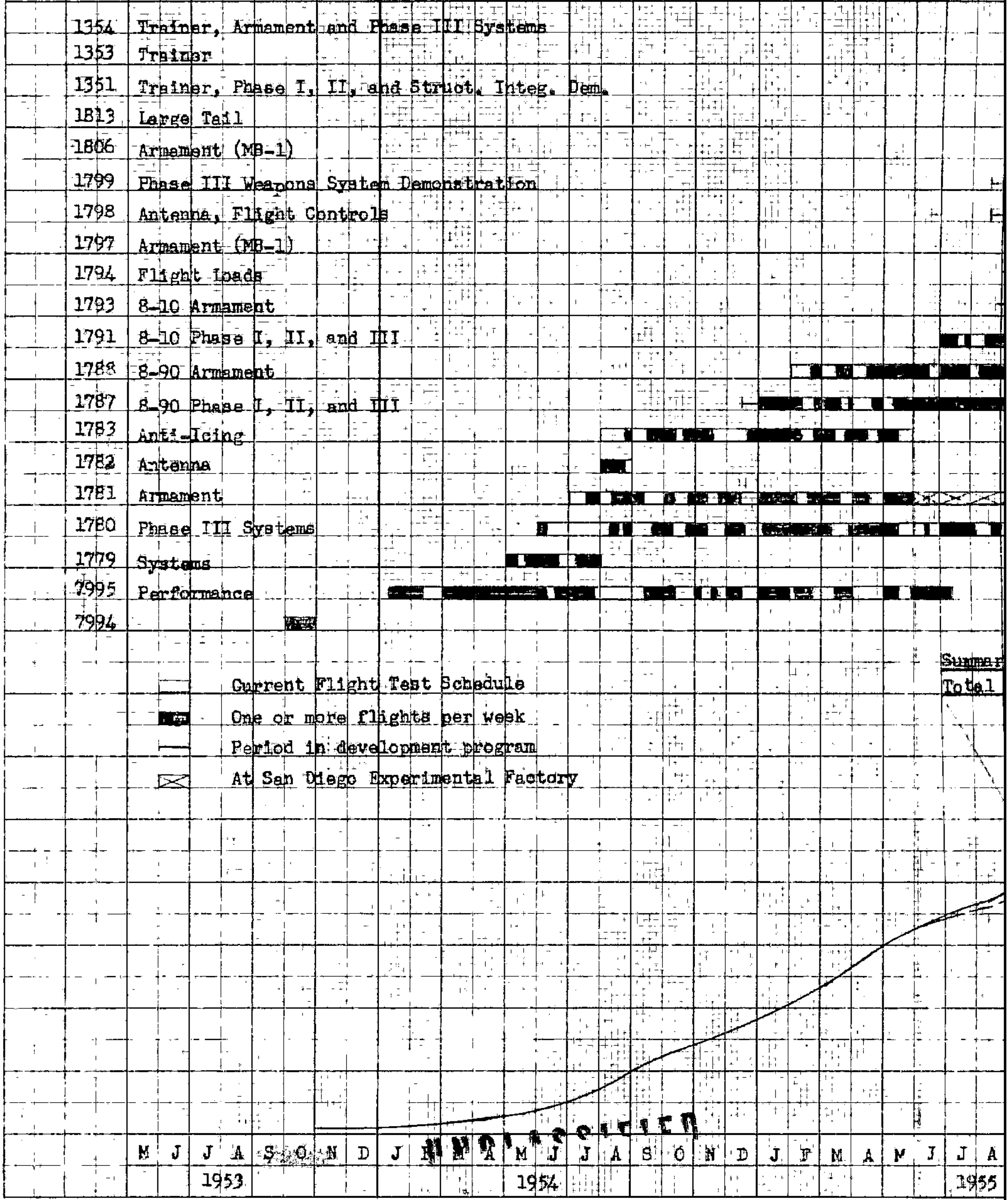
CONVAIR F-102 AND TF-102 ACCELERATED DEVELOPMENT PROGRAM
AIRCRAFT ASSIGNMENT SCHEDULE AND ACTIVITY STATUS



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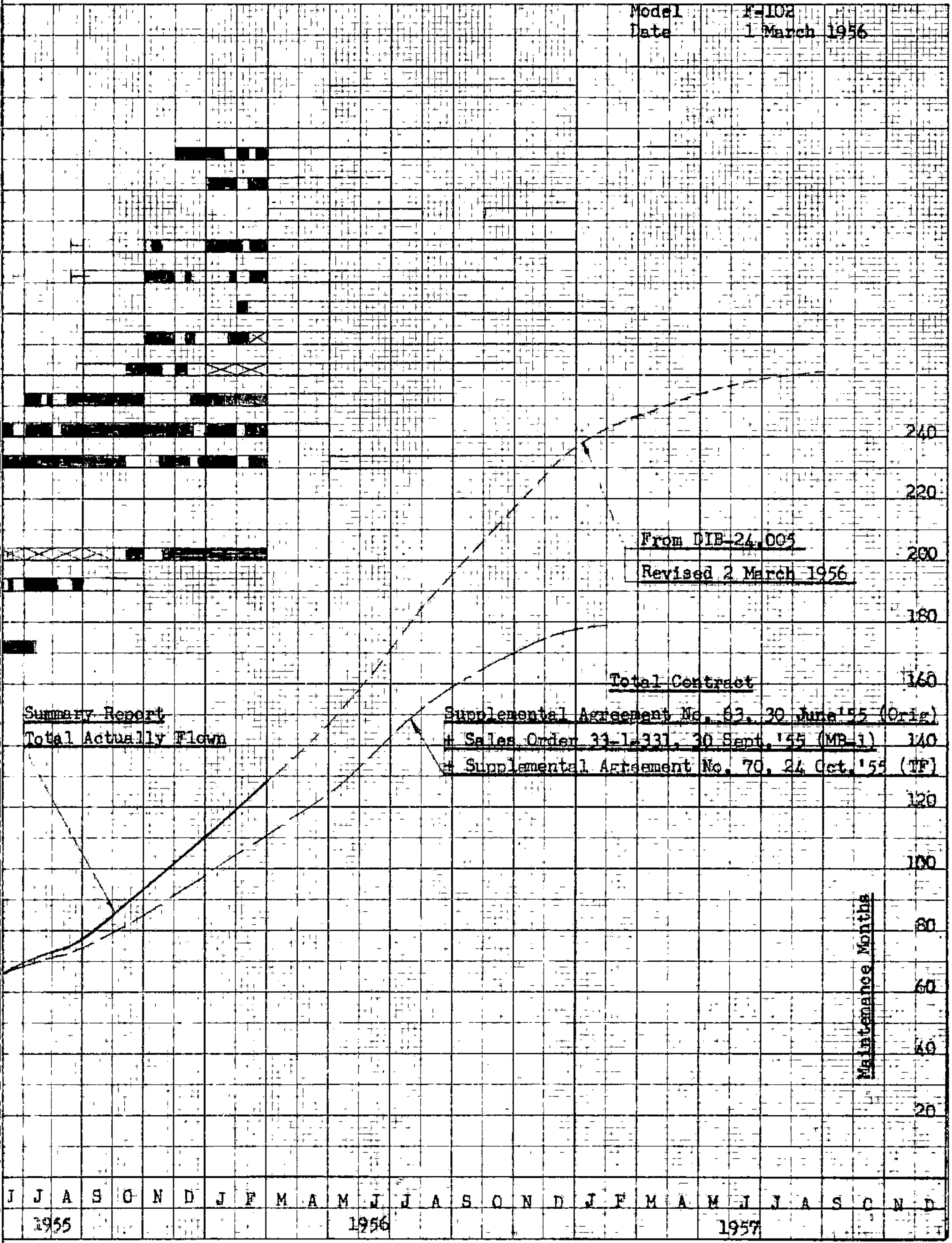
From DIB-24,005
Revised 2 March 1956
Total Contract
Supplemental Agreement No. 63, 30 June '55 (Orig)
+ Sales Order 33-1-331, 30 Sept. '55 (MB-1)
+ Supplemental Agreement No. 70, 24 Oct. '55 (TF)
Summary Report
Total Actually Flown
Maintenance Months
240
220
200
180
160
140
120
100
80
60
40
20
J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D
1955
1956
1957

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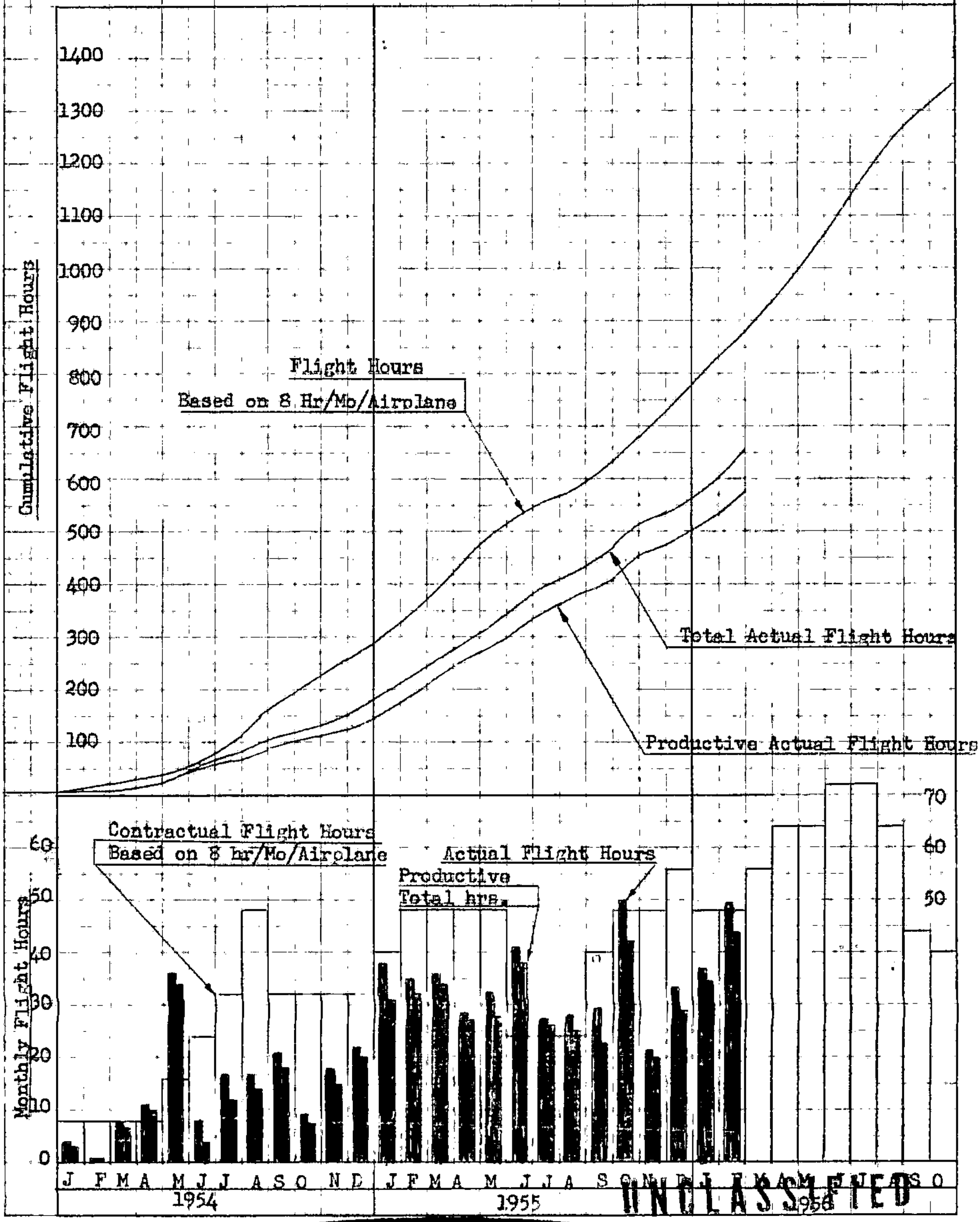
CONVAIR F-102 AND TF-102 ACCELERATED DEVELOPMENT PROGRAM-ESTIMATED FLIGHT TIME AND FLIGHT HOURS FLOWN



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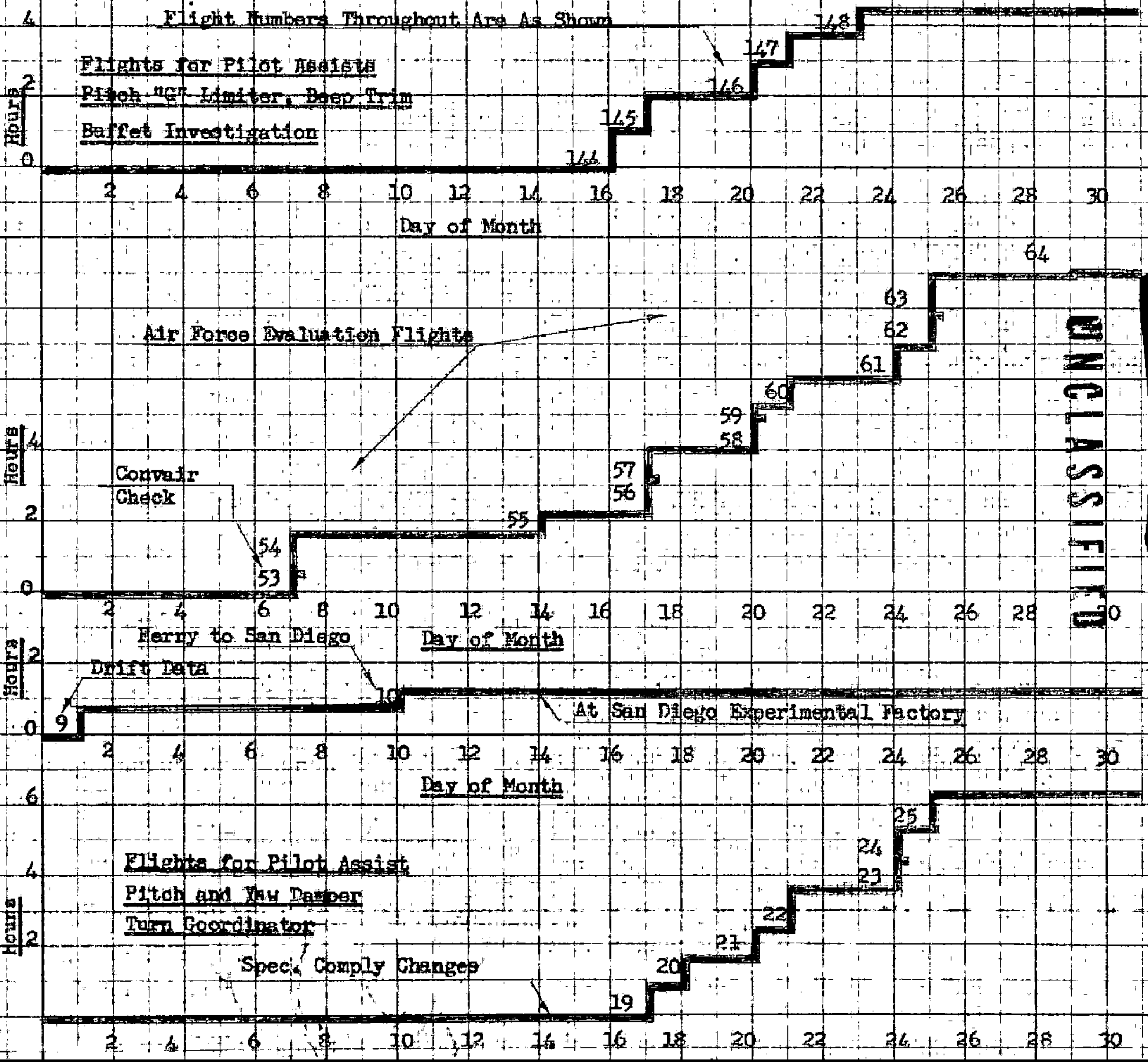
ENGINEERING FLIGHT TEST ACTIVITIES FOR FEBRUARY 1956

CONVAIR AIRPLANES AT EDWARDS AIR FORCE BASE

Flight Numbers Throughout Are As Shown

Airplane S/N	Flight Hours
53-1787	4.5
53-1791	9.1
53-1794	1.3
53-1798	6.4



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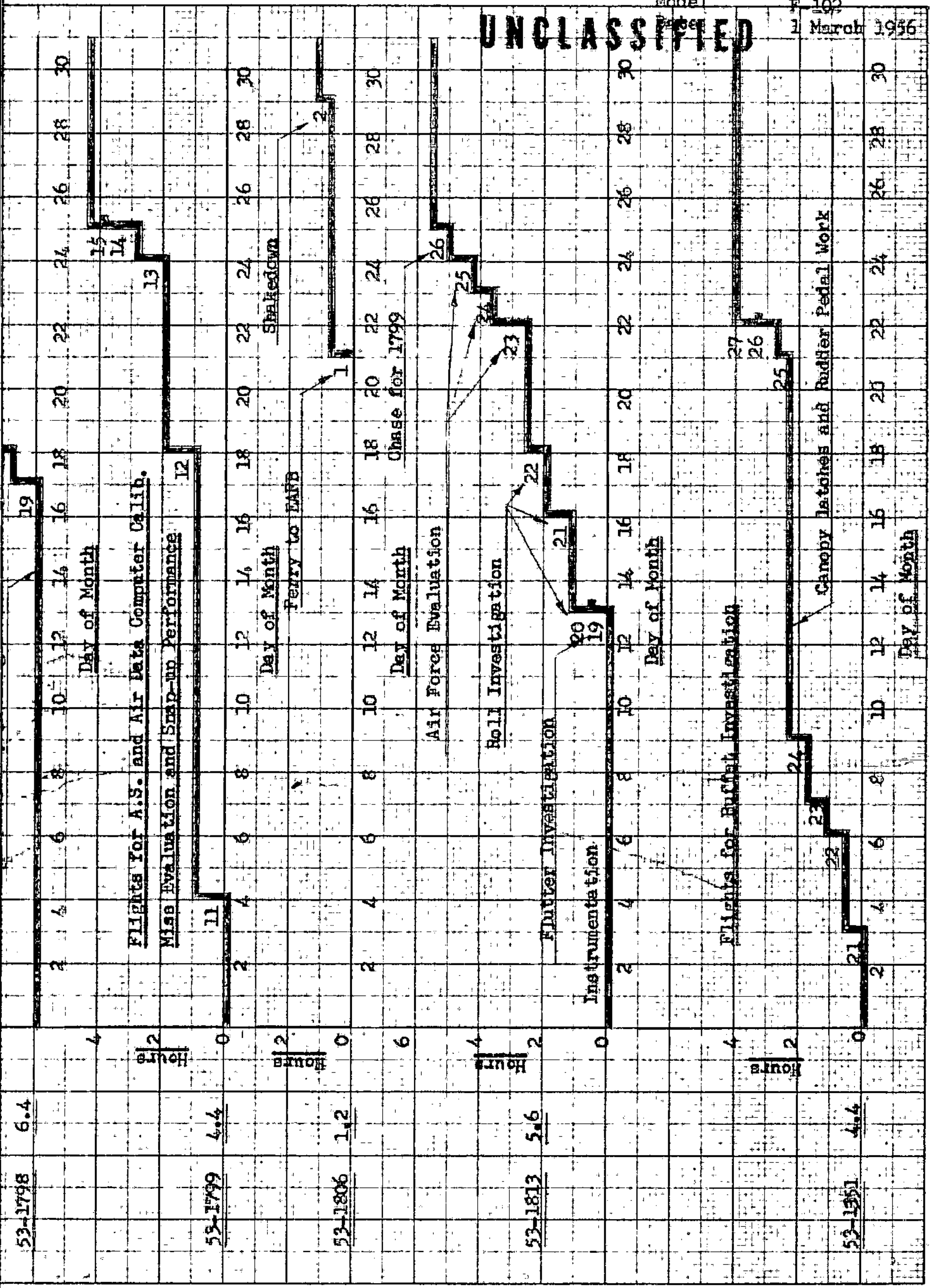

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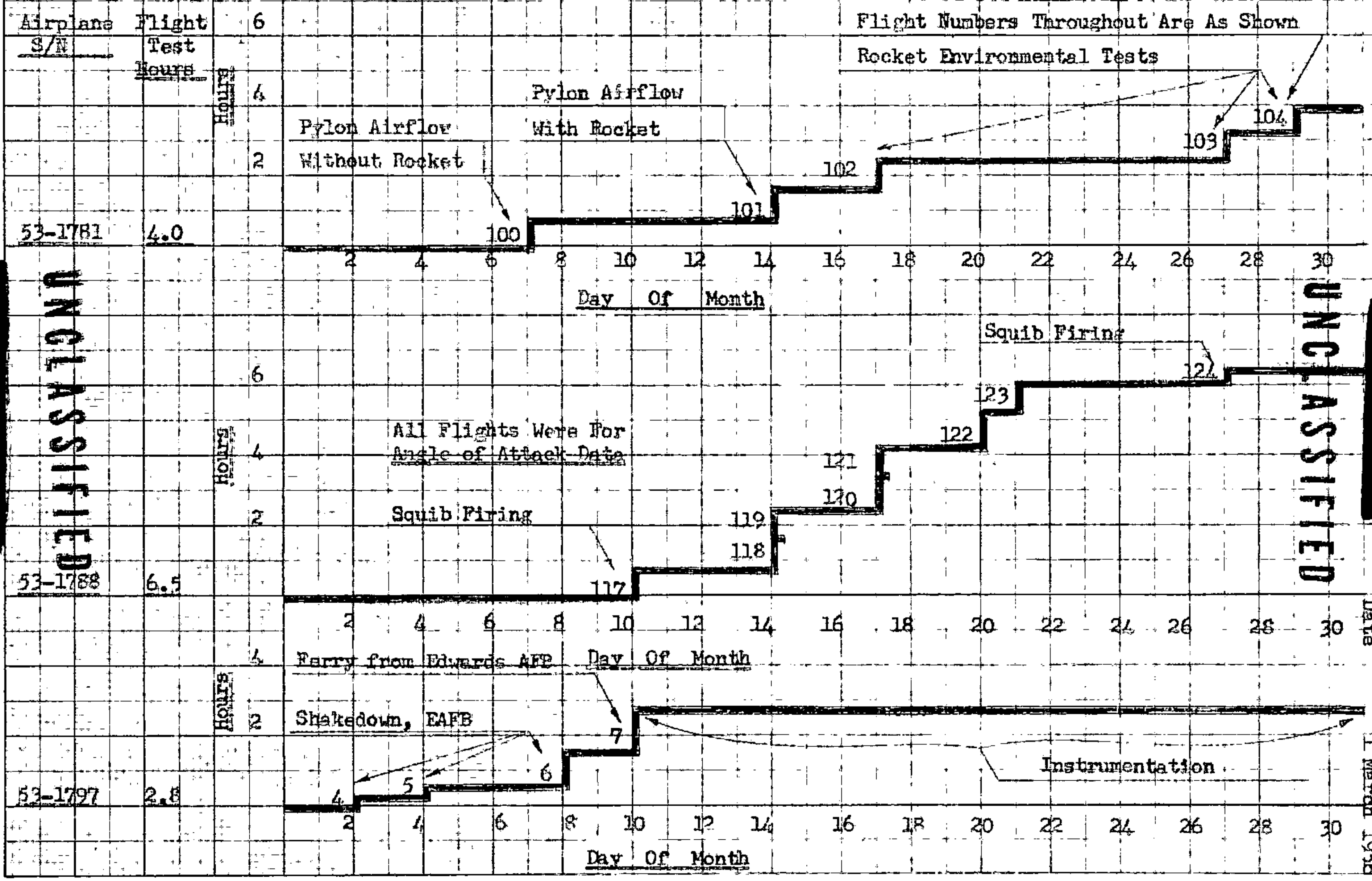
ENGINEERING FLIGHT TEST ACTIVITIES FOR FEBRUARY 1956
CONVAIR AIRPLANES AT HOLLOMAN AIR FORCE BASE



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1781

ROCKET TYPE MB-1 ARMAMENT TESTS

Assignment - (Reinstated Program)

This flight test program is authorized by Sales Order 33-1-331 dated 30 September 1955 to Contract AF 33(600)-5942. Convair will conduct the tests at Holloman Air Development Center, New Mexico, during a nine-month period beginning 1 October 1955. Douglas Aircraft Company will supply the missiles, their storage, and handling. This assignment has been divided into three general flight test stages as shown on the following page.

Summary of Activity - (Reinstated Program)

Airplane S/N 53-1781 was modified at the Convair San Diego Experimental Factory to a Rocket Type MB-1 launching configuration and reinstated for experimental flight tests.

At Holloman AFB the airplane was tested for stability characteristics with and without the rocket. Ground work was carried out to install instrumentation for tethered firing tests.

During February instrumentation for tethered firing continued and five flights were performed for pylon airflow with and without the rocket, yaw damper checkout and rocket environmental check.

Future Activity

In March the airplane will be in work status for tethered ground firing of rockets.

Launchings of inert rockets and airborne live rocket firing will be conducted, after completion of ground testing.

This airplane will be assigned to the Convair Palmdale Facility for pilot proficiency, target and chase plane missions per Sales Order No. 63-1-125, after completion of flight test program.

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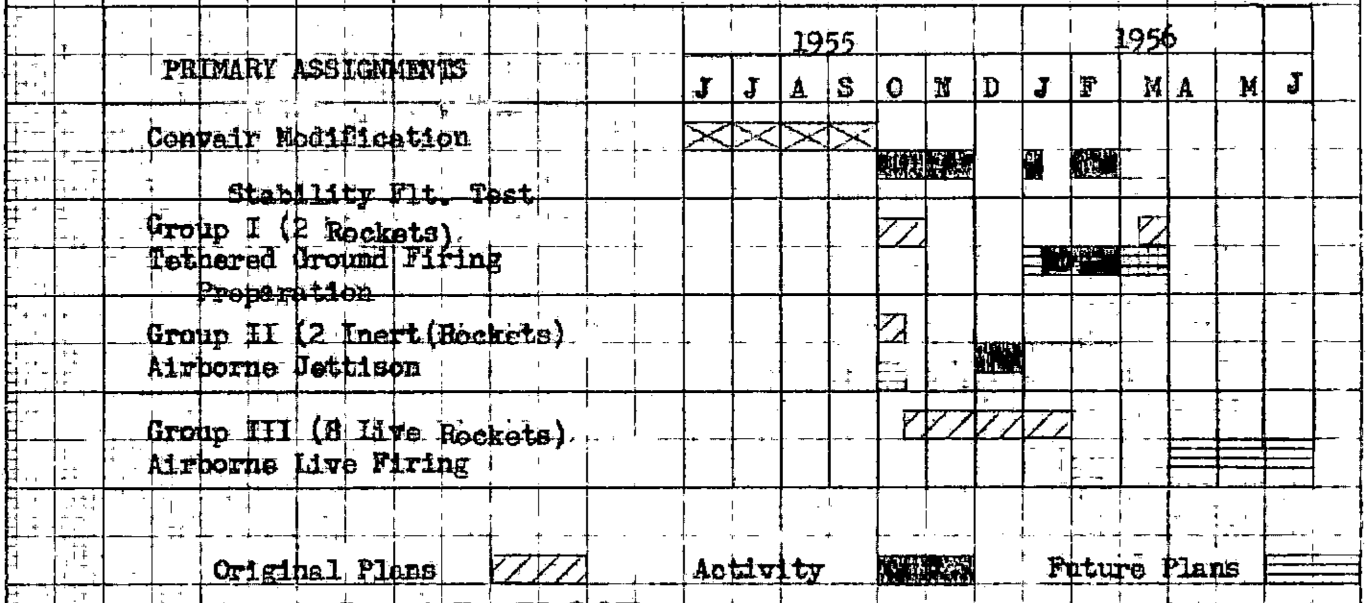
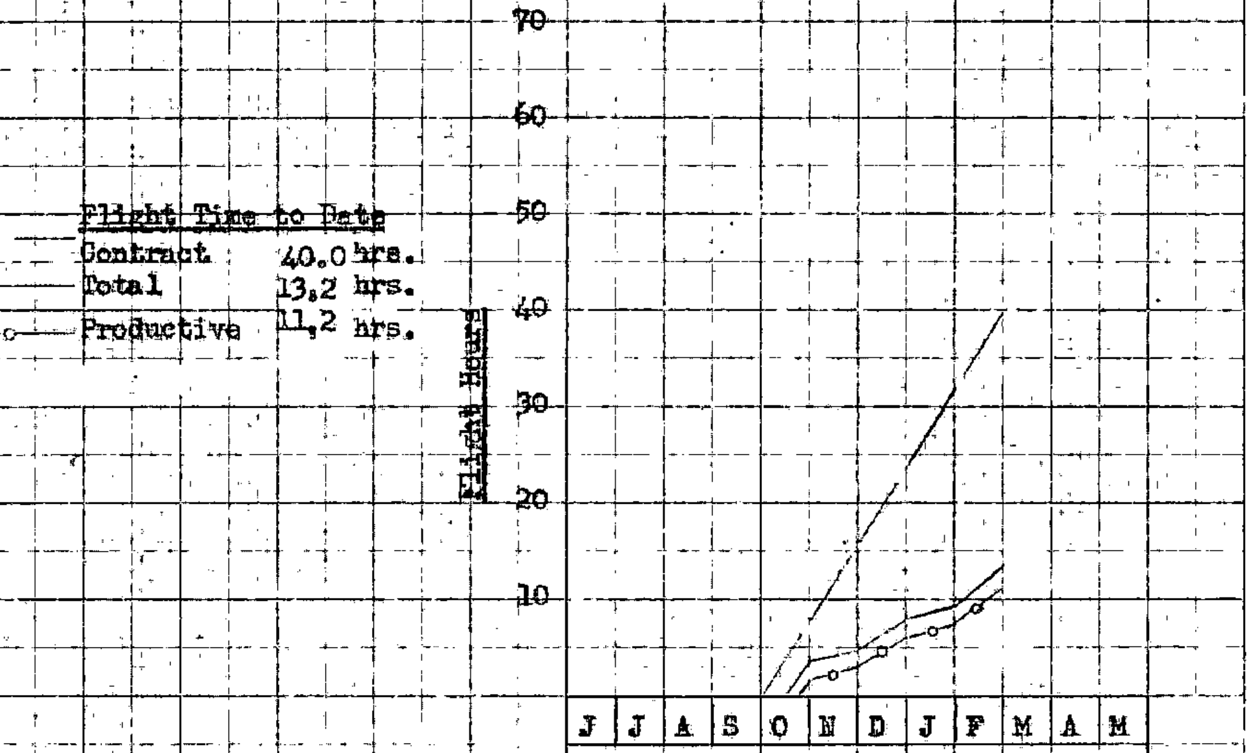
S/N 53-1781

(Reinstated Rocket Type MB-1 Program)



Convair Report No. ZC-8-131



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1787

Assignment

This airplane is assigned to Convair for a period of 12 months which commenced in January 1955. The formal statement of flight test assignment for this airplane as presented in Supplemental Agreement No. 63 dated 30 June 1955 is as follows: "Prototype tests of reduced weight aircraft, Phase I, II, and III." Convair has divided this assignment into five general flight test stages to accomplish the required flight test operations effectively. Graph on following page presents five stages as programmed and performed.

Summary of Activity

Phase I performance and preliminary flight characteristics have been accomplished preceded by qualitative evaluation of airplane systems and airspeed calibrations. Phase II testing has been completed. Roll rate data has been obtained to substantiate analog computer studies for the purpose of establishing roll rate limitations.

This airplane has confirmed the performance predictions for the 8-10 airplanes. The manual mode surface control system has been developed to a satisfactory degree and the trim servo provides a stable stick force gradient in the sonic, transonic, and supersonic speed ranges tested. Some additional tests may be required to obtain a more desirable stick force gradient. Much testing on this airplane was for investigation of high speed vibration or "buzz".

During February flight testing was performed for lateral acceleration warning light, pitch "g" limiter tie-in to pilot assist, beep trim development, pitch attitude and rate gain optimization, up elevator function, supersonic buffet study, and alternate airspeed source.

Future Activity

In March tests will be performed on pilot assist, lagged attitude-plus-rate method of obtaining pitch rate, alternate airspeed systems and buffet investigations. Tail pipe ejectors geometrically similar to those proposed for the F-102B will be tested on S/N 53-1787 commencing the week of 13 March and lasting 8 weeks as authorized by Contract AF 33(600)-30169.

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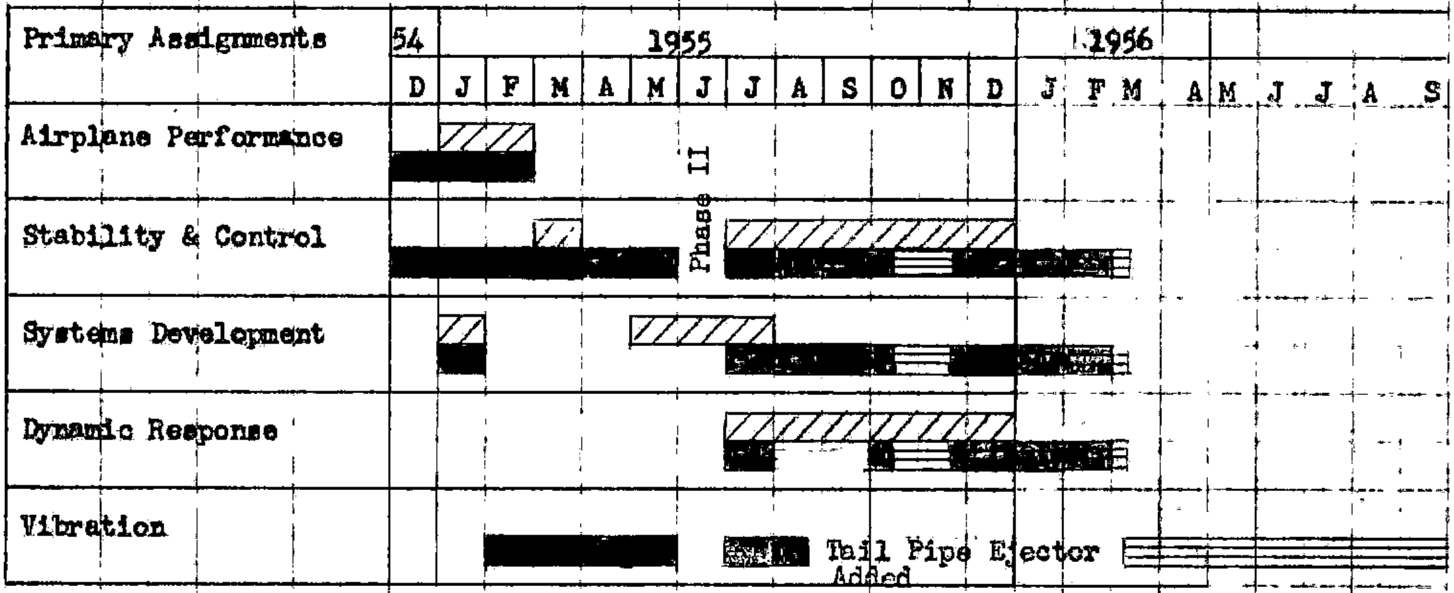
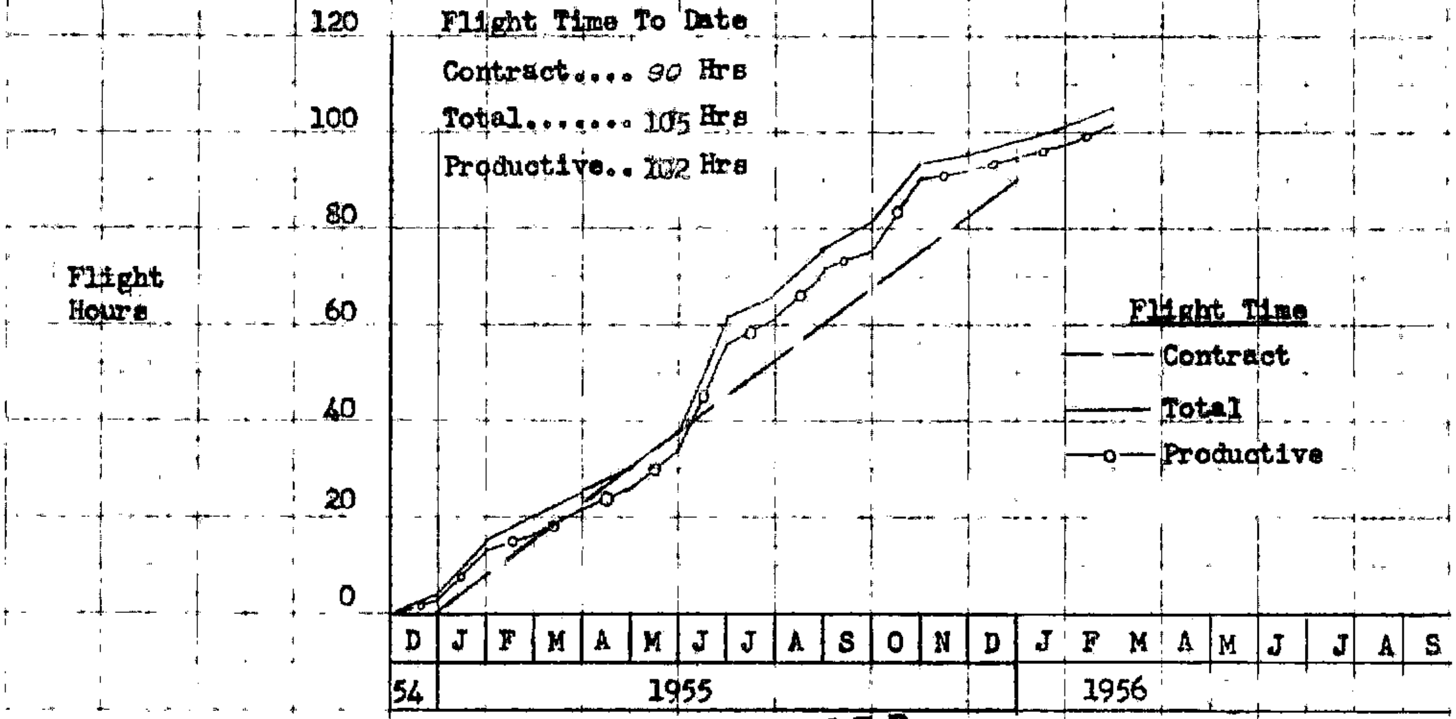
Summary of Activity
S/N 53-1787



Original Planning ▨ Activity ■ Future Planning ☰
Convair Report No. ZC-8-118



Flight Time To Date
Contract.... 90 Hrs
Total....... 105 Hrs
Productive.. 102 Hrs

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1788

Assignment

This airplane was assigned to Convair for a period of 5 months commencing in February 1955. Its formal statement of flight test assignment as presented in Supplemental Agreement No. 63, dated 30 June 1955, of Contract AF 33(600)-5942 is as follows: "To be utilized for subsonic verification of armament tests and supersonic tests and functional development of armaments including hardware and dispersion studies. These tests will be conducted up to combat altitude and maximum speed. Utilize for system test of MX-1179 after completion of armament tests above." Conviar has divided this assignment into six general test stages to accomplish the required flight test operations effectively. The graph on the following page presents six stages as programmed and performed.

Summary of Activity

Airplane S/N 53-1788 commenced its flight test program at Holloman AFB on 14 March 1955.

Convair missile and rocket firing programs are now completed. Missile and rocket separation characteristics were satisfactory. No effect on engine performance was encountered.

During February eight flights were performed for squib firing, negative pitch rate determination and angle of attack data using the convectron inclinometer.

Future Activity

Flight tests in March and April will be for raising armament door restrictions, angle of attack calibration, 2-inch rocket jump angle and Hughes Aircraft Falcon trajectory tests.

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This airplane will be assigned to the Convair Holloman Facility for chase plane missions per Sales Order No. 83-1-125, after completion of flight test program.

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Summary of Activity
S/N 53-1788

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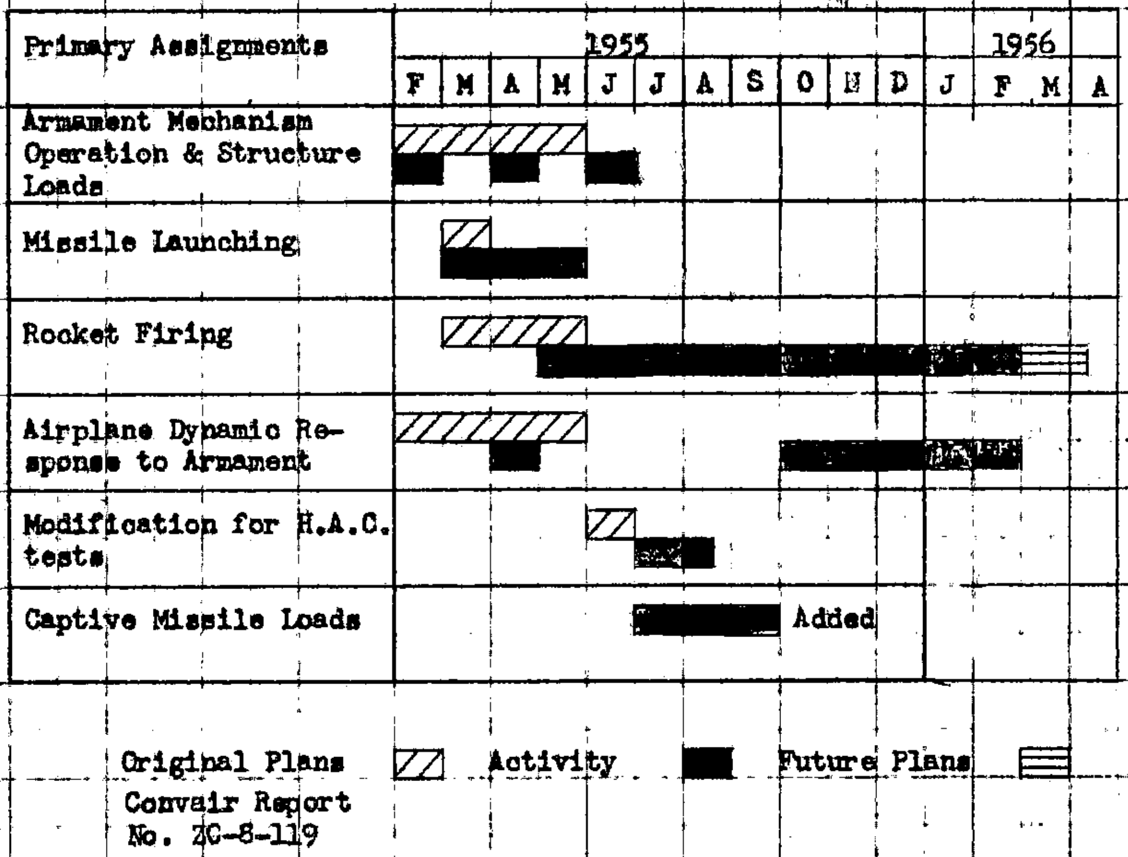

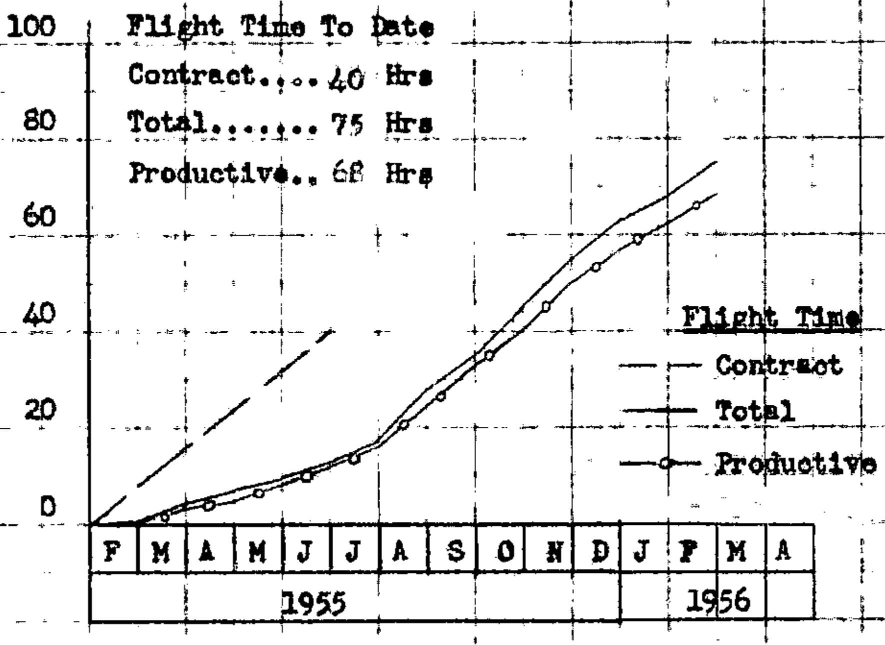

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1791

Assignment

This airplane is assigned to Convair for a period of 15 months commencing in June 1955. Its formal statement of flight test assignment as presented in Supplemental Agreement No. 63, dated 30 June 1955, of Contract AF 33(600)-5942, is: "To be utilized for recheck of Phase I, II, and III plus cooling as it applies to first production light weight model." Convair has divided this assignment into seven general flight test stages to accomplish this flight test operation effectively. Graph, Page 18, shows these stages as programmed and performed.

Summary of Activity

Airplane S/N 53-1791 arrived at Edwards AFB on 21 June 1955. During the flutter tests to remove speed restrictions the airplane demonstrated an unsatisfactory buzz condition above Mach. No. 1.20. From July 1955 through January 1956 the primary flight testing consisted of buzz investigation. The buzz problem was corrected by installation of extended ramps on the lips of the engine air intake ducts enabling buzz-free flight to be accomplished at speeds up to the specification 1.5 Mach number. Flutter restrictions were removed simultaneously with the successful buzz correction flights.

During the flights, numerous items of secondary importance were tested - such as nose wheel door airloads and deflection, dive brake effectiveness, Shelley unit compartment static pressure, pilot assist and turn coordinator checkout, air-oil cooler airflow optimization, cabin air conditioning, MLG flexibility and side brace loads, and time lag constant in static pressure system. The airplane was laid up for one month for Specification Compliance Changes.

Throughout February 1956 the Air Force carried out Phase IV performance testing.

Future Activity

After Phase IV, tests will be accomplished on external fuel tanks, MLG operating time, sideslip feedback, spin and stall tests, canopy defog, MLG position warning, carbon monoxide, dive brake temperature and effectiveness, rudder effectiveness with production dive brakes and elevator stick force per "g".

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Summary of Activity
S/N 53-1791

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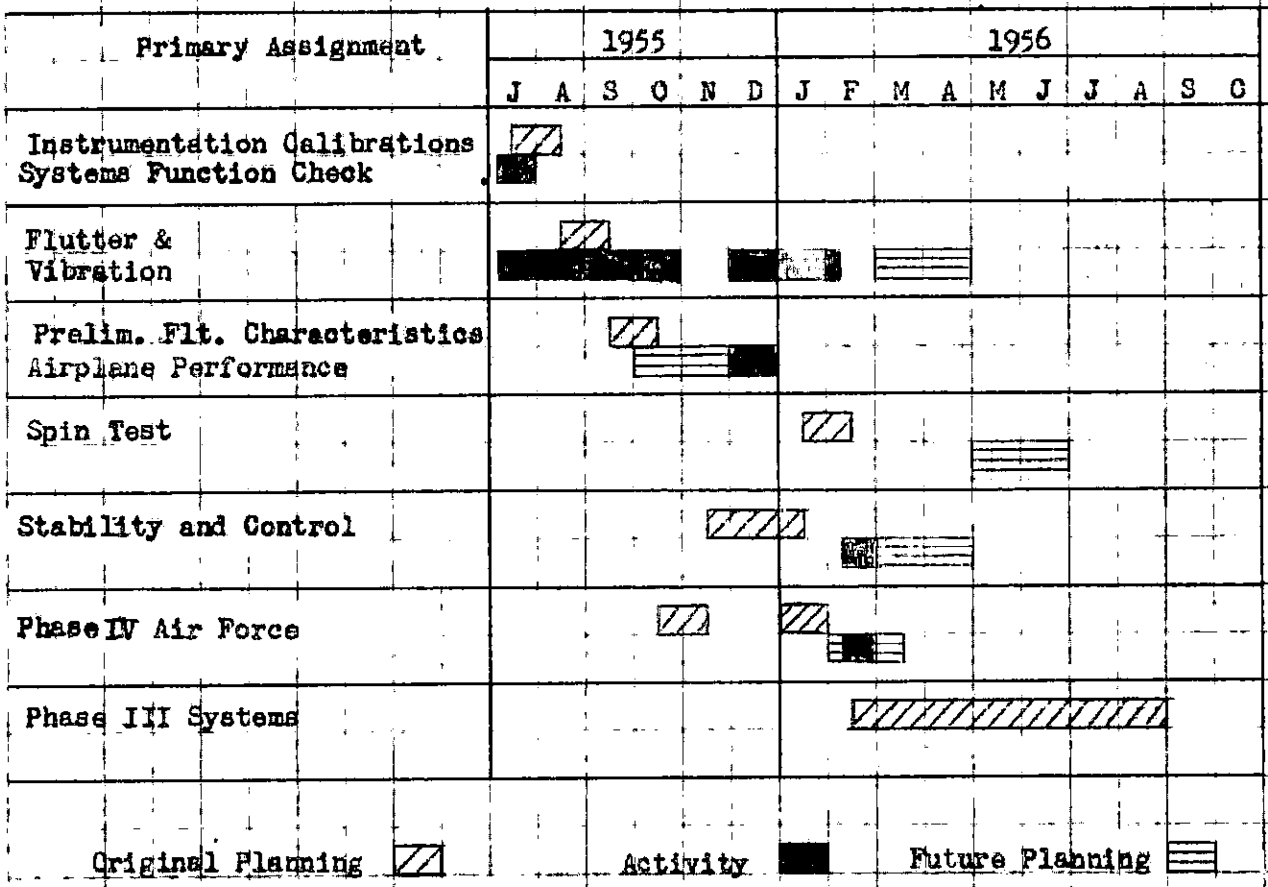
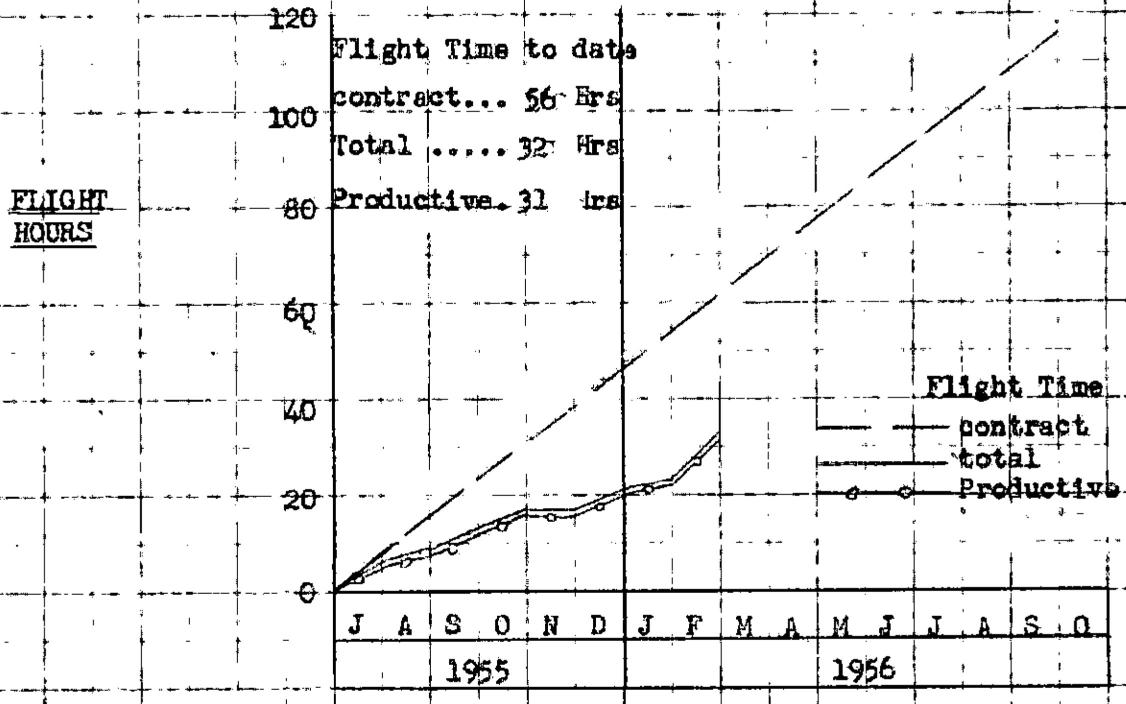


Convair Report
No. ZC-8-120



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES

AIRPLANE S/N 53-1797

Assignment

This airplane is assigned to Convair for a period of 7 months. The flight test program is authorized by Supplemental Agreement No. 63 dated 30 June 1955 and Sales Order 33-1-331 dated 30 December 1955 to Contract AF 33(600)-5942 to demonstrate the Rocket Type MB-1 extension armament system for the F-102 airplane. Convair will conduct the tests at Holloman AFB during the period from January through November 1956. Douglas Aircraft Company will supply all storage and ground handling requirements and will be responsible for rocket development. Hughes Aircraft Company will supply all facilities, maintenance and support required for the XMG-14 fire control system. The graph on the following page shows the program.

The test objective is to evaluate the extension method of launching the Rocket Type MB-1 and to obtain data for development of the airplane fire control system and the rocket. Secondary objectives are airplane response, rocket performance, and the effect of rocket blast and exhaust upon airplane structure and power plant.

The test objectives will be accomplished by firing 36 rockets under varying parameters. Of these, 7 rockets will contain telemetering units.

Summary of Activity

After shakedown flights at Edwards AFB, this airplane was ferried to Holloman AFB on 10 February 1956. Instrumentation and installation work was carried on throughout the remainder of the month.

Future Activity

Flights will be for instrument calibration, armament hardware check, raising of door restriction, rocket fly-around, jettison check, rocket firing, XMG-14 checkout and system evaluation.

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Page 20
Report No. Z2-8-112-6
Model F-102
Date 1 March 195

SUMMARY OF ACTIVITY

Airplane S/N 53-17

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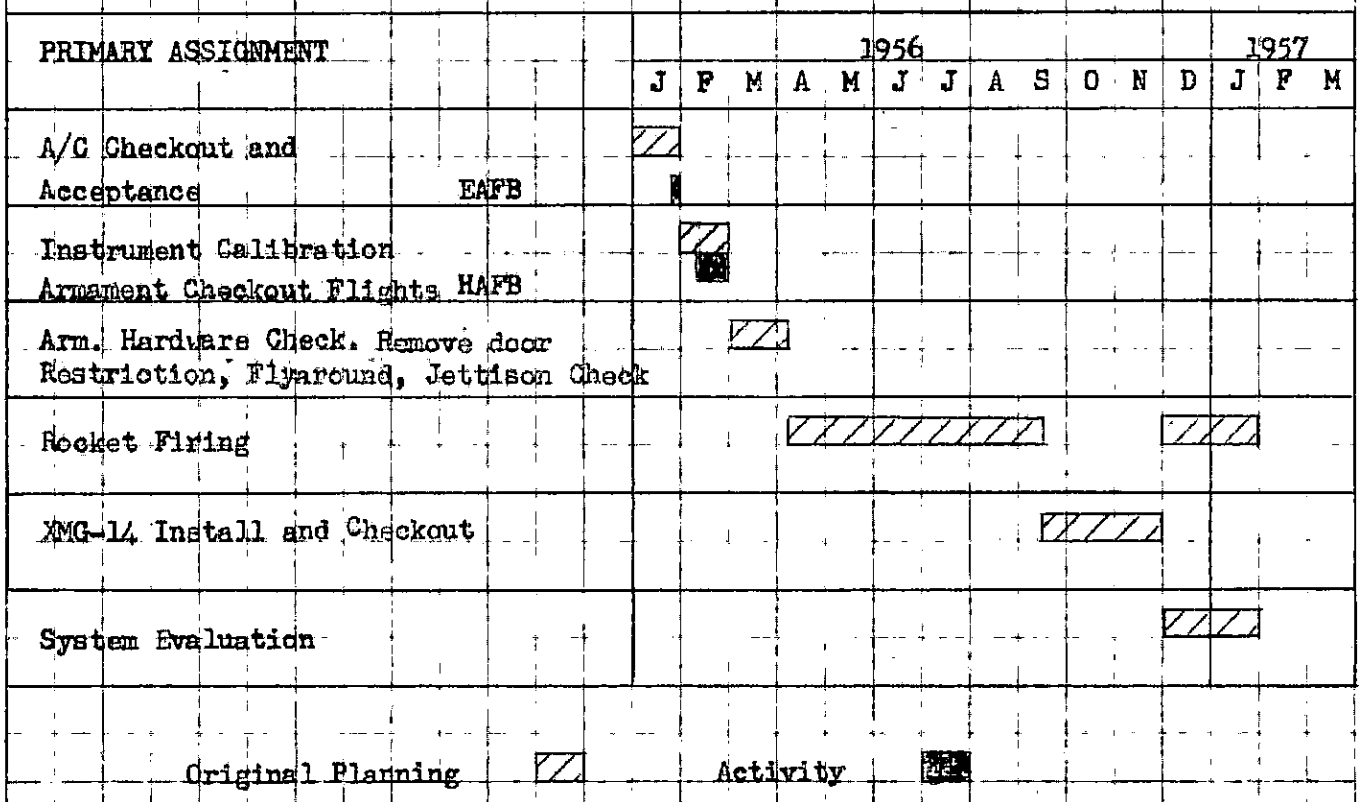

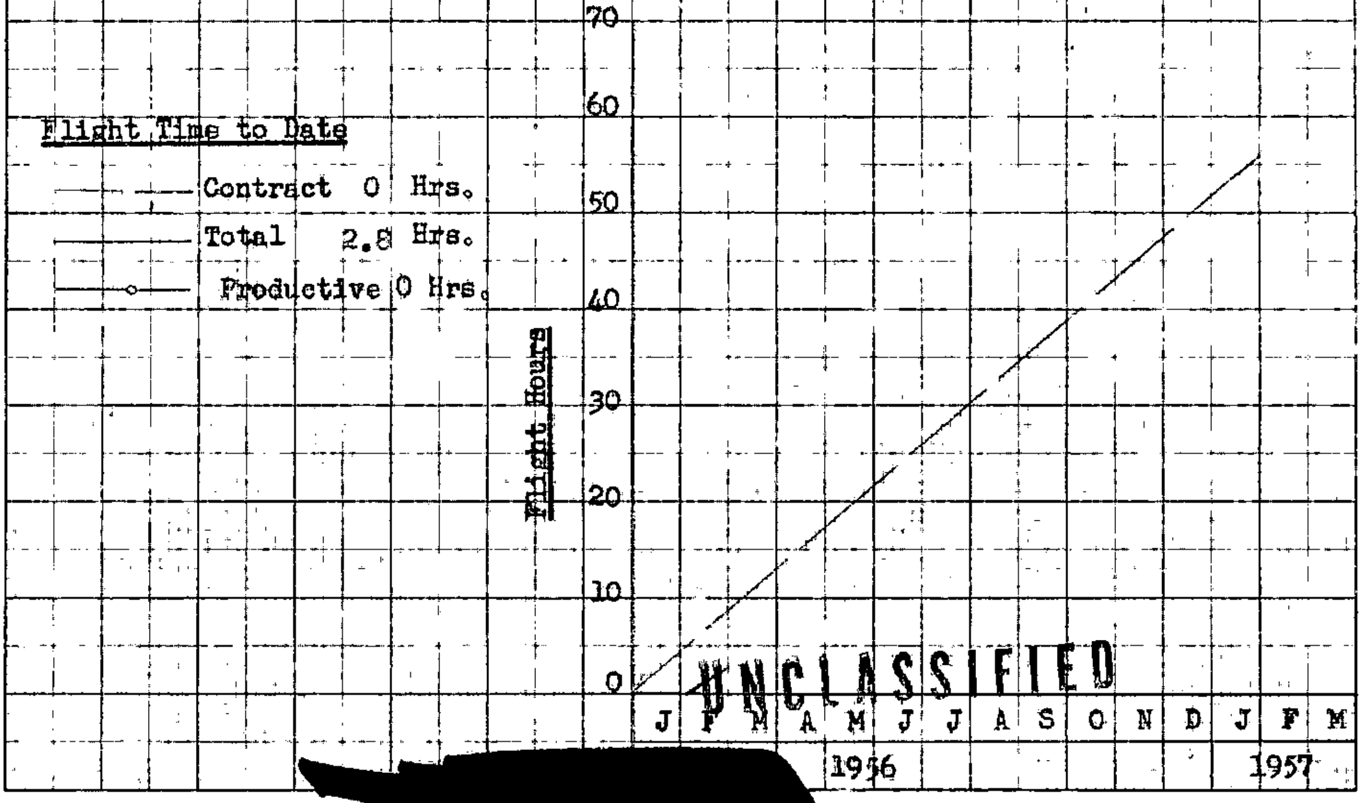

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES

AIRPLANE S/N 53-1798

Assignment

This airplane was assigned to Convair for a period of one month for antenna tests. Its formal statement of test assignment as presented in Supplemental Agreement No. 63 on Contract AF 33(600)-5942 is as follows: "Development and evaluation of flush antenna for fire control, CN and L as well as precipitation static investigation."

Development of flight controls will be accomplished during an 8-month period following the antenna tests. This is described in ZC-8-178 dated 11 October 1955. The program is shown on the following page.

Summary of Activity

Antenna testing per ZC-8-124 and Addendum 1 was accomplished from 22 September to 26 October 1955. Results of the tests were generally satisfactory but additional flights are required to complete the antenna program.

Flight tests with external wing tanks have been delayed due to lack of parts.

During February seven flights were performed for check out of specification compliance changes, pilot assist, pitch damper, yaw damper, turn coordinator and airspeed and outside air temperature calibration. This completed the control system's initial flight evaluation.

Future Activity

Control system ground tests will be conducted in March followed by external tank installation and antenna tests.

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SUMMARY OF ACTIVITY

S/N 53 - 1798

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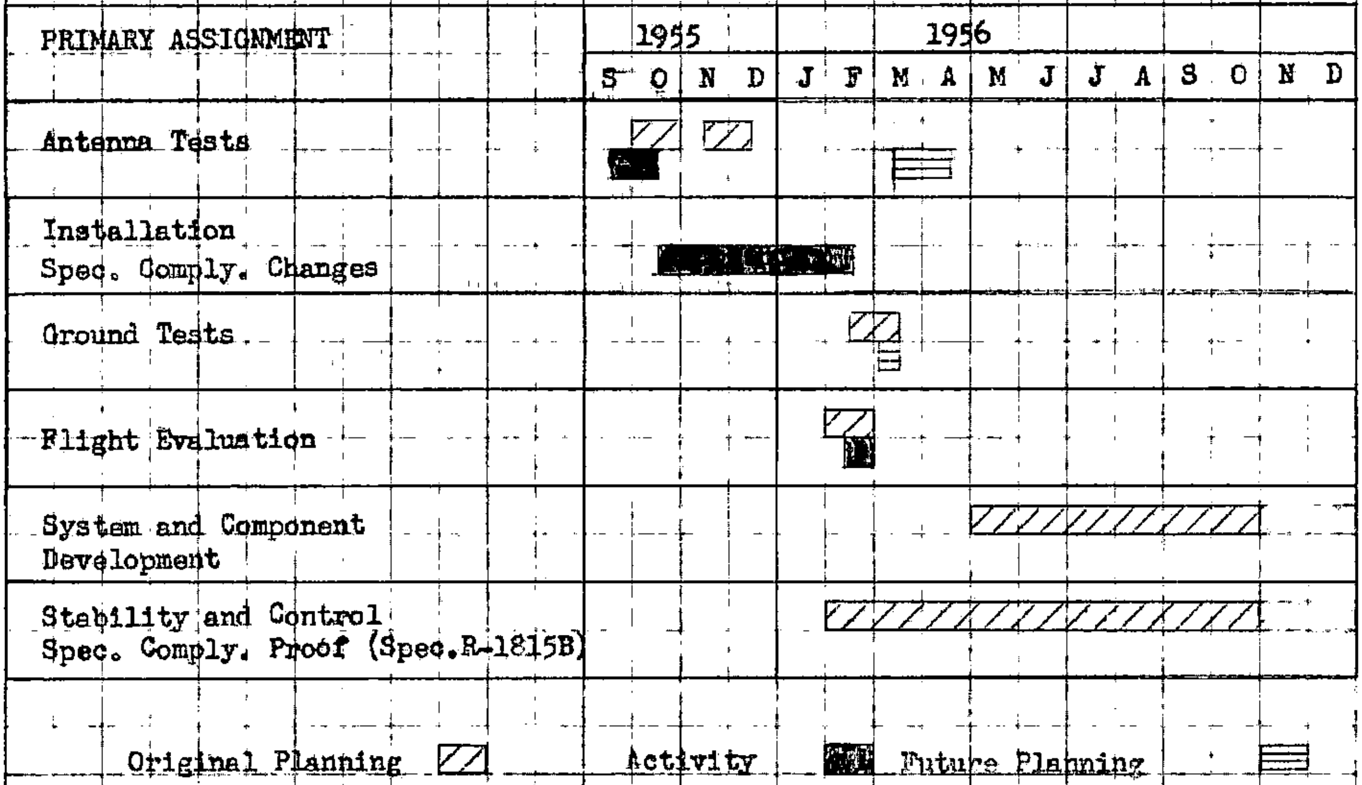
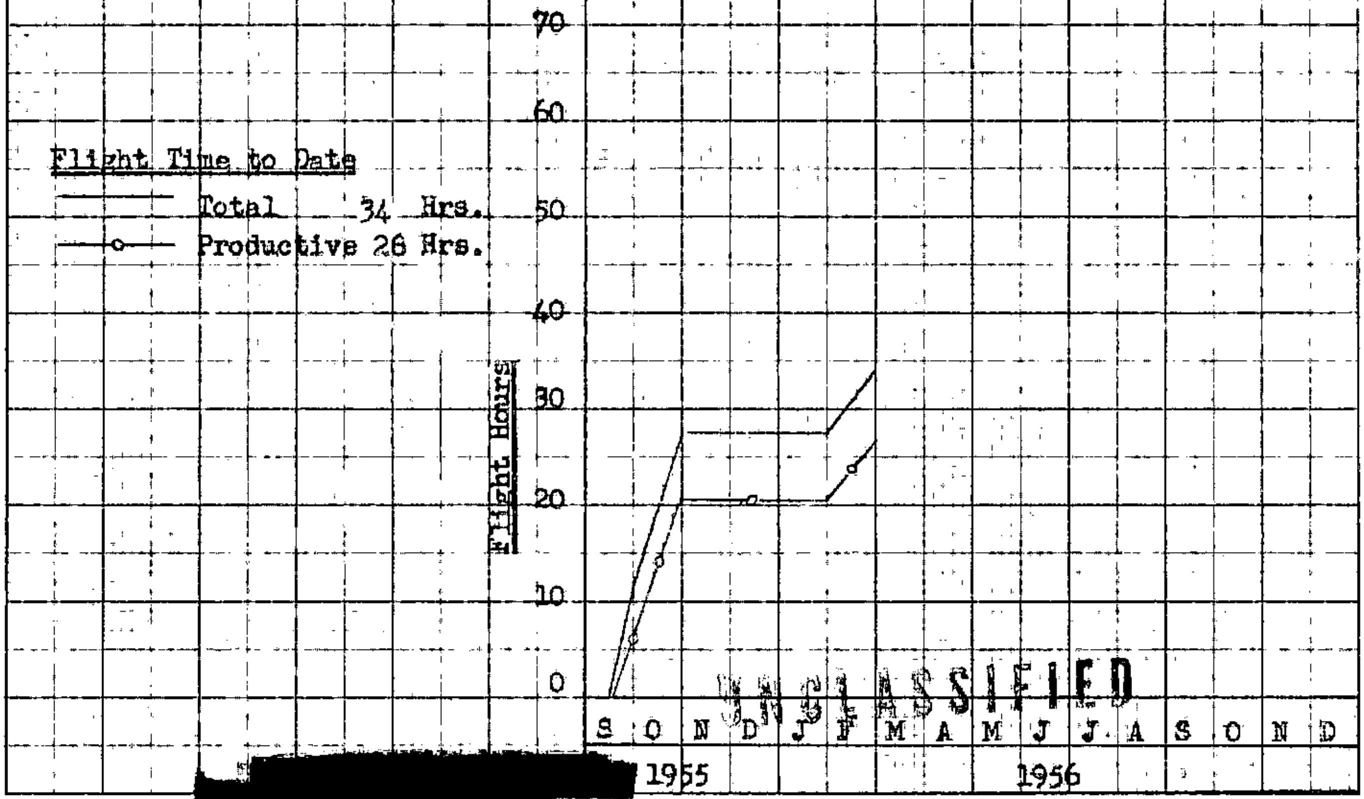


Convair Report No. ZC-8-124 (Antenna Tests)
Convair Report No. ZC-8-167 (Controls System Tests)



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1799

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Assignment

This airplane is being used for Weapons System Demonstration authorized by AMC letter MCPHFC (WCSFG) PRO-10 FJS/jem dated 24 June 1955. Activities are to be accomplished jointly by Convair and Hughes Aircraft Company on this ninth F-102A airplane, USAF S/N 53-1799.

"The tests will be conducted in two parts, Part I and Part II. Part I tests, conducted at Edwards Air Force Base, consists of tests for checkout and weapon system calibration."

"Part II tests will be conducted at Holloman Air Development Center. Part II consists of demonstration firing of instrumented missiles against parachute and drone targets. The final demonstration flight will be a salvo firing of tactical missiles against a drone target. Snap-up demonstration also will be conducted. The graph on Page 24 presents the program."

Summary of Activity

The airplane arrived at Edwards Air Force Base from Palmdale on 4 November 1955.

Ground work was conducted for installation of weapon system demonstration instrumentation, incorporation of specification compliance changes, and installation of a P-23 engine. Five flights were conducted in January for shake-down and miss evaluation.

During February five flights were performed. Two miss evaluation flights were flown and three airspeed calibration flights were completed. The F-102A production airspeed system was calibrated below Mach 0.85. The Fire Control System air data computer was calibrated except for the angle of attack inputs.

Future Activity

Completion of Part I at Edwards Air Force Base. Weapon system demonstration will follow at Holloman Air Force Base.

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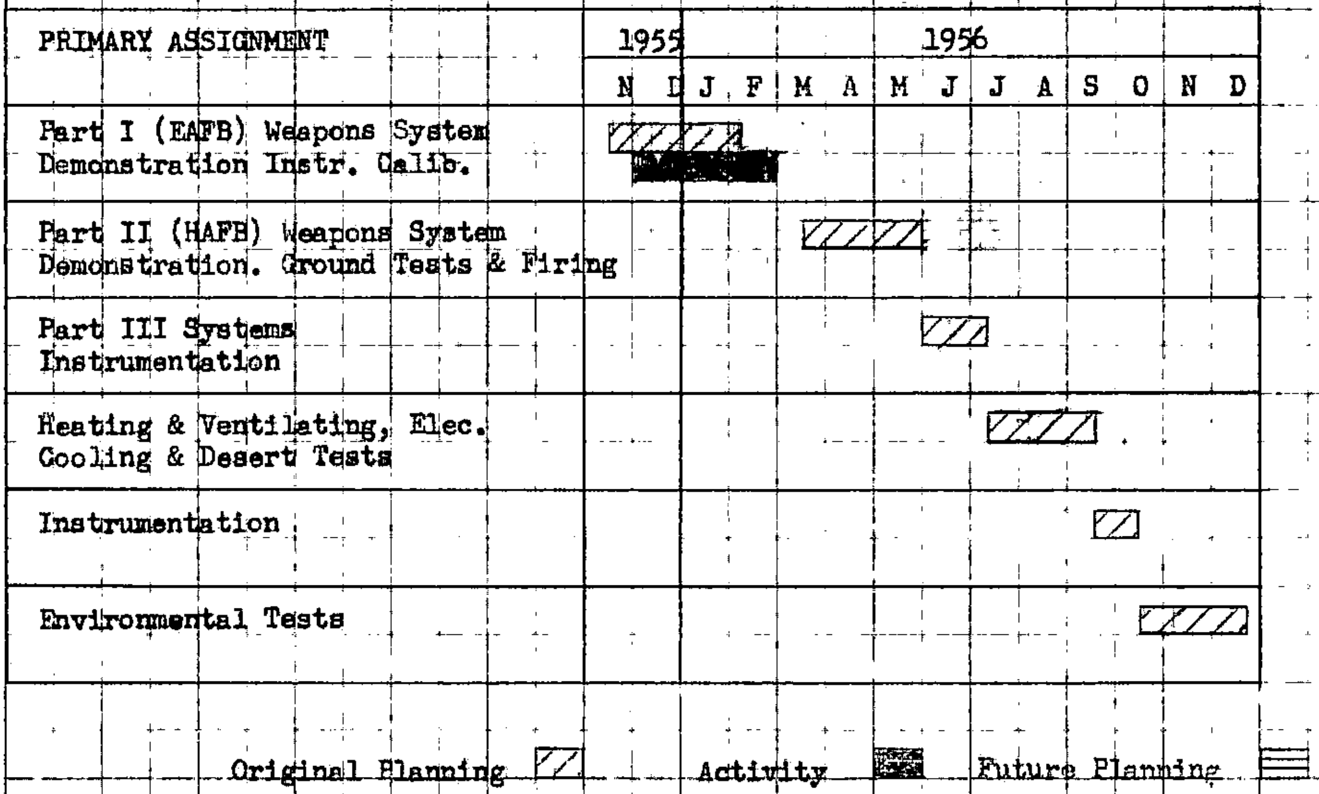
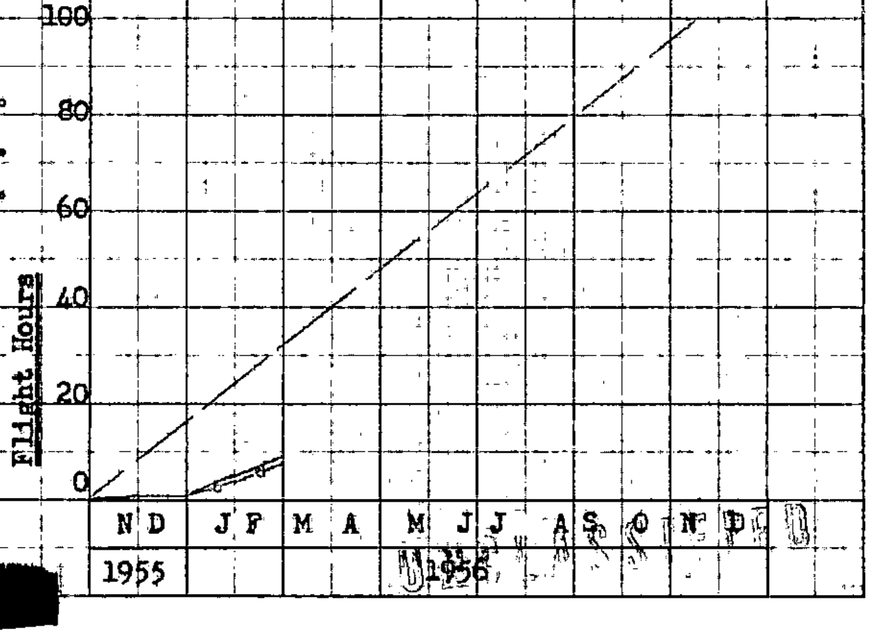
SUMMARY OF ACTIVITY

S/N 53-1799



Flight Time to Date

— — Contract 24 Hrs.
—— Total 9 Hrs.
—o— Productive 8 Hrs.



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1813

Assignment

This airplane is assigned to Convair for a period of 8 months. The formal statement of the flight test assignment as presented in CCN 6N- 2221(201) dated 31 October 1955 is: "This airplane shall be modified by installing a new vertical tail of approximately 94 square feet. The airplane shall then be flight tested for eight (8) months to determine the resulting improvement in directional stability and rolling performance. During the flight test period the Contractor shall revise the original modification as may be necessary ..."

Convair has divided this program into six general flight test stages. The graph on the following page shows the six stages as programmed and performed.

Summary of Activity

Roll coupling effects with the large tail have been the primary subject for flight tests for S/N 53-1813 since its arrival at Edwards Air Force Base on 10 December 1955. In addition to the roll performance evaluation, flutter and directional stability have been tested.

During February flights were performed to determine lateral response to roll maneuvers. Air Force pilots conducted evaluation flights for performance and roll characteristics. The results of testing indicates that roll coupling presents no problem with the large tail airplane. Roll tests were performed at what is believed to be the most severe condition of 12,000 feet altitude and M .95 to 1.0. The roll testing is considered to be completed.

Future Activity

Flight tests will be performed in March to remove the present negative "g" limitation (3 seconds) on the Sundstrand constant speed drive oil system, lateral stick force warning at 5 degrees displacement, elevon camera fairings, MLG fairing link loads and landing gear operating time and linear thrust fuel control. Thereafter the airplane will be returned to the San Diego Experimental Factory for the side-stick installation, as authorized by Contract AF 33(600)-30169.

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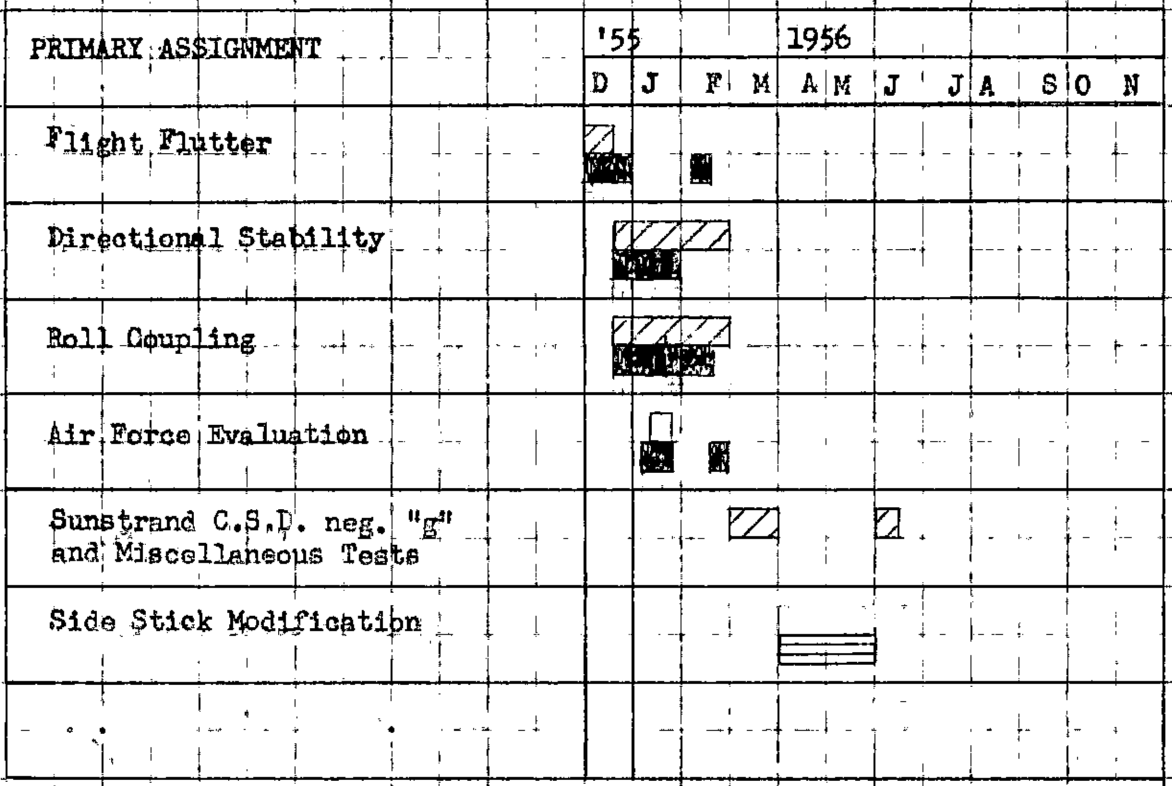
MMARY OF ACTIVITY

S/N 53 - 1813

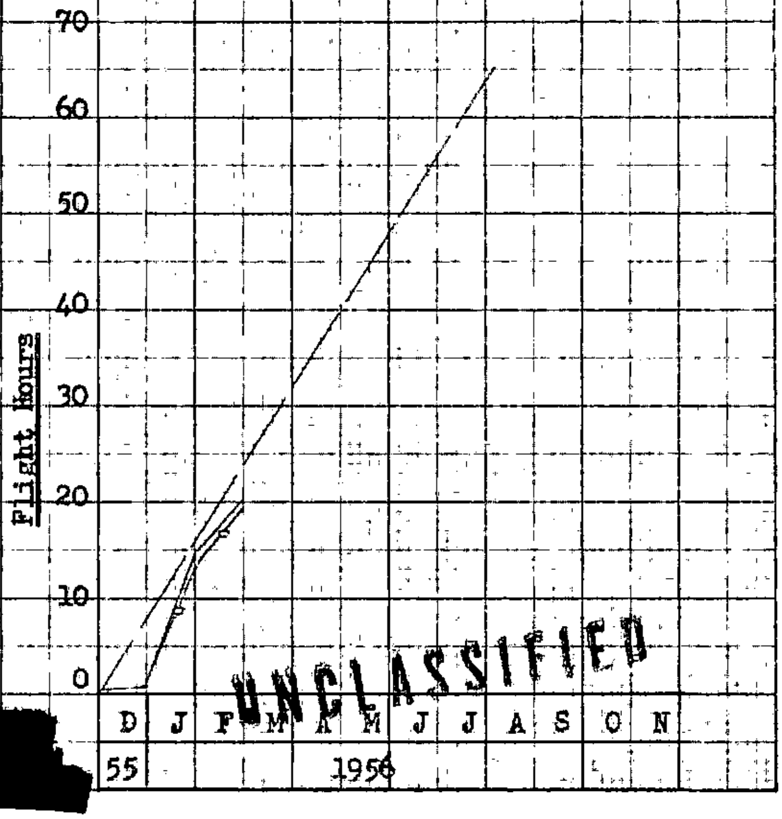


Original Planning ▨ Activity ■ Future Planning ☰

Convair Report No. ZC-8-171

Flight Time to Date

Contract	24	Hrs.
Total	20	Hrs.
Productive	19	Hrs.



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES

AIRPLANE S/N 54-1351

Introduction

This airplane was assigned to Convair for a period of 9 months for Phase I and II testing. Its formal statement of test assignment as presented in Sup. Agreement No. 70, 24 October 1955 is: "To be utilized for Phase I flight characteristics and basic performance, Phase II testing by Air Force and Phase II dynamic response, stability and control, power plant operation, general systems evaluation and structural integrity demonstration." (See graph, following page)

In order to accomplish the structural integrity demonstration it will be necessary to maintain the airplane for a period in addition to that described above. These tests cannot be accomplished until the data from 80 percent of limit load tests on S/N 53-1794 are available. The Air Force is cognizant of this requirement.

Summary of Activity

This airplane arrived at Edwards Air Force Base 18 October 1955. All flight testing has been primarily to investigate and correct buffet. Various configurations of spoilers, vortex generators and boundary layer slats were tried. The best buffet remedy is with 1-1/2" alternating vortex generators on the canopy leading edge in combination with boundary layer slats between the ducts and the fuselage. With this arrangement the airplane is considered to be satisfactory for the conducting of performance evaluation.

Future Activity

In March flight testing will be for airspeed calibration, performance testing and optimization of configurations for further remedying the buffet.

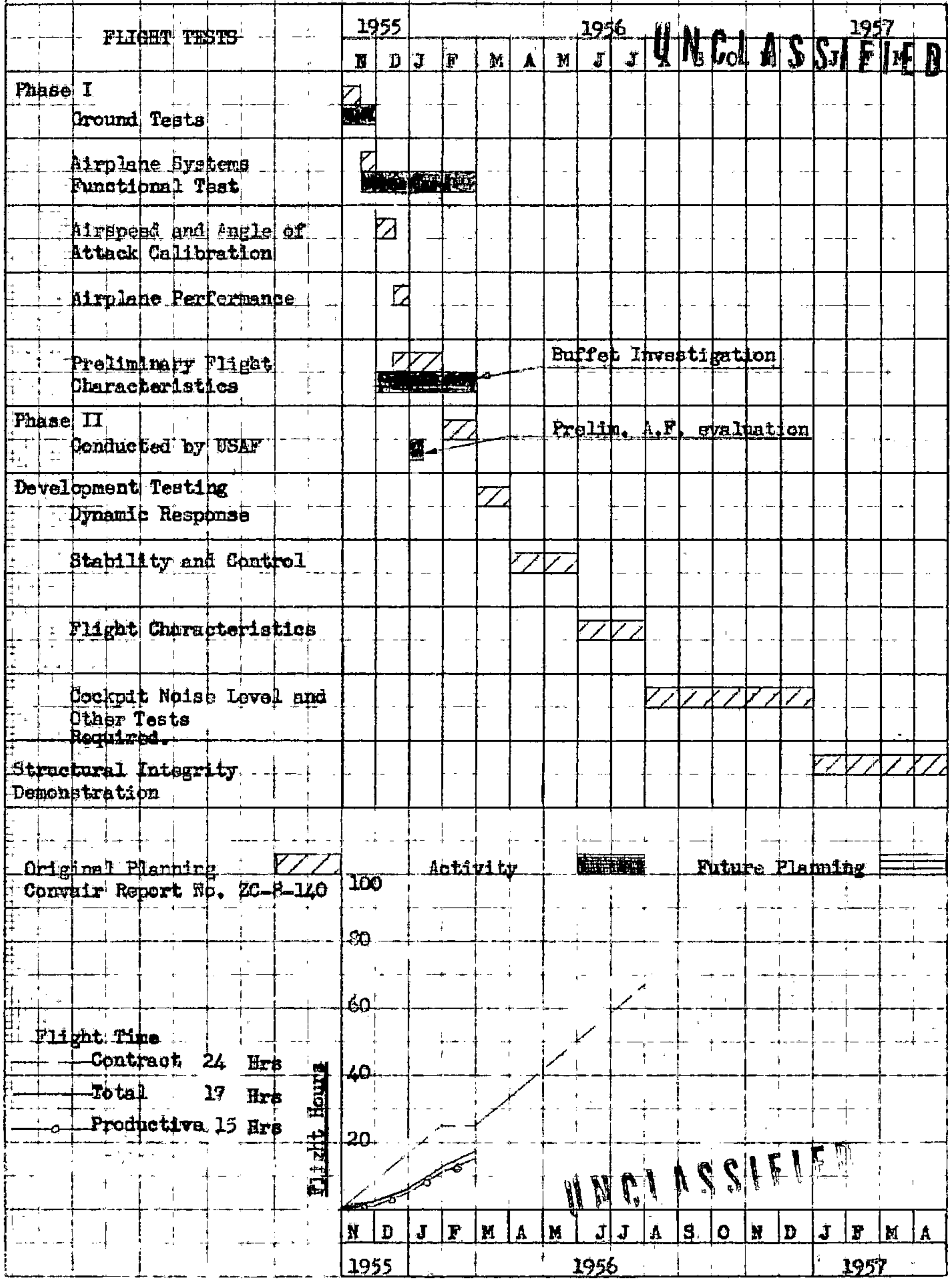
UNCLASSIFIED

[redacted]

Summary of Activity

S/N 54-1351

UNCLASSIFIED



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UNCLASSIFIED

UNASSIGNED TESTS

Tests to be conducted which have not been assigned to the program of a specific airplane will be listed in a later report.

UNCLASSIFIED

[redacted]

[redacted]

UNCLASSIFIED

AIRPLANES CURRENTLY
NOT
ASSIGNED
TO A
TEST PROGRAM
OR AT
SAN DIEGO FACTORY

UNCLASSIFIED

[redacted]

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

UNCLASSIFIED

AIRPLANE S/N 52-7994

Assignment

This airplane was assigned to Convair for a period of 18 months which commenced in October 1953. The test assignment of this airplane was to demonstrate and evaluate the basic flying and operating qualities of the YF-102 airplane. This program was authorized by Air Force Contract No. AF33(600)-5942, Supplemental Agreement No. 3. Graph, Page 32, presents the six flight test stages that were outlined in Convair Report No. ZC-8-101 and as performed.

Summary of Activity

Airplane S/N 52-7994 began its test program on schedule. No major mechanical malfunctions were noted. A slight modification to the landing gear was made and flutter tests were in progress. On 2 November 1953, on Flight No. 7, the airplane crashed* and was removed from the test program. This flight test program was then transferred to airplane S/N 52-7995, the No. 2 YF-102 airplane.

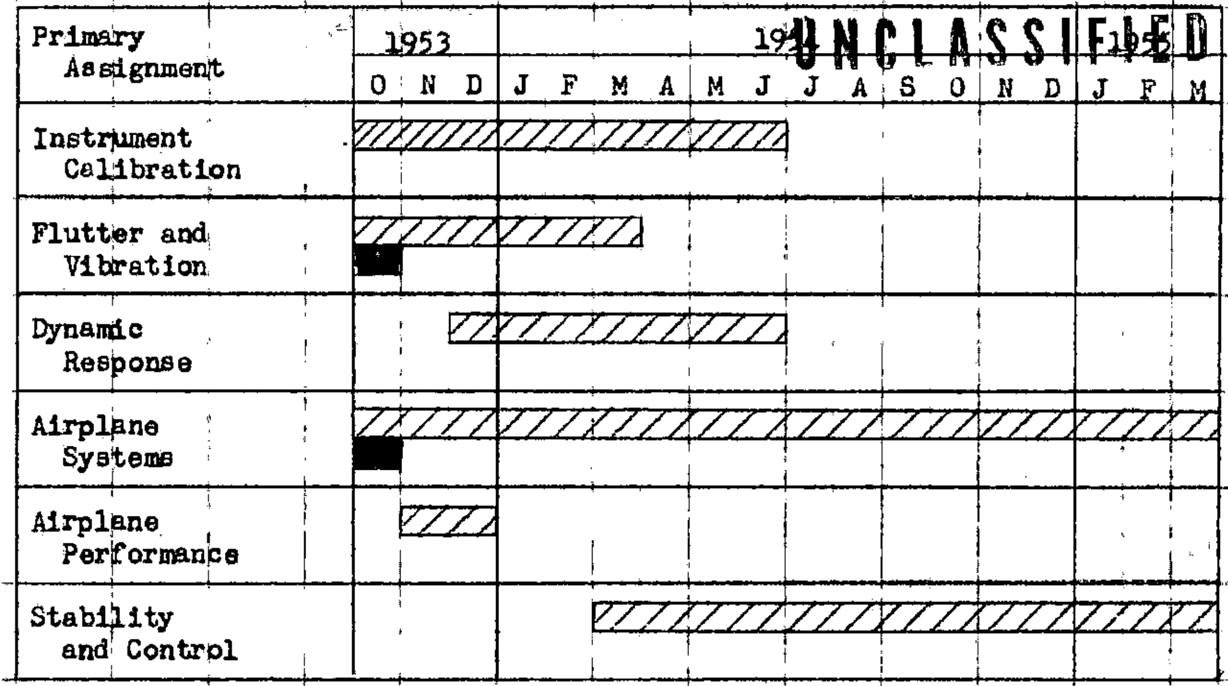
* Power loss on take-off. Gear-up landing off runway. Convair pilot survived.

UNCLASSIFIED

[redacted]

SUMMARY OF ACTIVITY
S/N 52-7994

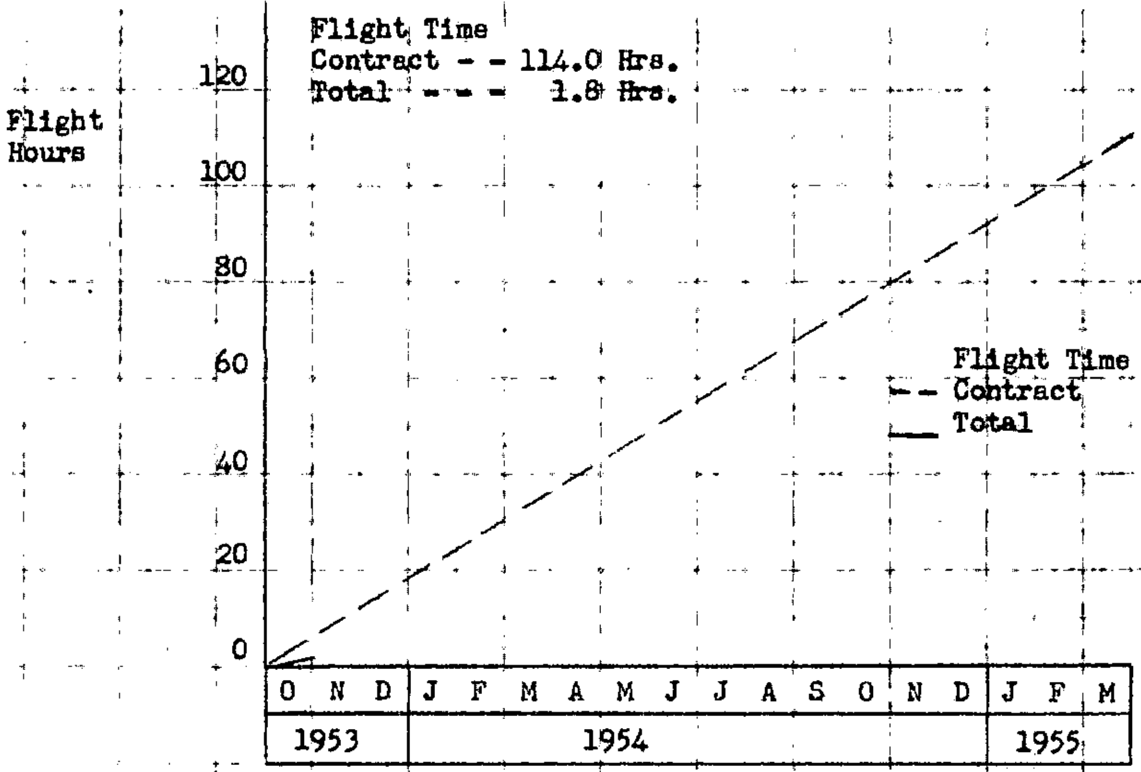
UNCLASSIFIED



Airplane Crashed 2 November 1953

Original Planning ▨ Activity ■

Convair Report No. ZC-8-101



UNCLASSIFIED

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UNCLASSIFIED

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES

AIRPLANE S/N 52-7995

Assignment

This airplane was assigned to Convair for a period of 17 months which commenced in January 1954. The formal statement of test assignment for this airplane as presented in Supplemental Agreement No. 63 dated 30 June 1955, was as follows: "To be utilized for safety of flight tests, initial power plant performance, cooling, pitch and yaw rate damper systems, and pilot relief development. Assume minimum flutter investigation and Phase II, which was not accomplished on No. 1 prototype." Convair divided this assignment into six general flight test stages in order to effectively accomplish the required flight test operation. Graph, Page 34, presents the six stages as originally programmed in Convair Report No. ZC-8-103, and as performed.

Summary of Activity

Preliminary airplane performance and airplane flight characteristics, including flutter and vibration tests, were completed in the early months of this flight test program. Stability and control tests continued throughout the program as a result of control system changes made to develop the 8-10 versions. Roll rate studies were accomplished in connection with flight characteristics and computer analysis.

This airplane was the second experimental model and performed its early flights with uncambered leading edges and straight wing tips. These were changed to cambered leading edges with reflexed tips and later the tips were changed to full camber without reflex.

Noise levels at crew stations during ground operations, cockpit noise, radome glycol distribution, performance and stability tests, wing fence evaluation, and many development tests of the engine components and systems were conducted during the period of 19 months of field operations and 163 flights.

Contributions to the F-102 program obtained from flight tests of S/N 52-7995 are:

A. Developed "Yellow Canary" tail fairing.
B. Demonstrated cambered leading edge of wings.
C. Obtained information on reflexed and cambered wing tips.
D. Developed extended nose.

Results from these items are incorporated in the 8-10 version of the F-102A airplanes.

The airplane has been assigned to Convair at Palmdale per Sales Order 63-1-125 as support aircraft for an indefinite period.

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UNCLASSIFIED

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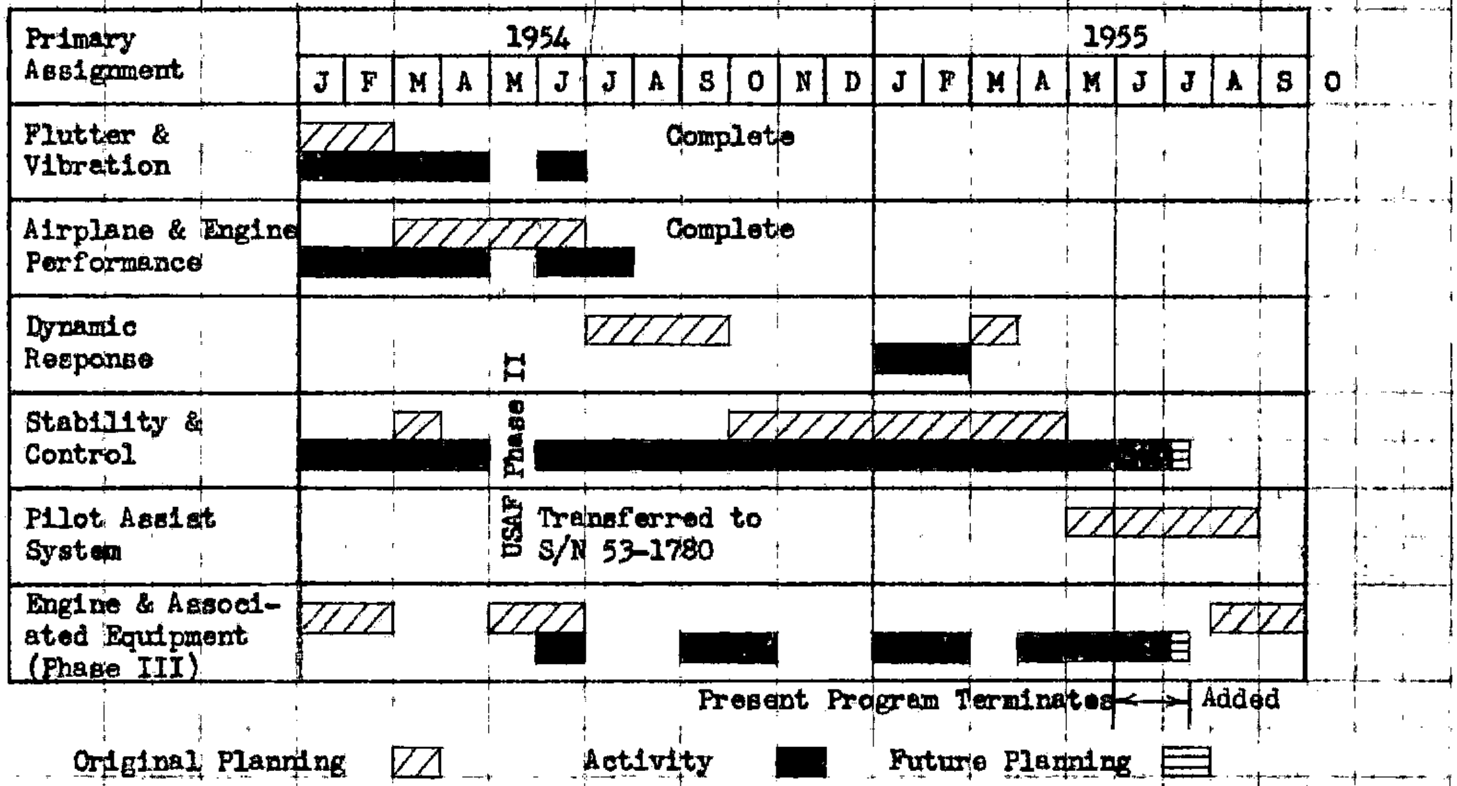
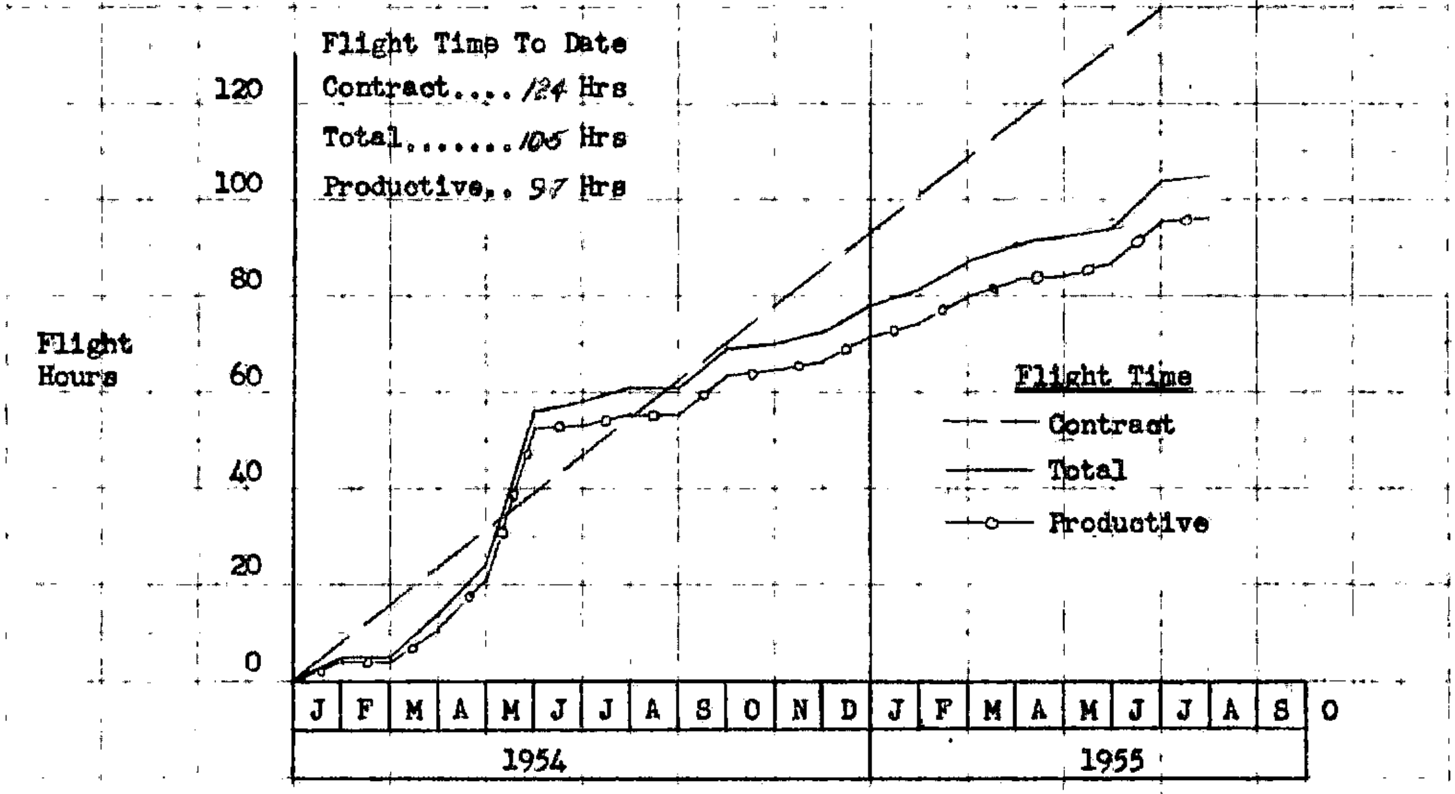
Summary of Activity

S/N 52-7995



Convair Report No. ZC-8-103



UNCLASSIFIED

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UNCLASSIFIED

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1779

Assignment

This airplane was assigned to Convair for a period of 4 months which commenced in May 1954. The primary purpose of Convair testing was the development testing of the pitch and yaw damper systems. This test program was authorized by Air Force Contract AF33(600)-5942, Supplemental Agreement No.63. Graph, Page 36, shows the three flight test stages which were outlined in Convair Report No. ZC-8-108 and as performed.

Summary of Activity

The first test flight of this program was conducted on 18 May 1954. The flight testing proceeded on schedule with dynamic response and stability and control, as related to the pitch and yaw damper systems, being the prime considerations.

Lateral stability tests proved satisfactory and the optimum yaw damper gain settings were obtained. The pitch damping system in its configuration at the end of the development program did not produce satisfactory damping at all airspeeds at both high and low altitudes with a constant gain setting.

The final flight of this program was conducted on 30 July when this aircraft was delivered to Palmdale, California, for modification prior to delivery to Hughes Aircraft Company.

This airplane has been assigned to the Convair Palmdale facility for pilot proficiency, target and chase plane missions per Sales Order No. 63-1-125.

[redacted]

UNCLASSIFIED

UNCLASSIFIED

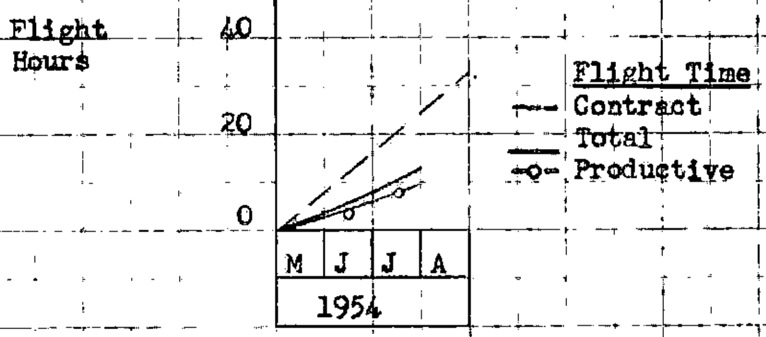
SUMMARY OF ACTIVITY
S/N 1779

Primary Assignments	1954 M	1954 J	1954 J	1954 A
Dynamic Response	■	▨ ■	▨ ■	▨
Stability & Control	■	▨ ■	▨ ■	▨
Instrument Calibration	■	▨		

Original Planning ▨ Activity ■

Convair Report No. ZC-8-108

Flight Time to Date
Contract ---- 32.0 Hrs.
Total ——— 12.6 Hrs.
Productive --- 9.7 Hrs.



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1780

Assignment

This airplane was assigned to Convair for a period of 15 months which commenced in July 1954. Its formal statement of flight test assignment as presented in Supplemental Agreement No. 63 dated 30 June 1955 was as follows: "To be utilized for Phase III testing of components sub-systems (pneumatic, hydraulic, electrical and control) tests and development." Convair divided this assignment into seven general flight test stages to accomplish the required flight test operation effectively. Graph on following page shows these seven stages as programmed and performed.

Summary of Activity

This airplane was utilized primarily for testing and refinement of the flight control and automatic flight control systems which will be incorporated in the production version. Dynamic response data was obtained for development of the Hughes Aircraft Company fire control system.

Results of tests on S/N 53-1780 have contributed greatly to the development of a satisfactory flight control system for the F-102 airplanes. These have proven that the design philosophy in the control system is sound and is developed to the point where the principal problems now are component reliability.

All control tests have been transferred to airplanes S/N 1787 and S/N 1798 except for Pneumatic Roll Rate Limiter test which was done in October 1955.

Convair's experimental flight testing of this airplane was completed on 11 October 1955.

This airplane has been assigned to the Convair Palmdale Facility for pilot proficiency, target and chase plane missions per Sales Order No. 63-1-125.

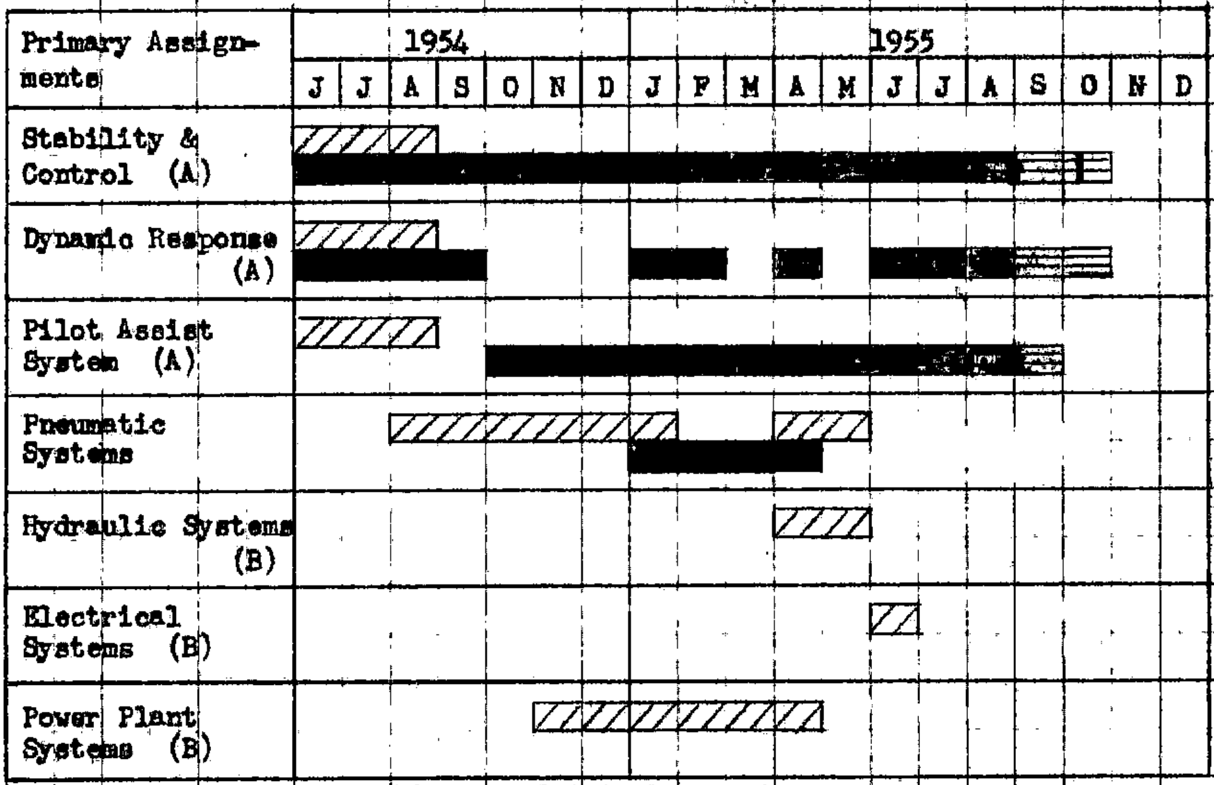
UNCLASSIFIED

[redacted]

Summary of Activity

S/N 53-1780

UNCLASSIFIED



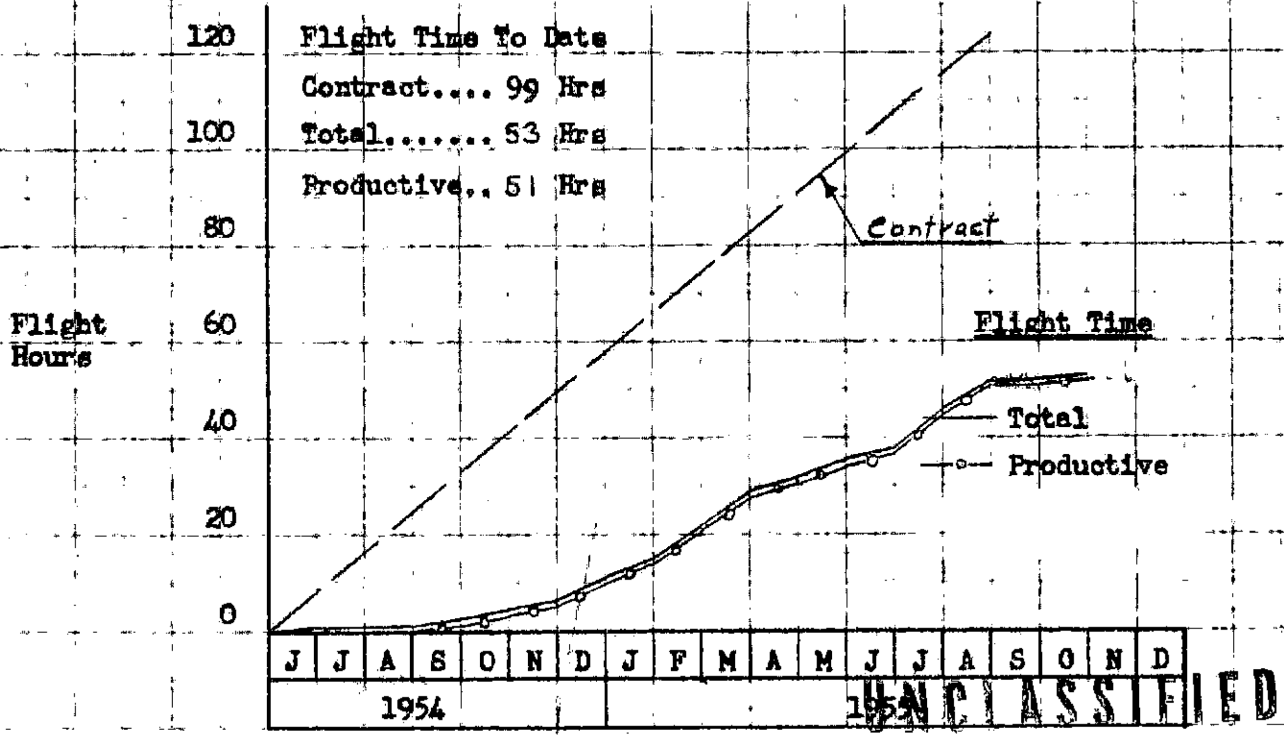
Added to Original Test Program → →

Convair Report No. ZC-8-111

Original Planning ▨ Activity ■ Future Planning ☰

(A) Test Assignments Transferred from Airplane S/N 53-1779.

(B) Cancelled. Spec. compliance items to be completed on 8-10 airplanes.



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

UNCLASSIFIED

AIRPLANE S/N 53-1781

Assignment - (Originally Programmed)

This airplane was assigned to Convair for a period of 14 months which commenced in June 1954. The formal statement of flight test assignment as presented in Supplemental Agreement No. 63 dated 30 June 1955 is as follows: "To be utilized for Phase III armament firing, functional tests and development, with normal and alternate armament installation." Convair has divided this assignment into five general flight test stages in order to effectively accomplish the required flight test operation. Graph, following page, shows these stages as originally programmed in Convair Report No. ZC-8-112 and as performed.

Summary of Activity - (Originally Programmed)

Armament mechanism operation and structure loads were partially completed in the early part of the test program and later completed in May. The captive missile loads and missile launching program were completed as well as the 2.75 and 2.0-inch rocket programs. The armament tests were generally satisfactory. Flight handling characteristics and stability at combat ceiling with the missile bay doors open were good.

Contributions to the F-102 program obtained from the originally scheduled flight tests of S/N 53-1781 are:

A. Information on separation, dispersion and trajectory of rockets and missiles, satisfactory.
B. Airplanes response to rocket and missile firing, satisfactory. Firing full load of rockets at maximum altitude caused no "snuffing".
C. Developed ATM to take care of negative G flight condition.

This airplane completed its originally assigned flight test task. It has been modified at the San Diego Experimental Factory to a Rocket Type MB-1 launching configuration.

"CURRENT ACTIVITY - PAGE 11 "

UNCLASSIFIED

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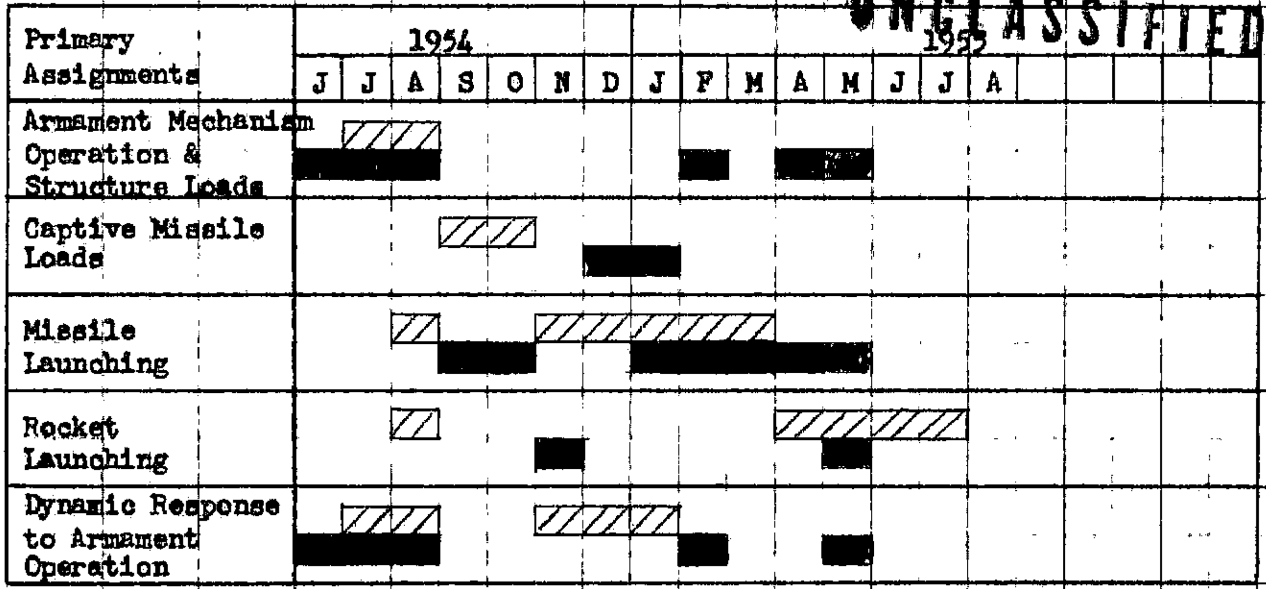
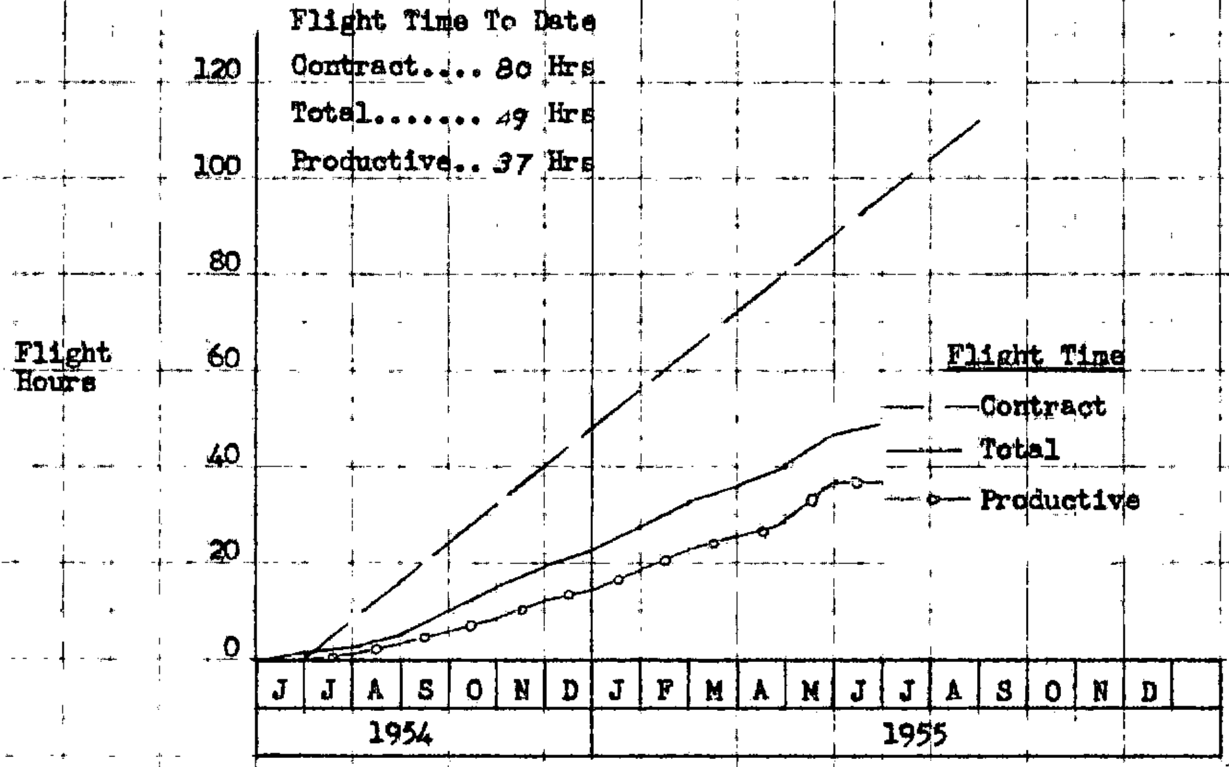
Summary of Activity
S/N 53-1781
(Originally Programmed)

UNCLASSIFIED



Original Planning ▨ Activity ■ Future Planning ☰

Convair Report No. ZC-8-112



Flight Time To Date
Contract.... 80 Hrs
Total....... 49 Hrs
Productive.. 37 Hrs

UNCLASSIFIED

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UNCLASSIFIED

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1782

Assignment

This airplane was assigned to Convair for a period of one month to demonstrate that the functional characteristics of the production antenna installations were satisfactory for communication and navigation equipment for the F-102 airplane. The testing was to further correlate flight measurements with antenna model studies made at Convair. This testing was authorized by Air Force Contract No. AF 33(600)-5942. The testing undertaken on this airplane is shown on graph on page 42.

Summary of Activity

Although airplane S/N 53-1782 was scheduled for testing in August 1954, testing commenced 28 June 1954. After the initial shakedown flights, instrument calibrating was the first undertaking. Performance, communications and navigation data were obtained in July. Compressor stall investigations were conducted concurrently with the communications and navigation tests. Satisfactory results were obtained from these functional tests of the equipment.

The test program was completed on 18 August 1954, when the aircraft was delivered to Palmdale, California, for modification prior to delivery to Hughes Aircraft Company.

Contributions to the F-102 program obtained from flight tests of S/N 53-1782 are:

A. Determined that all antennas are compatible.
B. Completed part of the F-102 antenna specification compliance testing.
C. Demonstrated operation of the emergency fuel system, ring type dacron drag chute and electrical system power controller.

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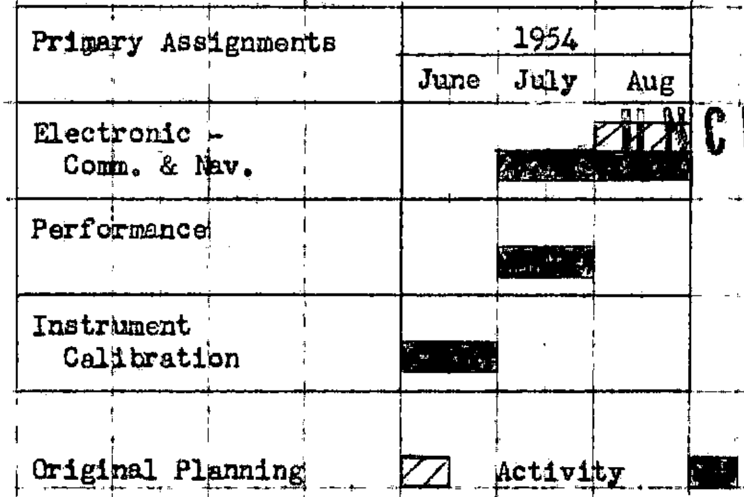
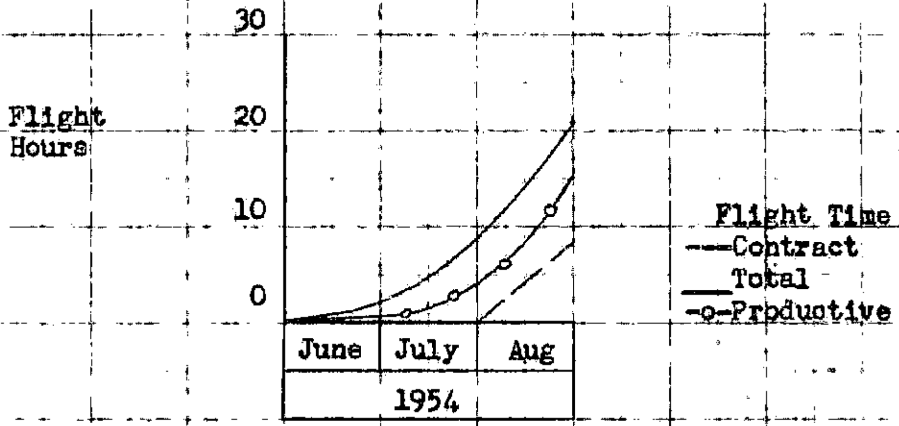
[redacted]

SUMMARY OF ACTIVITY
S/N 1782



UNCLASSIFIED

Flight Time to Date
Contract ---- 8 Hrs.
~~Total ——— 21.1 Hrs.~~
Productive-- 15.7 Hrs.



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UNCLASSIFIED

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1783

Assignment

This airplane was assigned to Convair for a period of 10 months which commenced in August 1954. The formal statement of flight test assignment for this airplane as presented in Supplemental Agreement No. 63 dated 30 June 1955 is as follows: "To be utilized for Phase III tests to include cooling problems. ATM refrigeration unit, cockpit air-conditioning, and correction of deficiencies disclosed during Phase I and II. The aircraft may also be used for Pilot Assist System development in event of difficulties with airplane S/N 52-7995." Convair has divided this assignment into seven general flight test stages in order to effectively accomplish the required flight test operations. Graph on page 44 presents these seven stages as originally programmed in Convair Report ZC-8-114, the periods of flight test activity on each stage, the future periods of activity currently planned on each stage, and the flight time accumulated to date.

Summary of Activity

The original flight test program for S/N 53-1783 was changed considerably. Instrumentation calibration, production items, engine and structural cooling, and pilot assist phases of the test program were deleted to permit delivery of this aircraft to WADC in January 1955. Dry air anti-icing, defogging, and cabin ventilation and pressurization tests were limited to qualitative testing to permit delivery to WADC in January 1955.

An added task of communications and navigation systems testing was completed during the second month of the program. These tests were transferred from S/N 1782.

Air Force pilots evaluated S/N 53-1783 in various icing conditions. In numerous flights, artificial icing conditions were employed behind tanker type aircraft. Preliminary information indicated satisfactory performance of this airplane in the icing conditions encountered. The anti-icing tests were completed on 3 May with Flight No. 82. As a result of the tests in icing conditions without wing and tail anti-icing, the anti-icing equipment has been deleted from the 8-10 airplane with a considerable saving in weight. S/N 53-1783 was ferried from WADC to EAFB and arrived at Edwards on 15 May. This airplane has been assigned to Convair at Palmdale per Sales Order 63-1-125 for pilot proficiency, target and chase plane missions.

Contributions to the F-102 program obtained from flight tests of S/N 53-1783 are:

A. Determined that the F-102 does not require wing and tail anti-icing.
B. Completed antenna specification compliance testing.
C. Demonstrated instrument flying characteristics.
D. Demonstrated night flying characteristics.

UNCLASSIFIED

[redacted]

Summary of Activity
S/N 53-1783

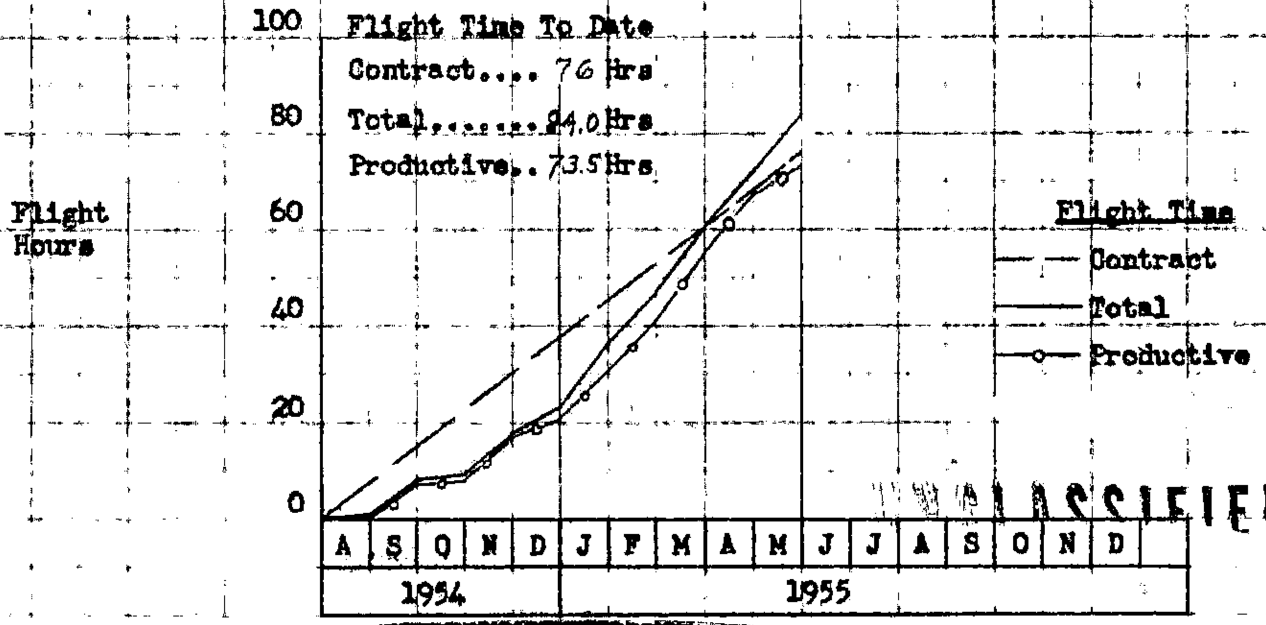
UNCLASSIFIED

Primary Assignments	1954					1955												
	A	S	O	N	D	J	F	M	A	M	J	J	A					
Instrumentation Calibration Production Items	▨			Deleted (A)														
Dry Air Anti-icing, defogging, cabin ventilation & pressurization	▨	▨	▨ ■	▨ ■	▨ ■				■	■		Deleted (B)						
Engine & structural cooling & cooling drag						▨	▨	▨				Deleted (A)						
Dynamic Response			Deleted (A)						▨	▨	▨							
Pilot Assist System			Deleted (A)									▨	▨	▨				
Natural Icing Flights - WADC		Added (C)				■	■	■	■	☰								
Communications & Navigation Systems		■	■			Added (D)												

Convair Proposes Termination ←

Original Planning ▨ Activity ■ Future Planning ☰

(A) Deleted to permit delivery of A/C to WADC Jan. 1955.
(B) Wing & tail anti-icing, cabin pressurization deleted. Cabin vent & defogging limited to qualitative tests to permit delivery to WADC Jan. 1955.
(C) Added to original test program - tests are being flown by WADC personnel.
(D) Transferred from Airplane S/N 53-1782.


Flight Time To Date
Contract.... 76 Hrs
Total....... 84.0 Hrs
Productive.. 73.5 Hrs

UNCLASSIFIED

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UNCLASSIFIED

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1793

Assignment

This airplane was assigned to Convair for a period of 12 months commencing in August 1955. Its formal statement of flight test assignment as presented in Supplemental Agreement No. 63 on Contract AF 33(600)-5942 is as follows: "To be utilized for Phase III tests to include subsonic verification of armament tests and supersonic test and functional development of armament including hardware and dispersion studies with production airframe systems. These tests will be conducted up to combat altitude and maximum speed." This assignment has been divided into six general flight test stages in order to fulfill the contract requirements effectively. The graph on the following page presents the program.

Summary of Activity

The structural loads imposed on the armament displacement mechanism were measured with dummy missiles in place and the airplane on the ground. The loads were within design limits.

Armament gear check-out tests were made in subsonic and supersonic flights at loads up to +3g. All loads were well within limits and gear operation times were acceptable. During December missile firing flights to check missile separation were in progress. On December 14 the airplane was damaged in a landing incident and as a result all tests have been delayed.

On January 10 the fuselage was shipped by truck to San Diego (Sales Order 33-2-327) for a two-month repair period.

Future Activity

Flight tests will be started in April on hardware performance, A/C stability and response to armament operation, missile firing, rocket separation, trajectory and dispersion with 24 rocket doors and 36 rocket doors, armament bay air loads and corrosion effects.

UNCLASSIFIED

[redacted]

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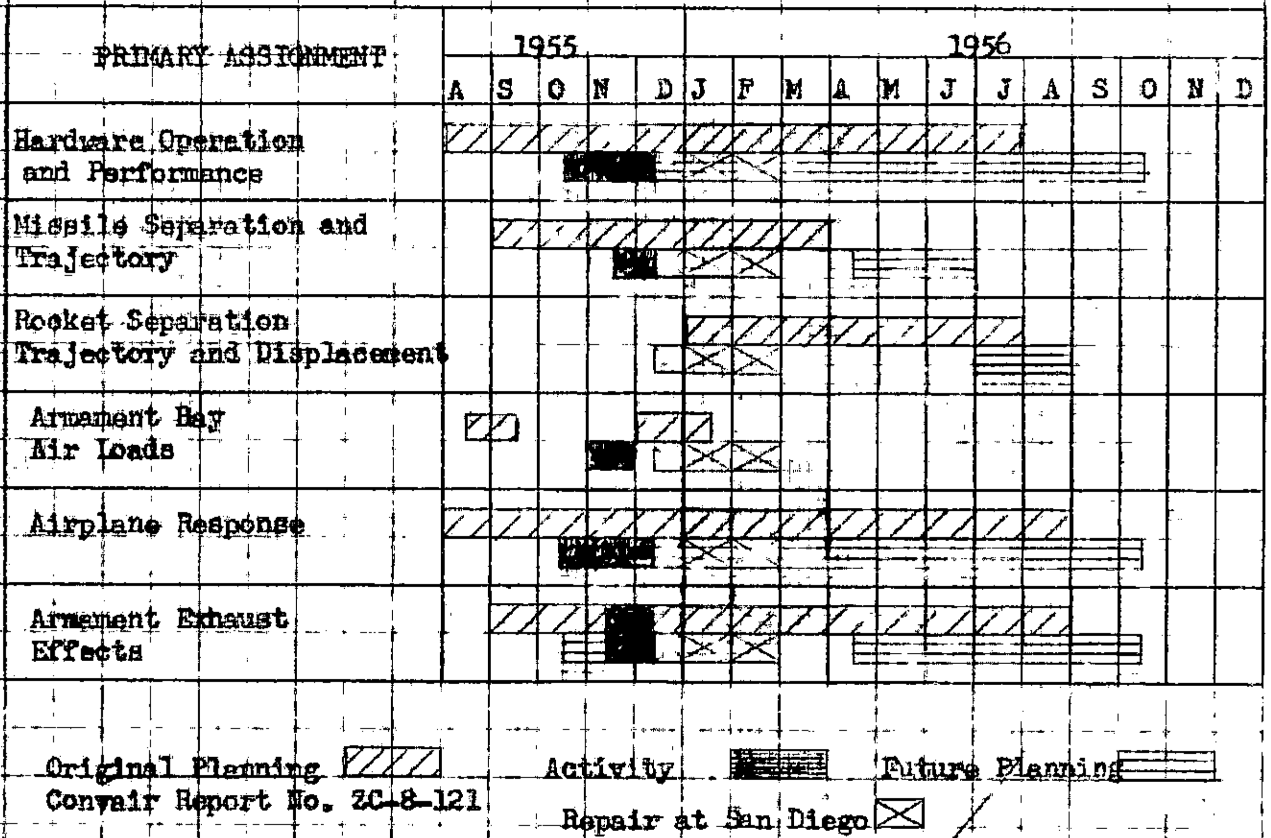
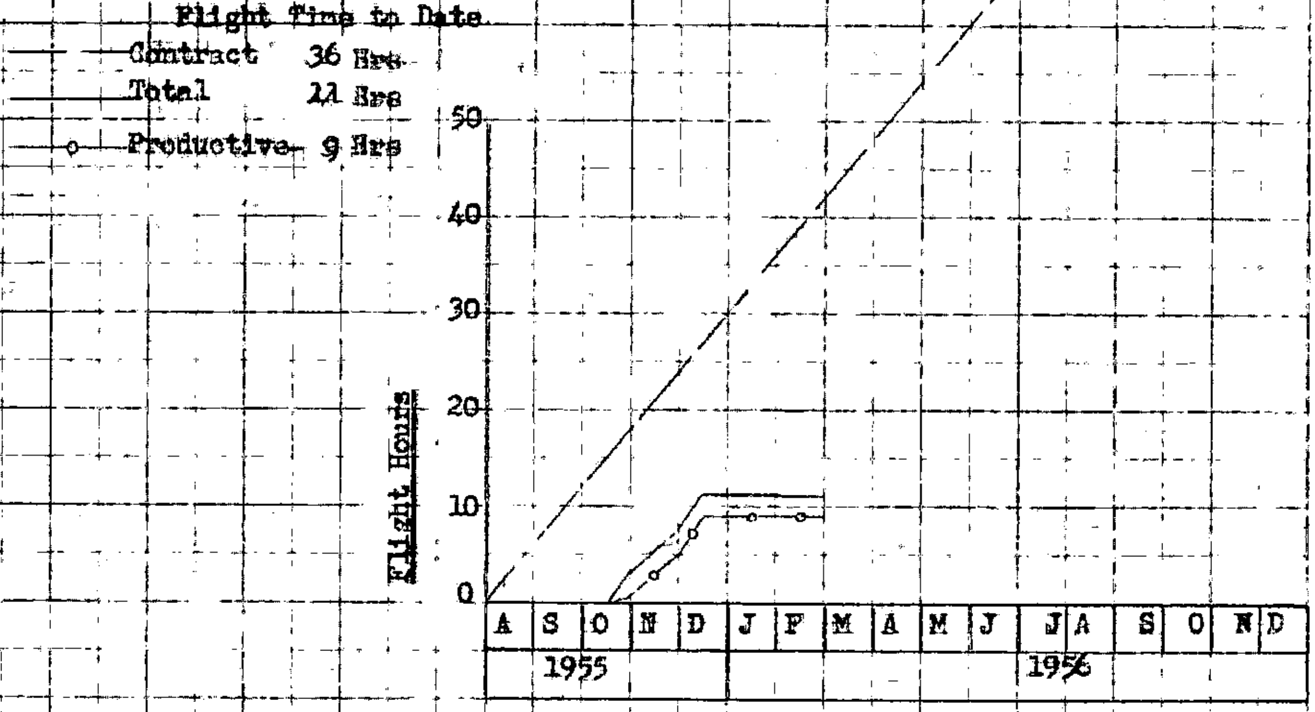
[illegible] -1?2-6
Report [illegible] P-10?
Date 1 March 1956

ASSIFIED

Summary of Activity

S/N 53-1793





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UNCLASSIFIED

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES

AIRPLANE S/N 53-1794

Assignment

This airplane was assigned to Convair for a period of 13 months. Its formal statement of test assignment as presented in Supplemental Agreement No. 63 on Contract AF 33(600)-5942 is as follows: "To be utilized for flight loads determination, operation of high pressure pneumatic system and structural integrity demonstration." Convair divided this program into eight general flight test stages to accomplish the required flight test operation effectively. Graph on following page shows the eight stages as programmed and performed.

Summary of Activity

The first shakedown flight and Air Force acceptance of this airplane was on 25 August 1955. Extensive instrumentation installation, rework, and checks with numerous delays caused by fuel leaks in the pressure manifolds and eight check flights were conducted throughout the rest of the stay at Edwards AFB. Extra instrumentation was required to investigate flight loads instrumentation drift.

In February the S/N 53-1794 was ferried to the San Diego Experimental Factory for a four-month period of structural modification to bring the airplane to the final production configuration and for specification conformance changes and further instrumentaion.

Future Activity

After 21 June 1956, there will be a shakedown period, during which the airplane's flight characteristics and instrument calibrations will be determined. Flight tests of the scheduled maneuvers will begin in the middle of July and will continue to a scheduled conclusion of the first part of the program by 31 December 1956. By that time demonstration of analytical loads will have been completed.

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SUMMARY OF ACTIVITY
S/N 53-1794

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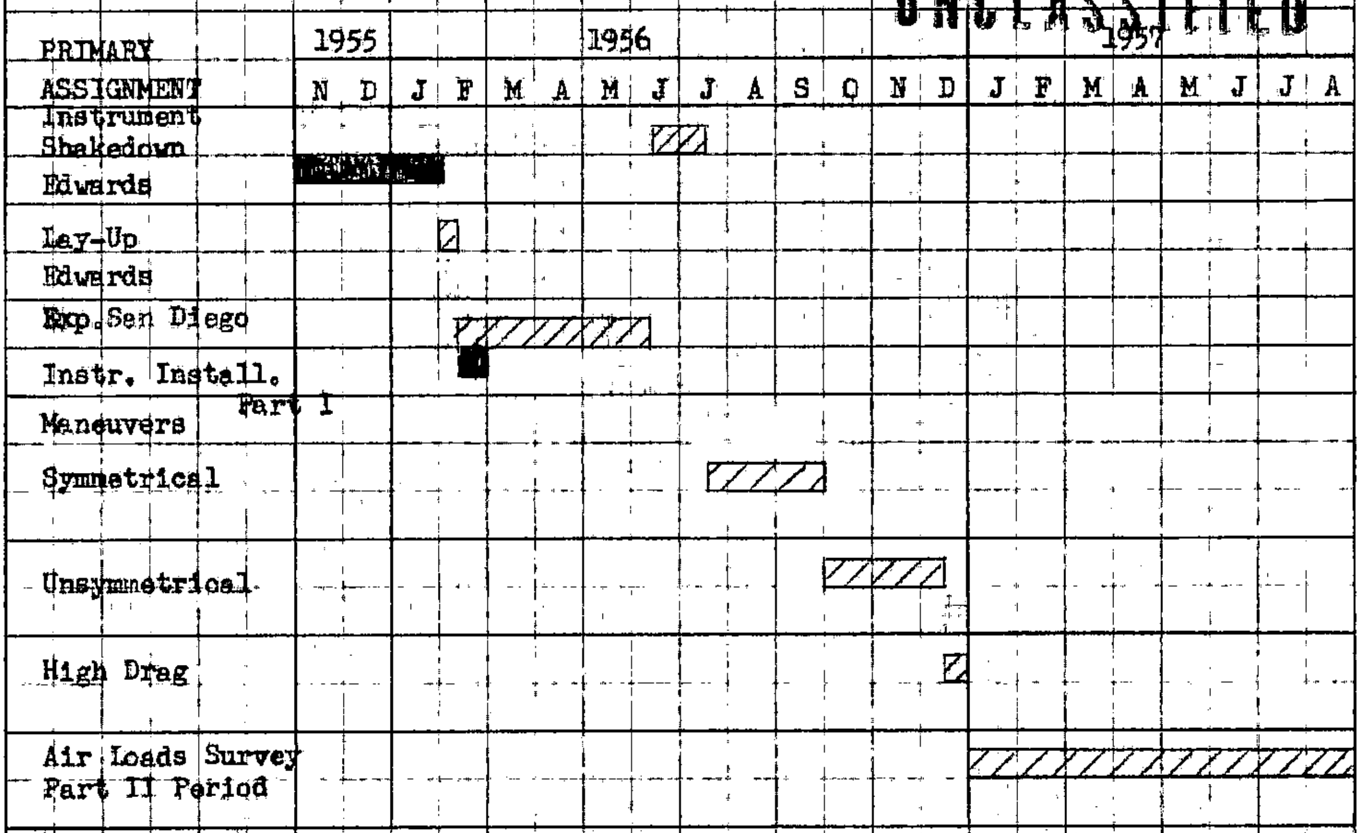
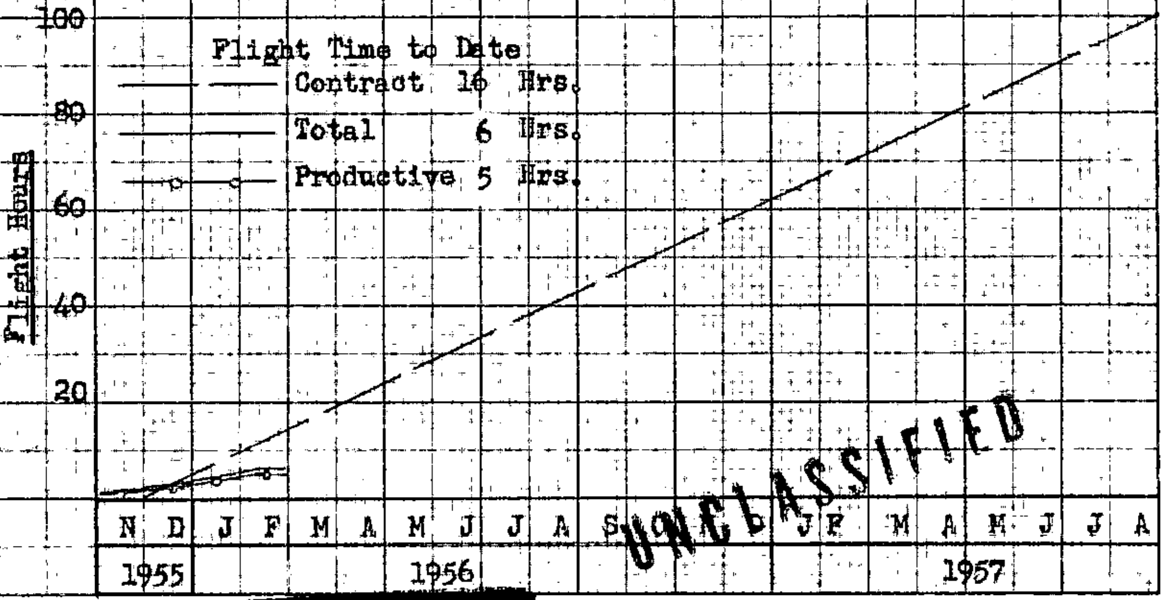


Original Planning ▨ Activity ■

Convair Report No. ZC-8-122 Revision -1



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DEFINITIONS

UNCLASSIFIED

For purposes of this report, the following definitions apply:

(1) Contract Flight Time - Based on 8 Hrs/maintenance month.

(2) Total Flight Time - Airplane total flight time during the period the airplane is assigned to Convair for Flight Testing.

(3) Productive Flight Time - Flight time during which data in furtherance of the development test program was obtained. Air Force Phase II testing is included in this time.

(4) Convair Test - Flights in which the aircraft has taken off with a Convair pilot, in direct pursuit of fulfilling a portion of the scheduled test program (includes aborts).

UNCLASSIFIED

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UNCLASSIFIED

CLASSIFICATION CHANGED TO:

UNCLASSIFIED

AUTHORIZED BY: DD 254 DATE 12/12/66

CONTRACT NO. F106/31546/ite

RECLASSIFIED BY: RCB

DEPT. 110-2 DATE 6/24/68

L. G. AVRITT, SECURITY OFFICER

CONVAIR DIVISION OF GENERAL DYNAMICS

UNCLASSIFIED

[redacted]

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2721

COPY NO. 12

File 182

Classification Changed to: UNCLASSIFIED

Authorized by: ED 254 Date: 6/10/60

Reclassified by: FCB Dept: 11 Date: 9/11/60

TITLE

F-102 ACCELERATED DEVELOPMENT TEST PROGRAM STATUS OF FLIGHT TEST ACTIVITIES

September, 1955

CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

CONFIDENTIAL

2721

CONSOLIDATED VULTEE

DIVISION

MODEL F-102

REPORT ZC-8-142-1

DATE 3 Oct 1955

TITLE

F-102 ACCELERATED DEVELOPMENT TEST PROGRAM STATUS OF FLIGHT TEST ACTIVITIES

September, 1955

PREPARED BY Lloyd Child

GROUP Engineering Flight Test

REFERENCE

CHECKED BY T. L. Maloy
T. L. MALOY

APPROVED BY P. M. Prophett
P. M. PROPHETT
Chief of Engineering Flight Test

NO. OF PAGES

NO. OF DIAGRAMS

REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED
1	Oct 1955	L. Child	Sept. 1955 data amplifies previous information	

GO 241A

CONFIDENTIAL

CONFIDENTIAL

DIVISION

MODEL F-102

REPORT ZC-8-142-1

DATE 3 Oct 1955

TITLE

F-102 ACCELERATED DEVELOPMENT

TEST PROGRAM

STATUS

OF

FLIGHT TEST ACTIVITIES

September, 1955

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REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED
1	Oct 1955	L. Child	Sept. 1955 data amplifies previous information	

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CONFIDENTIAL

INTRODUCTION

This summary report is presented to show the progress of the Accelerated Development Test Program for the F-102 airplane. Contractual commitments are compared with actual accomplishments on those airplanes currently being flight tested by Convair.

SUMMARY OF FLIGHT TEST ACTIVITY

The F-102 Accelerated Development Test Program flight operations conducted by Convair are proceeding according to schedule. Convair has completed testing on five of the 18 airplanes which will eventually be assigned to Convair for flight operations. To date, Convair test airplanes have been flown a total of 468 hours during 695 flights.

The periods of individual airplane flight test utilization and a summary of total flight time is presented in Graphs No. 1 and 1A. A summary of the test program and flight time to date by individual airplanes is presented in Graphs No. 3 through 12. A summary of flight time and the general purpose of the flights accomplished to date on all Convair test airplanes is presented in the following table:

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SUMMARY OF FLIGHT TEST ACTIVITIES FOR SEPTEMBER 1955

Convair test airplanes flew 29.1 hours in 41 flights during September 1955. Of this total 23 hours in 31 flights were flown toward the accomplishment of the Convair Development Test Program. The average productive flight time of the four airplanes being flown during September was 5.7 hours per airplane. Convair flew 30 flights and the Air Force flew one.

Flight test activities of special interest were:

(a) Airplane S/N 53-1780. The flight test program has been rearranged. Only one item, Pneumatic Roll Rate Limiter remains to be done.

(b) Airplane S/N 53-1787. Roll coupling test flights were performed with sideslip feedback to test with variations of initial load factor at start of roll and to investigate higher roll rates. A significant decrease in sideslip angle and normal acceleration appears to result. Tests to determine the buffet boundaries of the airplane were flown.

(c) Airplane S/N 53-1788. The captive missile program was completed. After angle of attack calibrations are done, the flight test program for this airplane should be completed.

(d) Airplane S/N 53-1791. Flights were made with an additional extension on the engine inlet duct ramp. This resulted in cessation of buzz. (Major General Albert Boyd made an Air Force evaluation flight. There were no aircraft discrepancies on the flight).

The current test airplane assignment and test facility assignment is as follows:

Serial No.	Location	Assignment
53-1780	Edwards AFB	Standby
53-1787	Edwards AFB	Stability & Control, Reduced Wt.
53-1788	Holloman AFB	Armament
53-1791	Edwards AFB	Phase I
53-1794	Edwards AFB	Instrumentation for test program
53-1798	Edwards AFB	Antenna tests

A summary of Convair flight operations during September is as follows:

Serial Number	Total Hours	Prod. Hours	Number of Flights	Days Flown	Major Ground Work Items
53-1780	0	0	0	0	Standby
53-1787	5.3	5.3	8	7	Modified pilot assist system to conform to latest 1780 A/C config.
53-1788	7.7	7.7	11	9	Armament
53-1791	4.2	3.6	8	6	Duct ramp extensions
53-1794	0	0	0	0	Engine change. Instrumentation
53-1798	11.9	6	14	13	Antenna
	29.1	22.6	41	35	

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FLIGHT TEST HOURS

Airplane S/N	Total	Convair Test	Other Agency	Ferry, Shakedown Familiarization, etc.
* 52-7994	1.8	1.8	0.0	0.0
** 52-7995	105.0	67.2	36.1	1.7
** 53-1779	12.6	9.9	0.8	1.9
53-1780	52.2	45.6	5.3	1.3
** 53-1781	49.4	42.5	0.0	6.9
** 53-1782	21.1	18.4	0.3	2.4
** 53-1783	84.0	23.5	59.8	0.7
53-1787	80.9	54.1	23.1	3.7
53-1788	35.3	33.0	0.9	1.4
53-1791	13.2	11.7	0.6	0.9
*** 53-1794	0.5	0.0	0.0	0.5
53-1798	11.9	4.9	0.0	7.0
Totals	467.9	312.6	126.9	28.4

NUMBER OF FLIGHTS

52-7994	6	6	0	0
52-7995	163	113	46	4
53-1779	24	19	1	4
53-1780	75	66	6	3
53-1781	84	72	0	12
53-1782	29	24	1	4
53-1783	94	35	57	2
53-1787	120	87	28	5
53-1788	60	57	1	2
53-1791	25	23	1	1
53-1794	1	0	0	1
53-1798	14	5	0	9
Totals	695	507	141	47

* Crashed (Tests added to 7995)
** Indicates Convair operations complete.
*** Indicates airplane has not yet entered its Flight Test Program.

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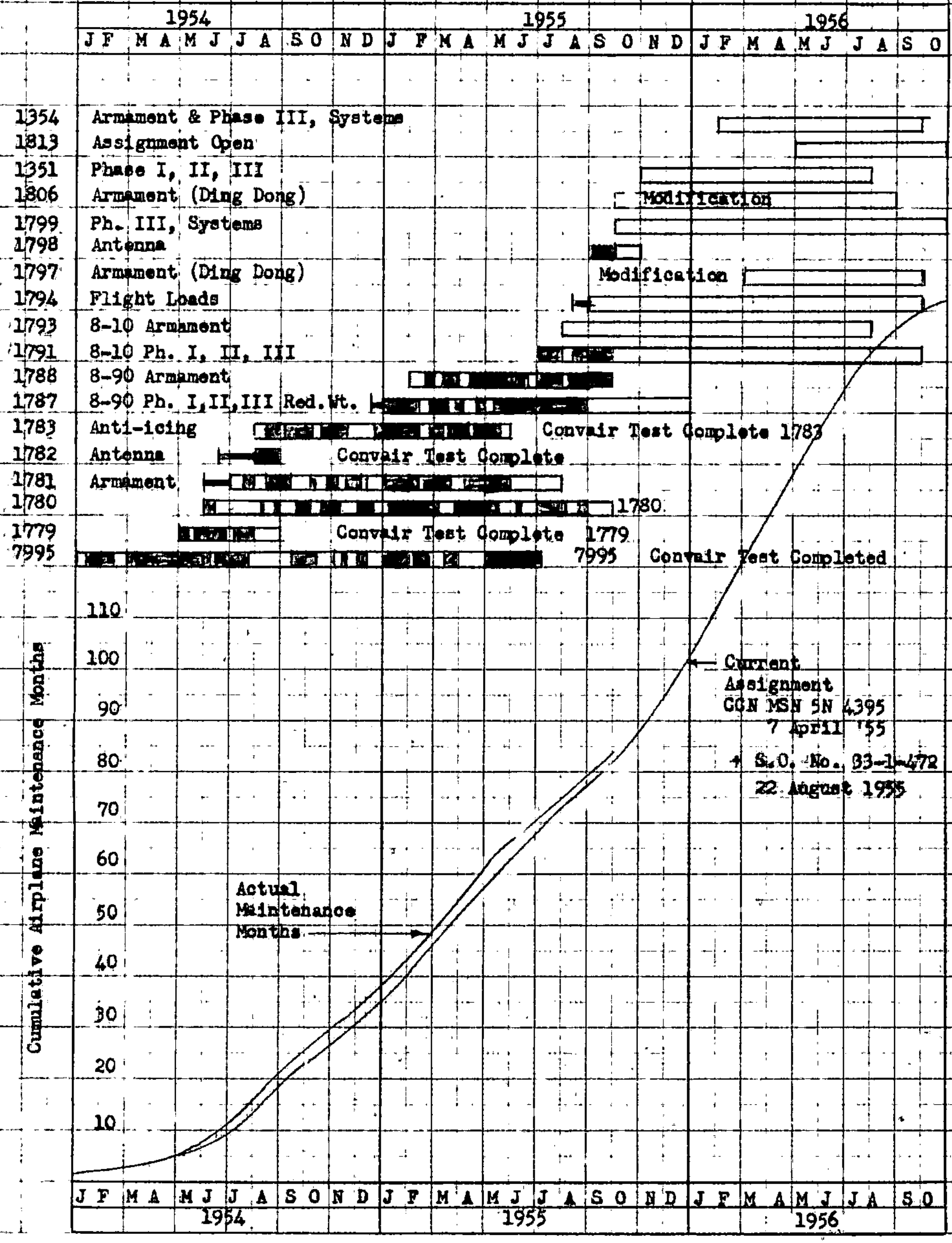
CONVAIR F-102 & TF-102 ACCELERATED DEVELOPMENT PROGRAM
AIRCRAFT ASSIGNMENT SCHEDULE AND ACTIVITY STATUS

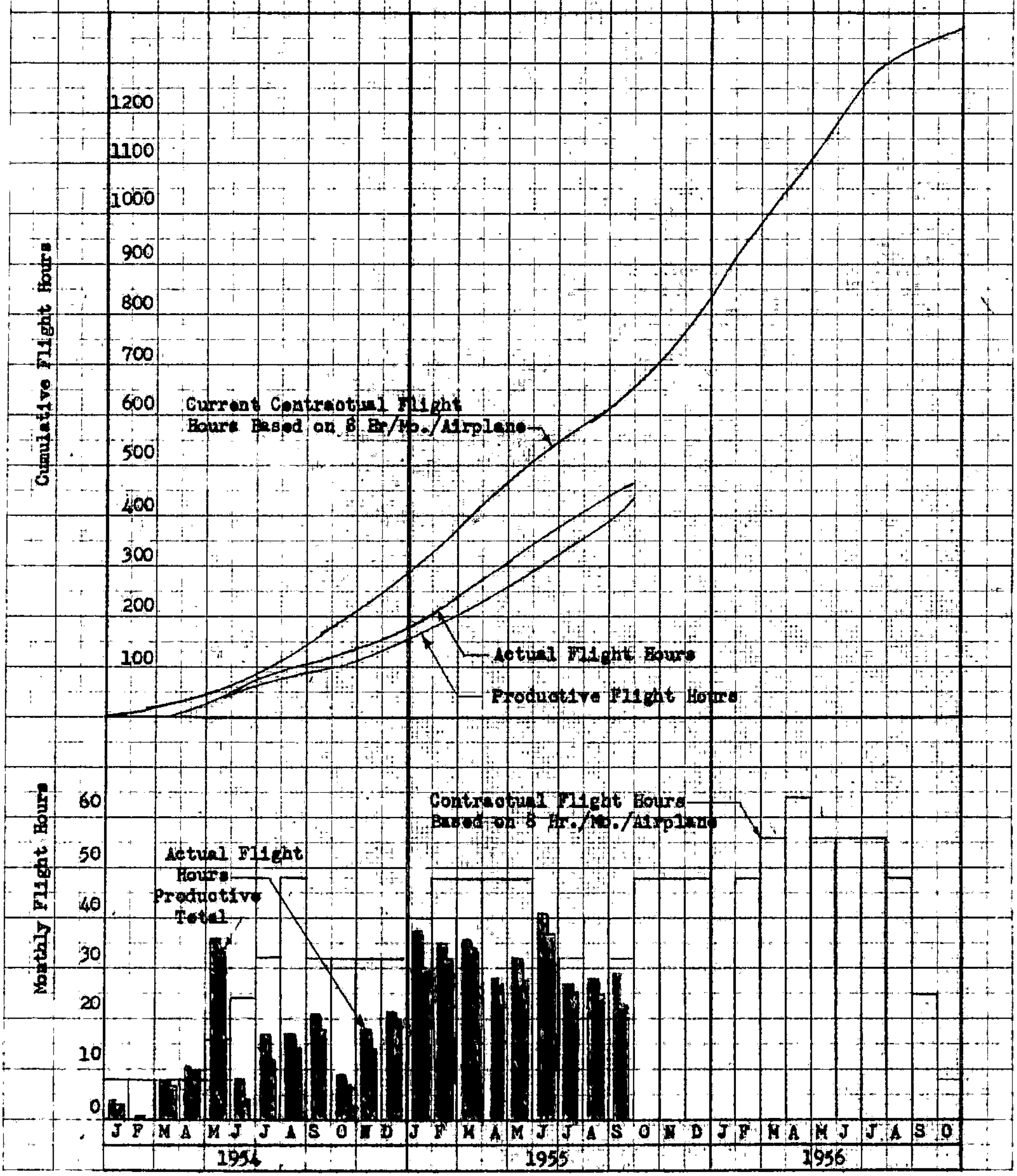
Graph No. 1



Graph No. 1A

CONVAIR F-102 & TF-102 ACCELERATED DEVELOPMENT PROGRAM

ESTIMATED FLIGHT TIME AND FLIGHT HOURS FLOWN



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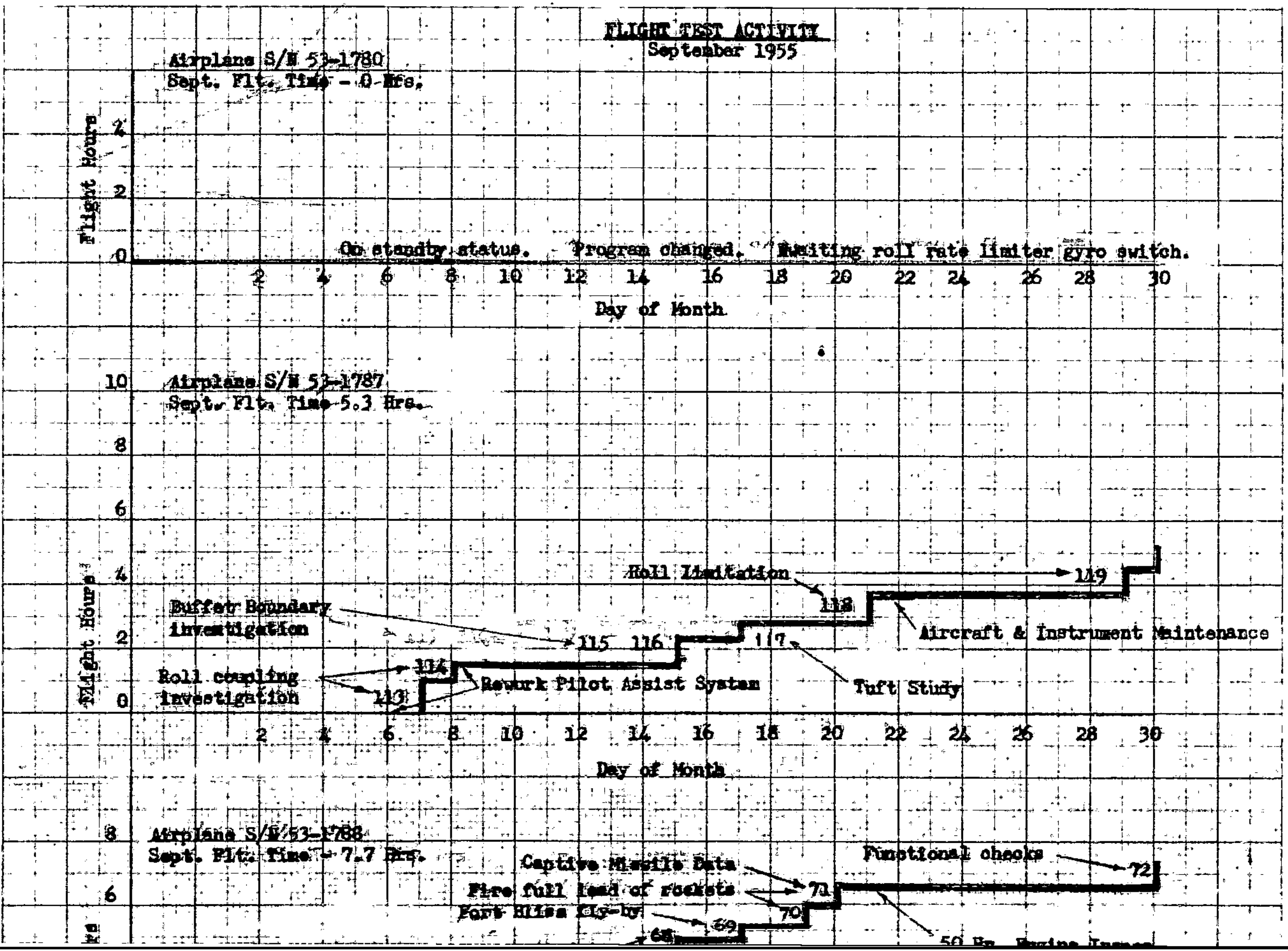
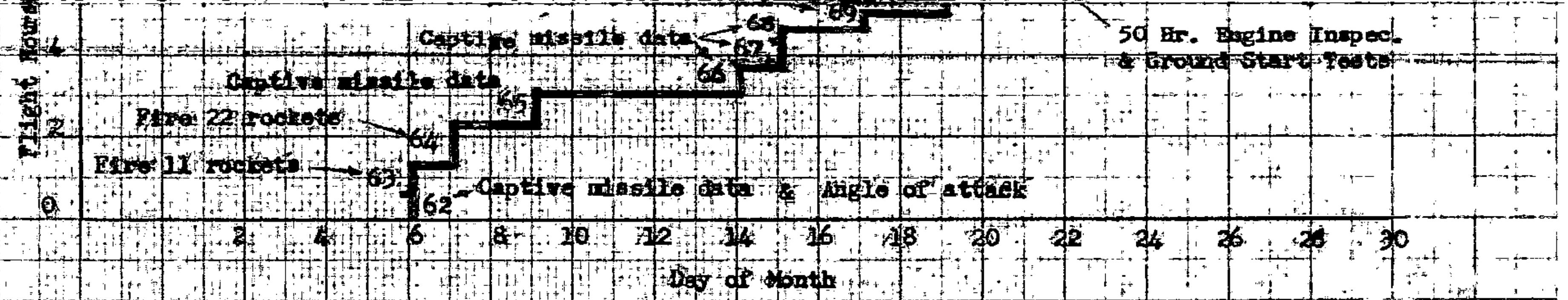
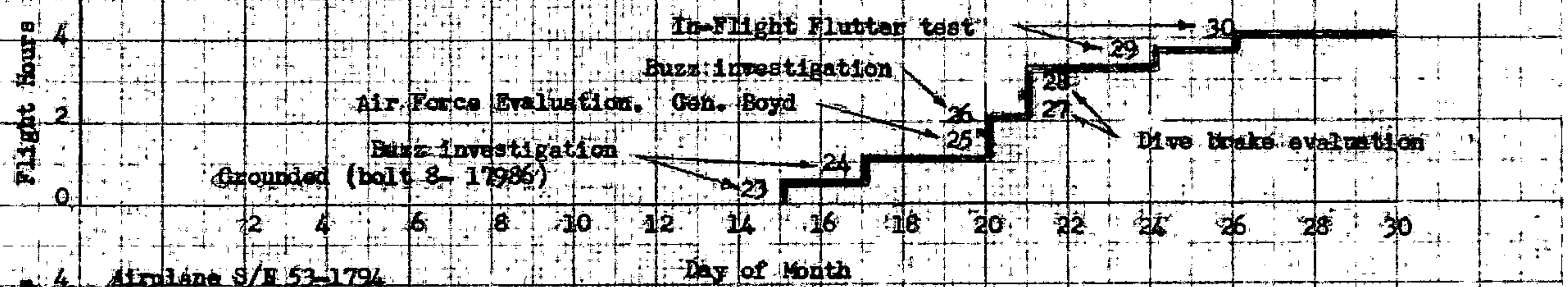


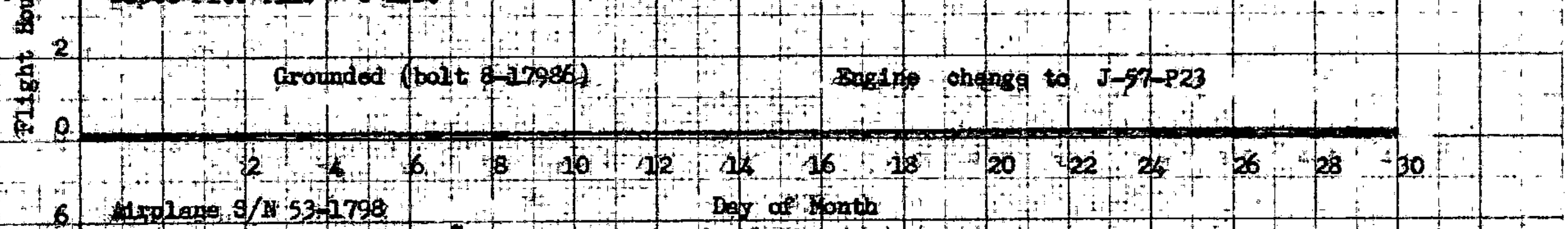
FLIGHT TEST ACTIVITY
September 1955



Airplane S/N 53-1791
Sept. Flt. Time = 4.2 Hrs.

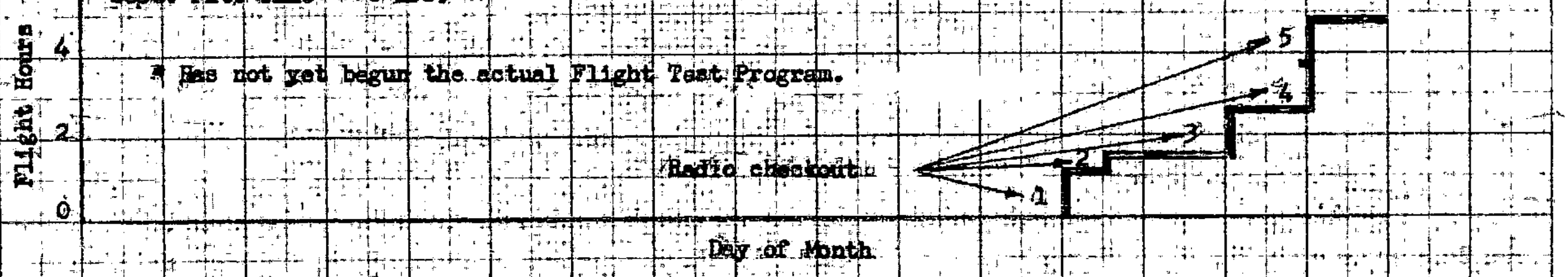


Airplane S/N 53-1794
Sept. Flt. Time = 0 Hrs.



Airplane S/N 53-1798
Sept. Flt. Time = 6 Hrs.*

* Has not yet begun the actual Flight Test Program.



INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1780

Assignment

This airplane is assigned to Convair for a period of 15 months which commenced in July, 1954. Its formal statement of flight test assignment as presented in CCN MSN 5N-4395(159) dated 7 April 1955 is as follows: "To be utilized for Phase III testing of components sub-systems (pneumatic, hydraulic, electrical and control) tests and development." Convair has divided this assignment into seven general flight test stages to accomplish the required flight test operation effectively. Graph No. 4 presents these seven stages as programmed and performed.

Summary of Activity

This airplane has been utilized primarily for testing and refinement of the flight control and automatic flight control systems which will be incorporated in the production version. Dynamic response data has been obtained for development of the Hughes Aircraft Company fire control system.

Results of tests on S/N 53-1780 have contributed greatly to the development of a satisfactory flight control system for the F-102 airplanes. These have proven that the design philosophy in the control system is sound and is developed to the point where the principal problems now are component reliability.

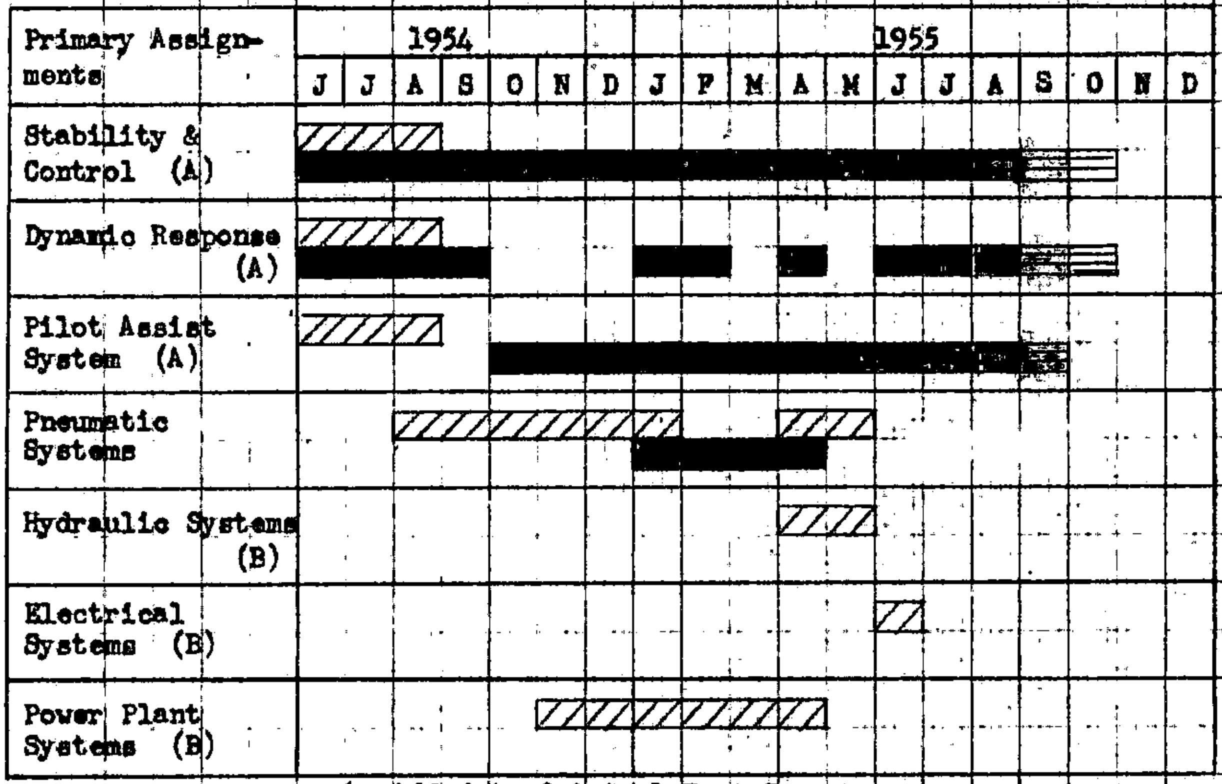
All control tests have been transferred to airplanes S/N 1787 and S/N 1798 except for Pneumatic Roll Rate Limiter tests.

The airplane is on a standby basis pending receipt of Roll Rate Limiter "Gyro" switch.

Summary of Activity

S/N 53-1780

Graph No. 4



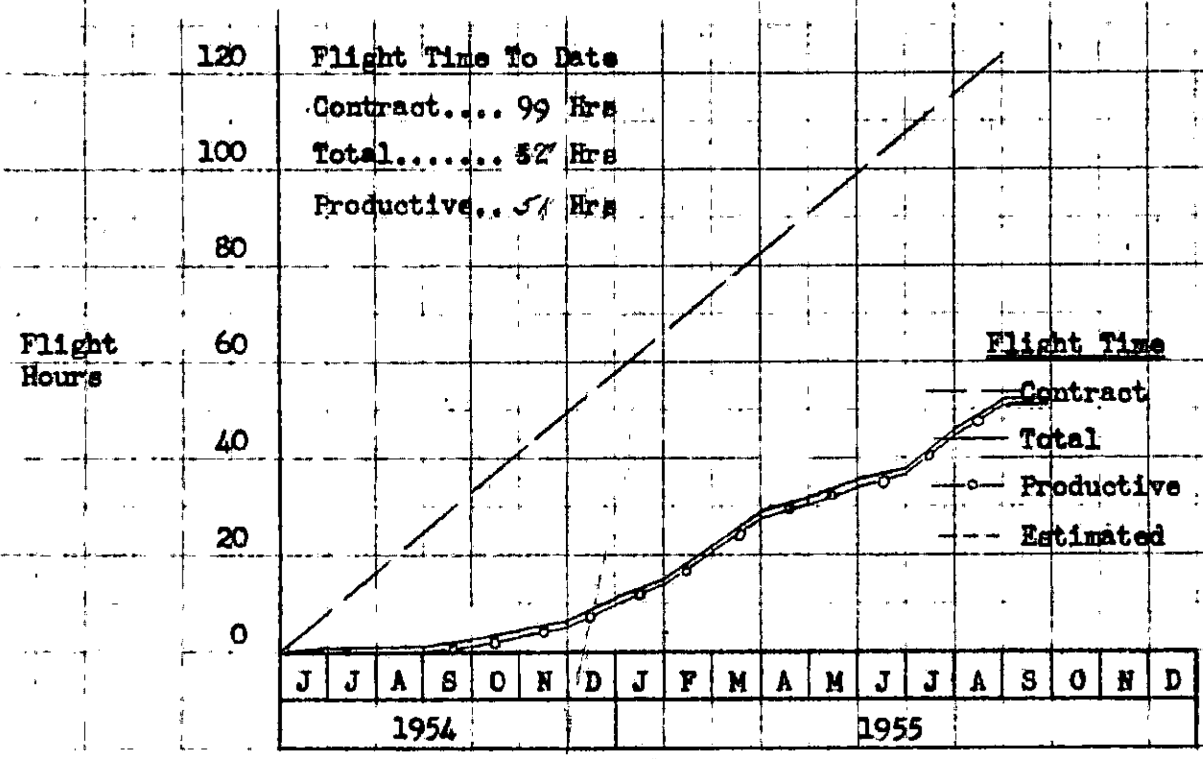
Added to Original Test Program →← →

Convair Report No. ZC-8-111

Original Planning ▨ Activity ■ Future Planning ☰

(A) Test Assignments Transferred from Airplane S/N 53-1779.

(B) Cancelled. Spec. compliance items to be completed on 8-10 airplanes.



Flight Time To Date

Contract.... 99 Hrs

Total....... 52 Hrs

Productive.. 51 Hrs

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1787

Assignment

This airplane is assigned to Convair for a period of 12 months which commenced in January 1955. The formal statement of flight test assignment for this airplane as presented in CCN MSN 5N-4395(159) dated 7 April 1955 is as follows: "Prototype tests of reduced weight aircraft. Phase I, II and III." Convair has divided this assignment into five general flight test stages to accomplish the required flight test operations effectively. Graph 5 presents these five stages as programmed and performed.

Summary of Activity

Phase I performance and preliminary flight characteristics have been accomplished preceded by qualitative evaluation of airplane systems and airspeed calibrations. Phase II testing has been completed. Roll rate data has been obtained to substantiate analog computer studies for the purpose of establishing roll rate limitations.

This airplane has confirmed the performance predictions for the 8-10 airplanes. The manual mode surface control system has been developed to a satisfactory degree and the trim servo provides a stable stick force gradient in the sonic, transonic and supersonic speed ranges tested. Some additional tests may be required to obtain a more desirable stick force gradient.

Much testing on this airplane has been for investigation of high speed vibration or "buzz". Now, the primary "buzz" investigation has been shifted to airplane S/N 53-1791.

Roll testing (inertia coupling) continued for the purpose of expanding the investigation to cover variations of initial load factor at start of roll; and, also, to investigate higher roll rates. These tests are being made with a sideslip signal feed-back to the rudder to obtain augmentation of directional static stability. Preliminary evaluation of data indicates a significant decrease in sideslip angle and normal acceleration during 360° rolls.

Tests to determine the buffet boundaries of the airplane were flown and the data are presently being analyzed.

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1787

Future Activity

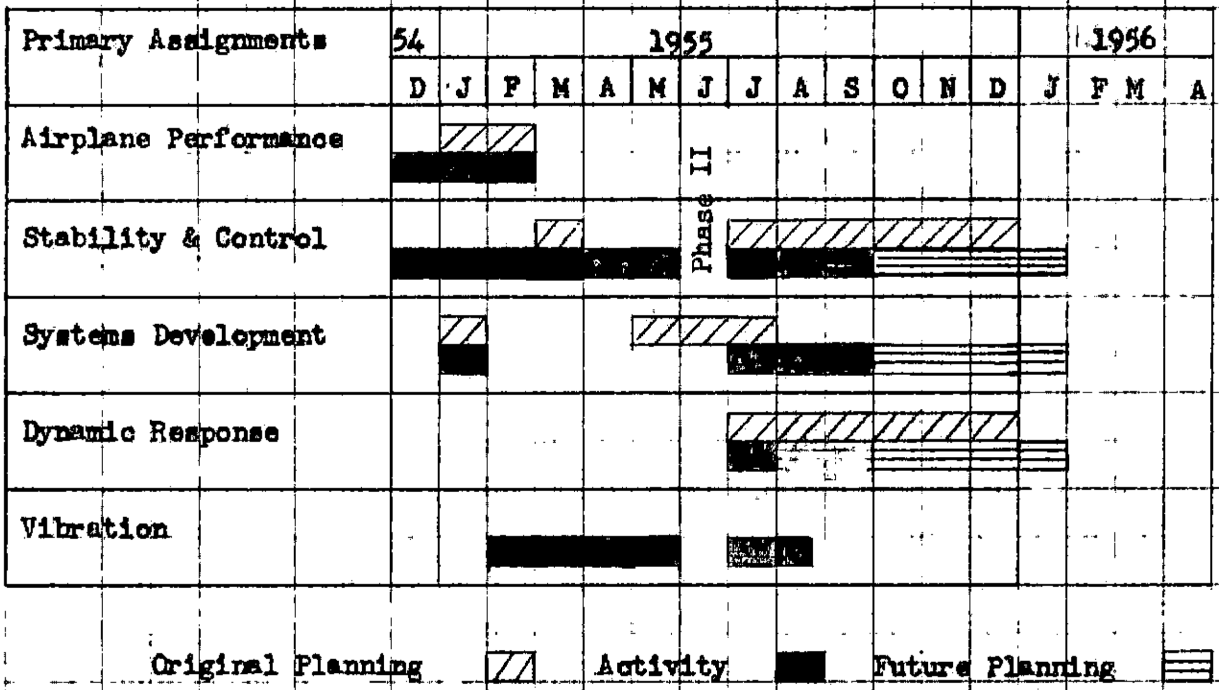
Programs for additional pilot assist system development testing and 8-10 control system component evaluation testing are being prepared. The roll testing (inertia coupling) is expected to continue until a satisfactory solution to the problem is resolved.

A program to establish the buffet boundaries has been assigned for future activity.

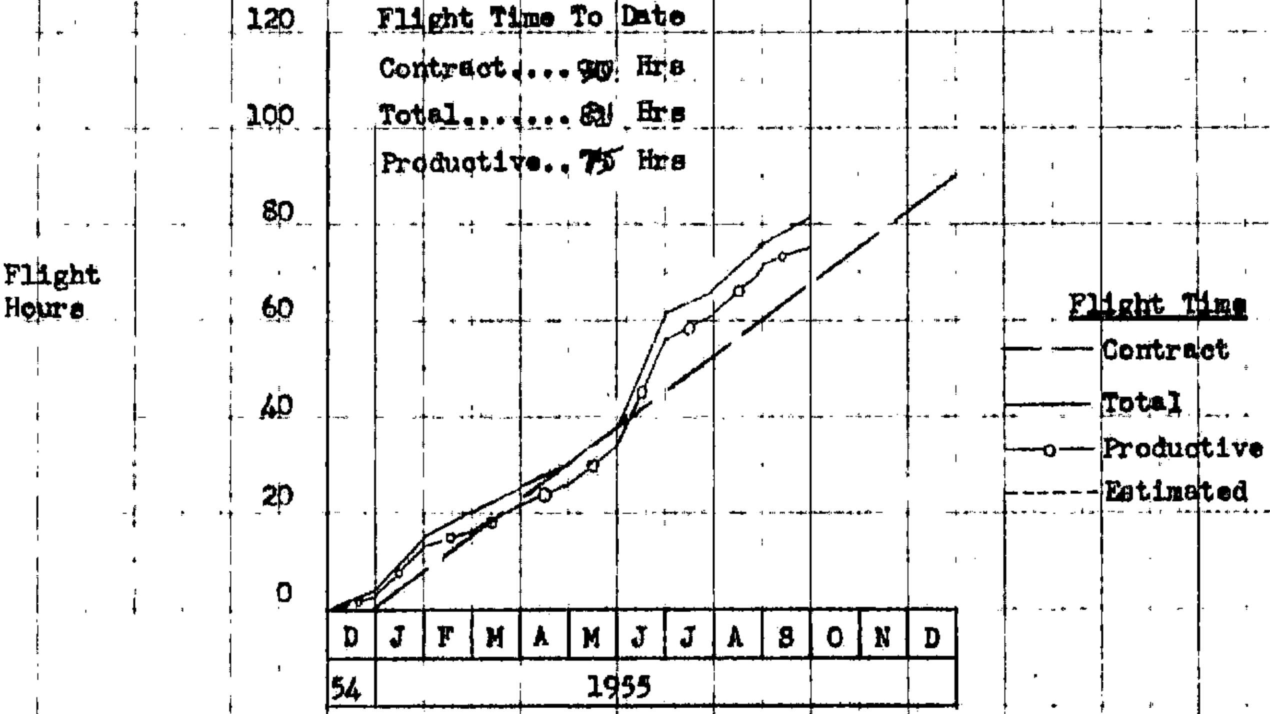
Summary of Activity

S/N 53-1787

Graph No. 5



Convair Report No.
ZC-8-118



Flight Time To Date
Contract.....90 Hrs
Total.......81 Hrs
Productive..75 Hrs

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1788

Assignment

This airplane was assigned to Convair for a period of 5 months commencing in February 1955. Its formal statement of flight test assignment as presented in CCN MSN 5N-4395(159) dated 7 April 1955 is as follows: "To be utilized for subsonic verification of armament tests and supersonic tests and functional development of armaments including hardware and dispersion studies. These tests will be conducted up to combat altitude and maximum speed. Utilize for systems test of MX-1179 after completion of armament tests above." Convair has divided this assignment into six general test stages to accomplish the required flight test operations effectively. Graph No. 6 presents these six stages as programmed and performed.

Summary of Activity

Airplane S/N 53-1788 commenced its flight test program at Holloman AFB on 14 March 1955.

CVAC missile and rocket firing programs are now completed. Missile and rocket separation characteristics were satisfactory. No effect on engine performance was encountered.

Data has been obtained which is to be used to eliminate the present speed restrictions at which the armament mechanism may be used.

Rocket firings for HAC jump angle computation are scheduled for October and November.

Angle of attack calibrations using the inclinometer method are scheduled. This should complete the test program for this airplane.

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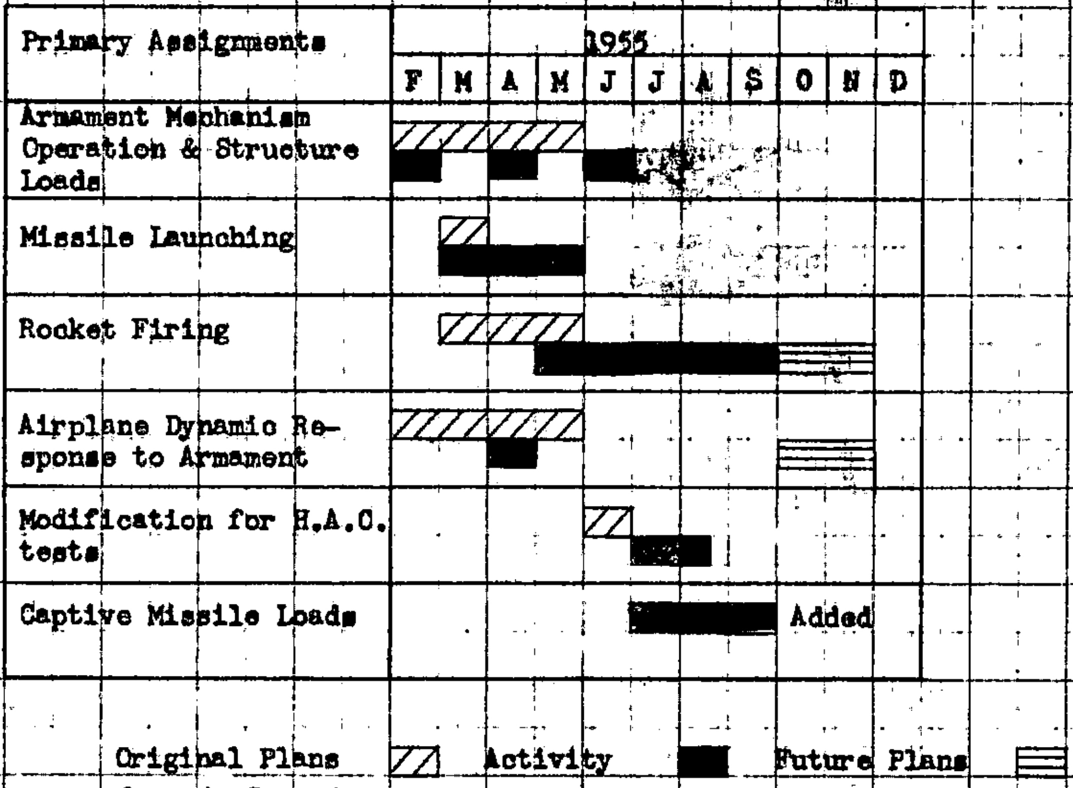
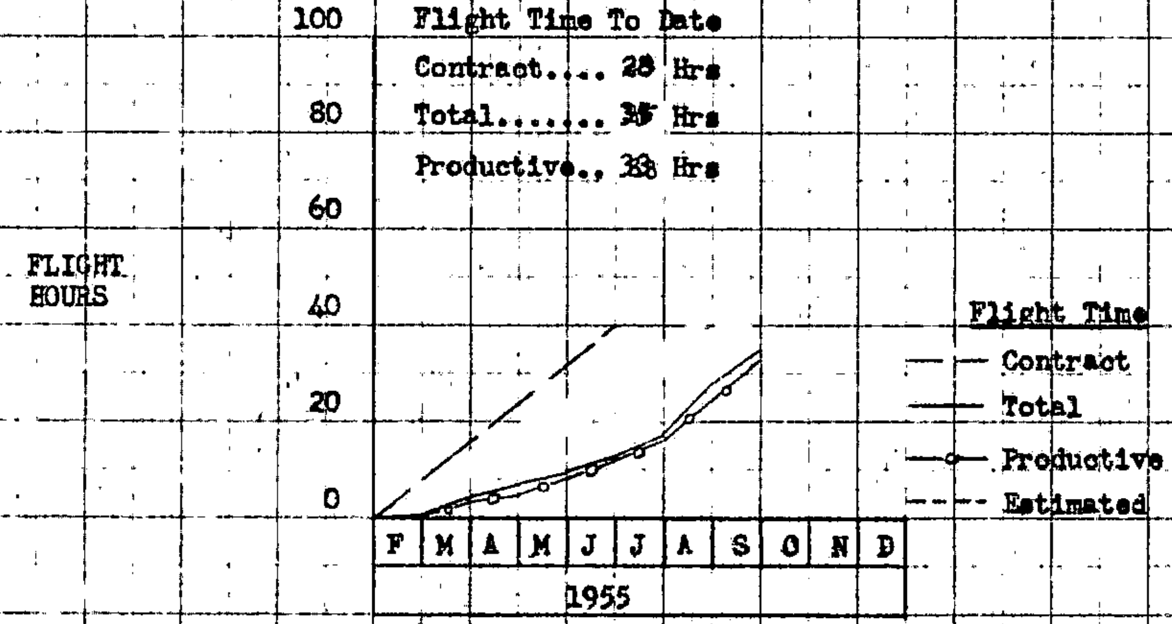
Summary of Activity
S/N 53-1768

Graph No. 6



Convair Report
No. ZC-8-119



Flight Time To Date

Contract.... 28 Hrs

Total....... 35 Hrs

Productive.. 33 Hrs

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1791

Assignment

This airplane is assigned to Convair for a period of 15 months commencing June 1955. Its formal statement of flight test assignment as presented in CCN MSN 5N-4395(159) is: "To be utilized for recheck of Phase I, II and III plus cooling as it applies to first production light weight model." Convair has divided this assignment into seven general flight test stages to accomplish this flight test operation effectively. Graph No. 7 presents these stages as programmed and performed.

Summary of Activity

Airplane S/N 53-1791 arrived at Edwards AFB on 21 June 1955. Test program inflight flutter tests and low speed vibration studies were conducted. The low speed, low frequency vibration study was completed on July 27, when the source of vibration was found to be a regulator valve.

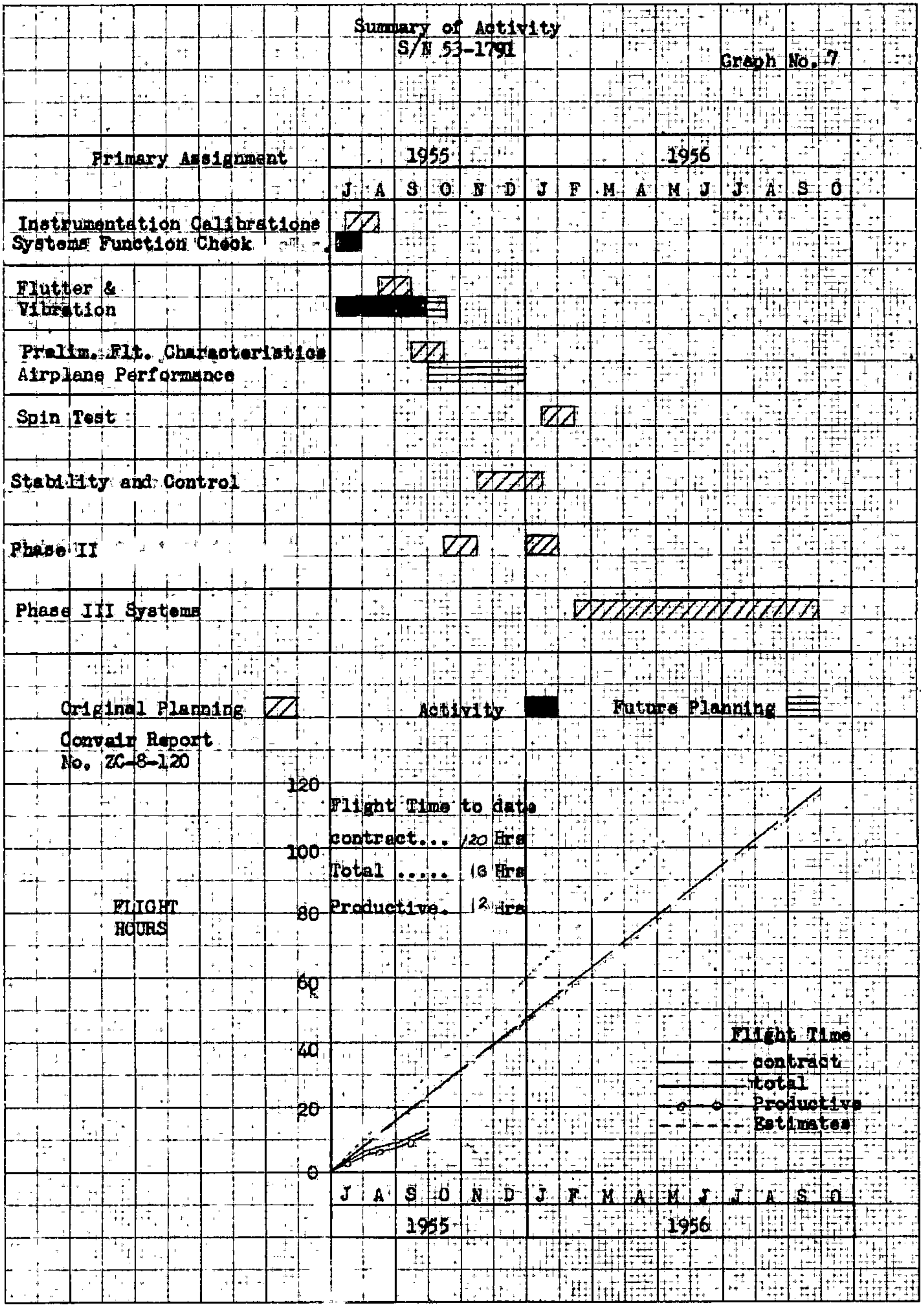
The flight flutter tests were continued through September along with buzz investigations. An additional extension to the engine inlet duct ramp extension resulted in cessation of buzz. Maximum Mach No. has not yet been reached.

(Major General Albert Boyd made an Air Force evaluation flight on September 20th. There were no aircraft discrepancies on this flight.)

Flight tests of flutter and buzz investigations with the longer extension of the inlet ramp are scheduled to be continued through October. Prior to the Phase II flights in the near future, some of the studies to be carried out are for investigation of pilot assist and turn coordination, in flight vibration, interim trim servo system, shelly unit compartment pressures, airspeed calibration and flight flutter tests with external fuel tanks.

Summary of Activity
S/N 53-1791

Graph No. 7



Convair Report
No. ZC-8-120

Flight Time to date
contract... 120 Hrs
Total ..... 16 Hrs
Productive. 12 Hrs

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES

AIRPLANE S/N 53-1794

Assignment

This airplane was assigned to Convair for a period of 13 months. Its formal statement of test assignment as presented in CCN MSN 5N-4395(159) dated 7 April 1955 is as follows: "To be utilized for flight loads determination, operation of high pressure pneumatic system and structural integrity demonstration." Convair divided this program into eight general flight test stages to accomplish the required flight test operation effectively. Graph No. 8 presents the eight stages as programmed and performed.

Summary of Activity

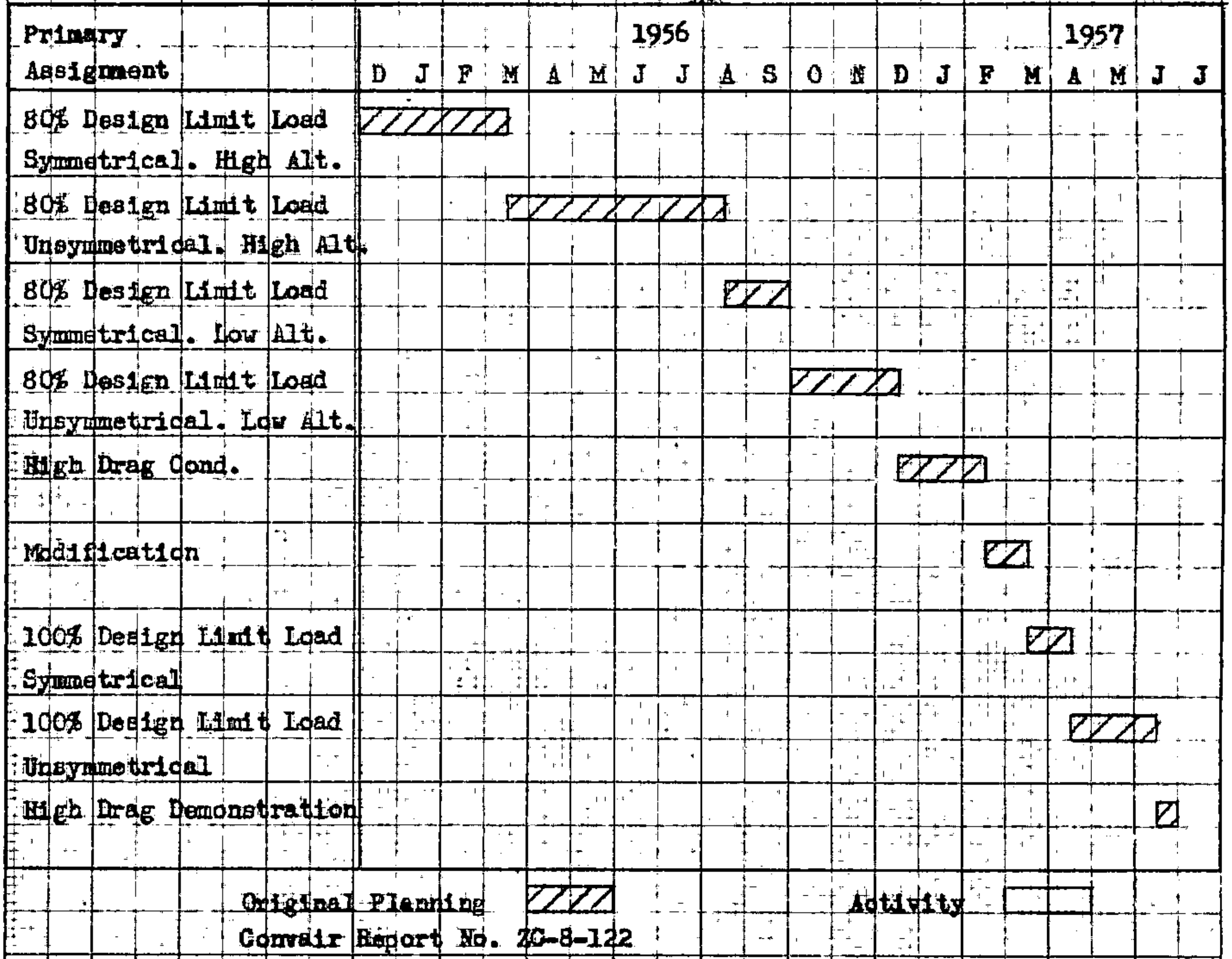
The first shakedown and Air Force acceptance flight of this airplane was completed at Edwards AFB on 25 August 1955.

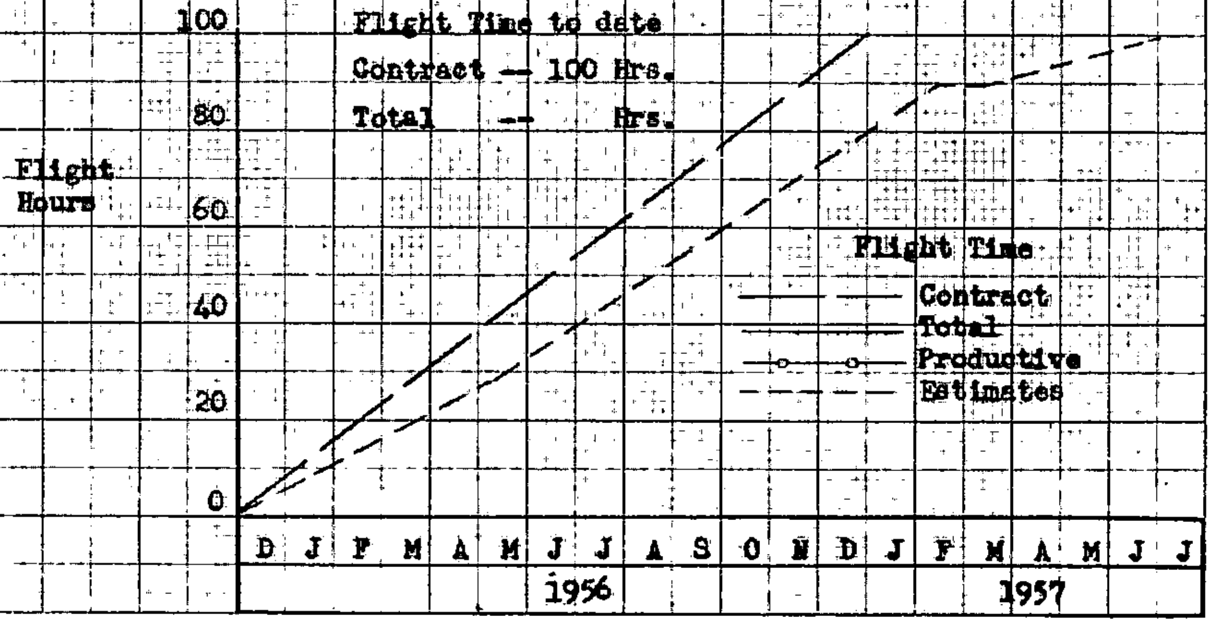
The airplane was on a work status throughout September. Its engine was changed from a J-57-41 to a J-57-P23.

The first flights of the Flight Test Program are scheduled for early December. Prior to that time, installation of test equipment and instrumentation check flights will be performed.

SUMMARY OF ACTIVITY
S/N 53-1794

Graph No. 8





INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES

AIRPLANE S/N 53-1798

Assignment

This airplane was assigned to Convair for a period of one month for antenna tests. Its formal statement of test assignment as presented in CCN MSN 5N-4395(159) dated 7 April 1955 is: "Development and evaluation of flush antenna for fire control, CN and L as well as precipitation static investigation."

The decision has also been made to develop flight controls for an 8-month period following the antenna tests. Air Force is cognizant of this decision, but it is not yet reflected in the contract.

Summary of Activity

This airplane arrived at EAFB on 13 September 1955. The airplane has undergone installation of the flight test instrumentation and special antenna equipment. Parts and instrumentation for pilot assist testing are being installed on a non-interference basis of the antenna program.

Testing of the omni-directional and glide path antennas was initiated on 22 September 1955.

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AIRPLANES CURRENTLY

NOT

ASSIGNED

TO A

TEST PROGRAM

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 52-7994

Assignment

This airplane was assigned to Convair for a period of 18 months which commenced in October 1953. The test assignment of this airplane was to demonstrate and evaluate the basic flying and operating qualities of the YF-102 airplane. This program was authorized by Air Force Contract No. AF33(600)-5942, Supplemental Agreement No. 3. Graph No. 10 presents the six flight test stages that were outlined in Convair Report No. ZC-8-101 and as performed.

Summary of Activity

Airplane S/N 52-7994 began its test program on schedule. No major mechanical malfunctions were noted. A slight modification to the landing gear was made and flutter tests were in progress. On 2 November 1953, on Flight No. 7, the airplane crashed*and was removed from the test program. This flight test program was then transferred to airplane S/N 52-7995, the No. 2 YF-102 airplane.

* Power loss on take-off. Gear-up landing off runway. Convair pilot survived.

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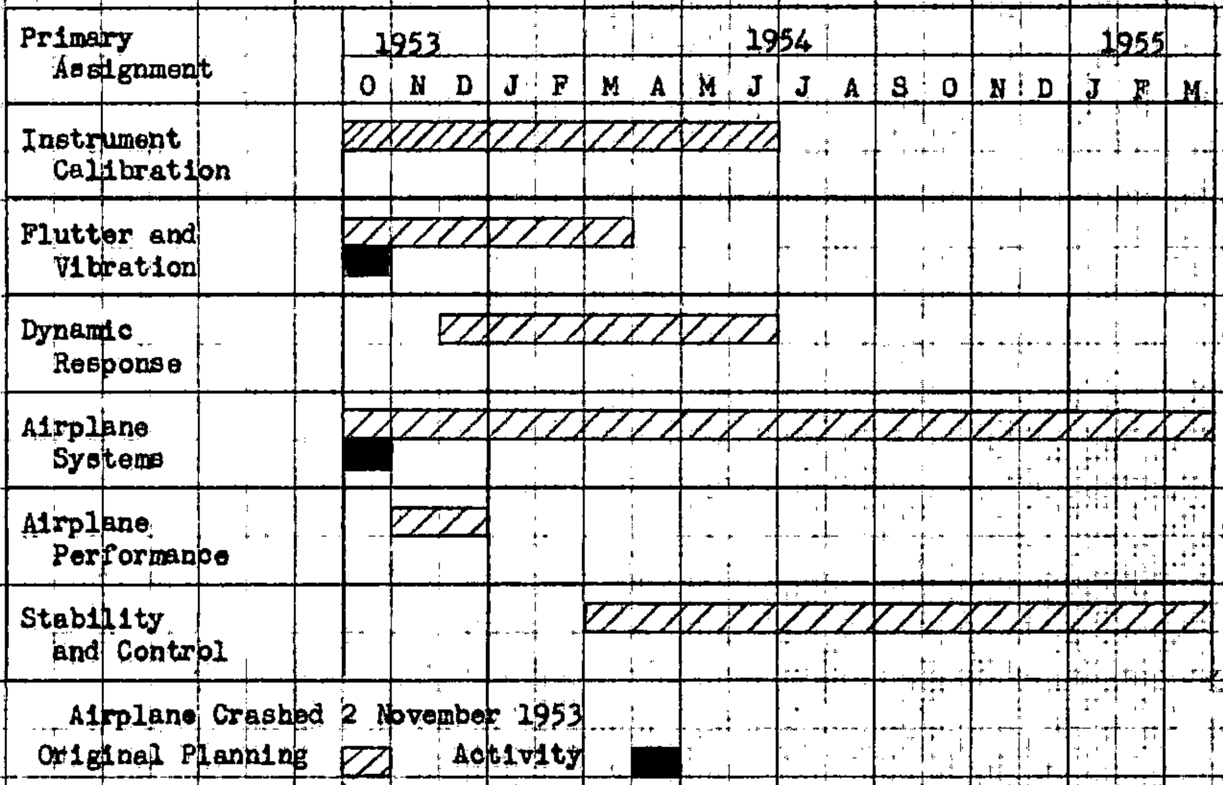
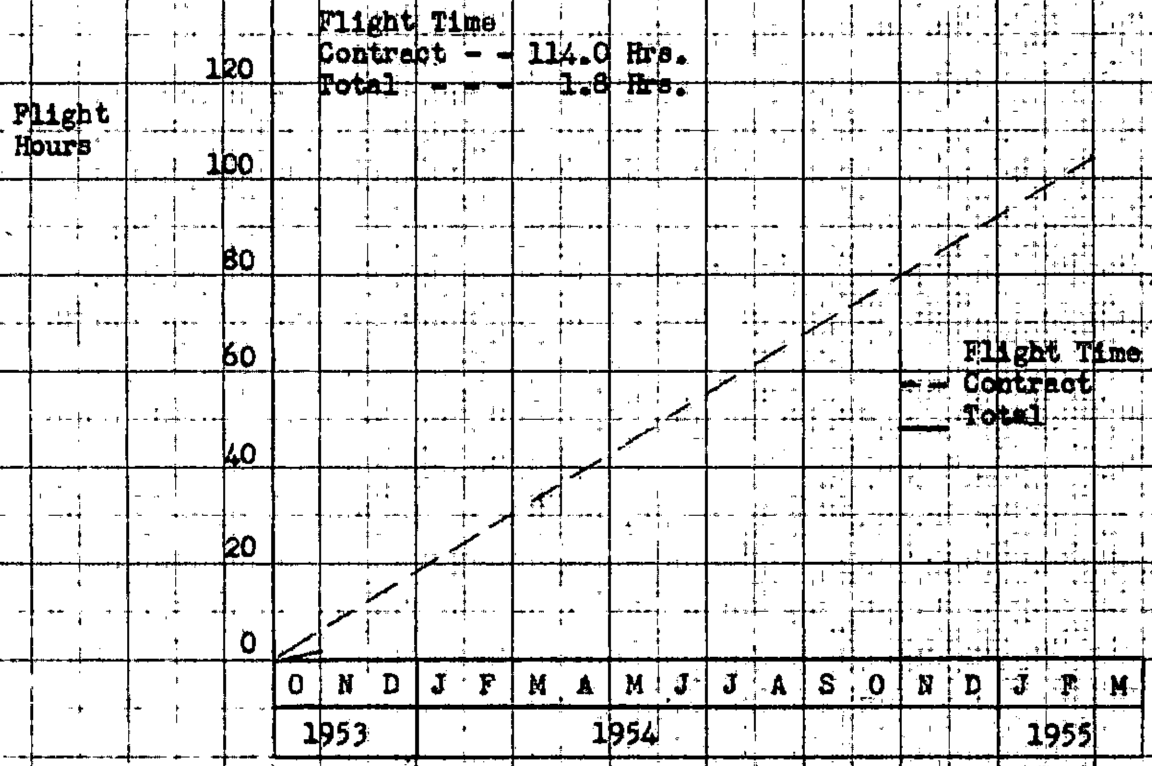
SUMMARY OF ACTIVITY
S/N 52-7994

Graph No. 10





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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES

AIRPLANE S/N 52-7995

Assignment

This airplane was assigned to Convair for a period of 17 months which commenced in January 1954. The formal statement of test assignment for this airplane as presented in CCN MSN 5N-4395(159) dated 7 April 1955, was as follows: "To be utilized for safety of flight tests, initial power plant performance, cooling, pitch and yaw rate damper systems, and pilot relief development. Assume minimum flutter investigation and Phase II, which was not accomplished on No. 1 prototype." Convair divided this assignment into six general flight test stages in order to effectively accomplish the required flight test operation. Graph No. 3 presents the six stages as originally programmed in Convair Report No. ZC-8-103, and as performed.

Summary of Activity.

Preliminary airplane performance and airplane flight characteristics including flutter and vibration tests were completed in the early months of this flight test program. Stability and control tests continued throughout the program as a result of control system changes made to develop the 8-10 versions. Roll rate studies were accomplished in connection with flight characteristics and computer analysis.

This airplane was the second experimental model and performed its early flights with uncambered leading edges and straight wing tips. These were changed to cambered leading edges with reflexed tips and later the tips were changed to full camber without reflex.

Noise levels at crew stations during ground operations, cockpit noise, radome glycol distribution, performance and stability tests, wing fence evaluation, and many development tests of the engine components and systems were conducted during the period of 19 months of field operations and 163 flights.

Contributions to the F-102 program obtained from flight tests of S/N 52-7995 are:

A. Developed "Yellow Canary" tail fairing.
B. Demonstrated cambered leading edge of wings.
C. Obtained information on reflexed and cambered wing tips.
D. Developed extended nose.

Results from these items are incorporated in the 8-10 version of the F-102A airplanes.

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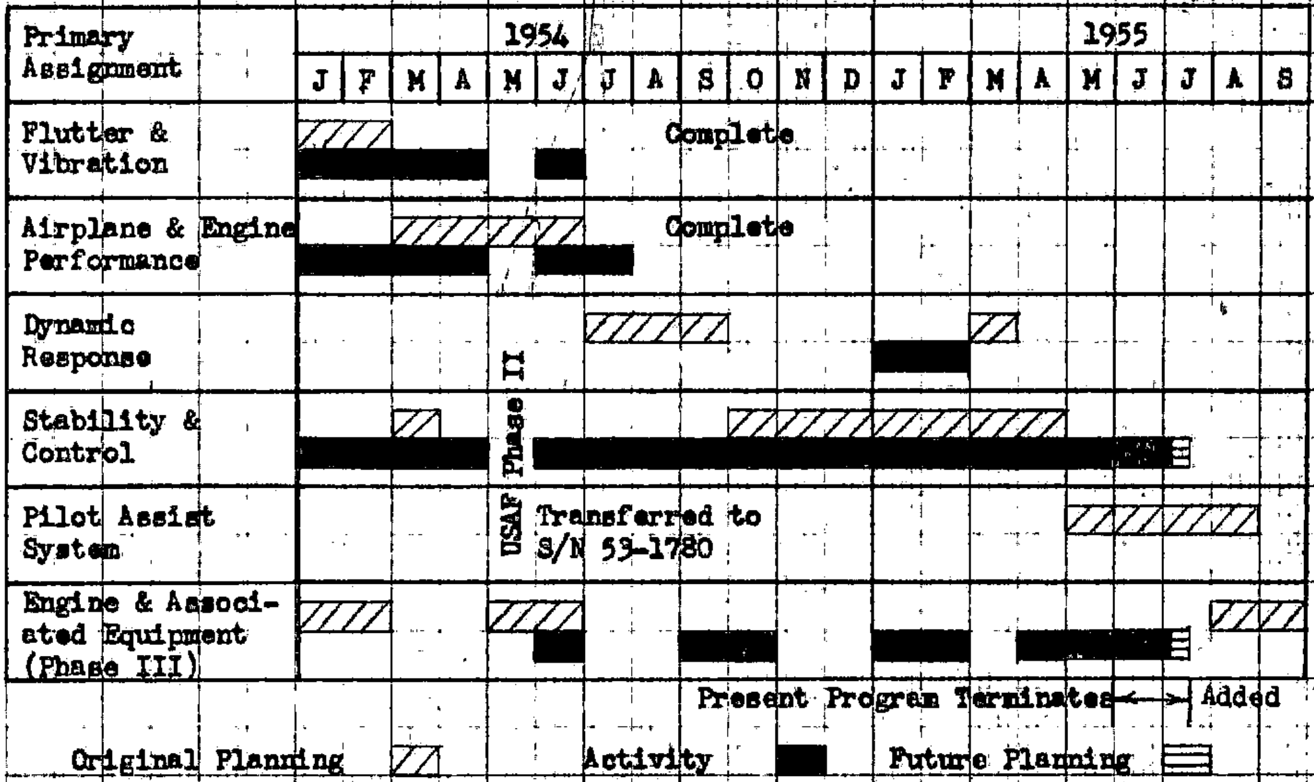
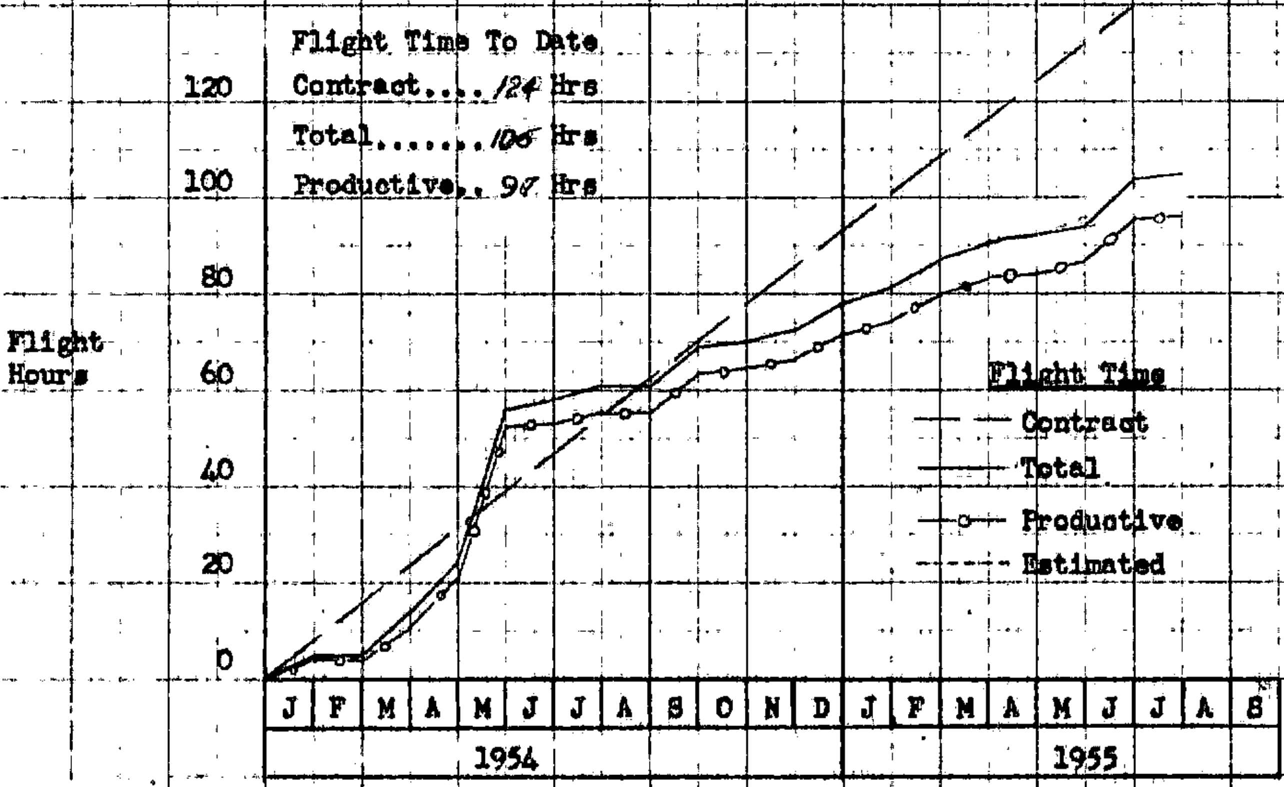
Summary of Activity

S/N 53-7995

Graph No. 3





Flight Time To Date

Contract.... 124 Hrs

Total....... 106 Hrs

Productive.. 97 Hrs

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

Airplane S/N 53-1779

Assignment

This airplane was assigned to Convair for a period of 4 months which commenced in May 1954. The primary purpose of Convair testing was the development testing of the pitch and yaw damper systems. This test program was authorized by Air Force Contract AF33(600)-5942, Supplemental Agreement No. 3. Graph No. 11 presents the three flight test stages which were outlined in Convair Report No. ZC-8-108, and as performed.

Summary of Activity

The first test flight of this program was conducted on 18 May 1954. The flight testing proceeded on schedule with dynamic response and stability and control, as related to the pitch and yaw damper systems, being the prime considerations.

Lateral stability tests proved satisfactory and the optimum yaw damper gain settings were obtained. The pitch damping system in its configuration at the end of the development program did not produce satisfactory damping at all airspeeds at both high and low altitudes with a constant gain setting.

The final flight of this program was conducted on 30 July when this aircraft was delivered to Palmdale, California, for modification prior to delivery to Hughes Aircraft Company.

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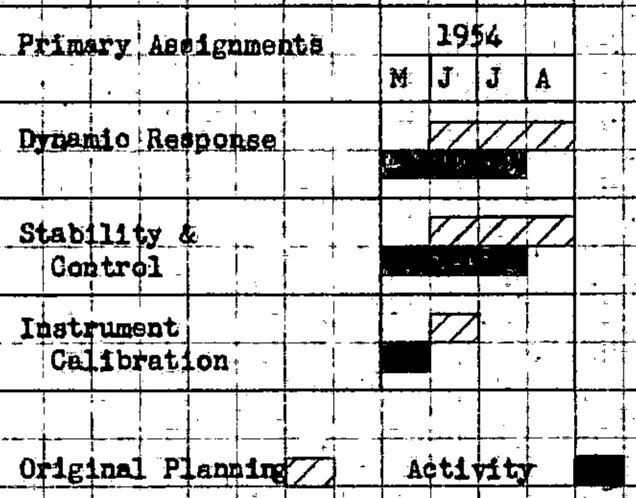
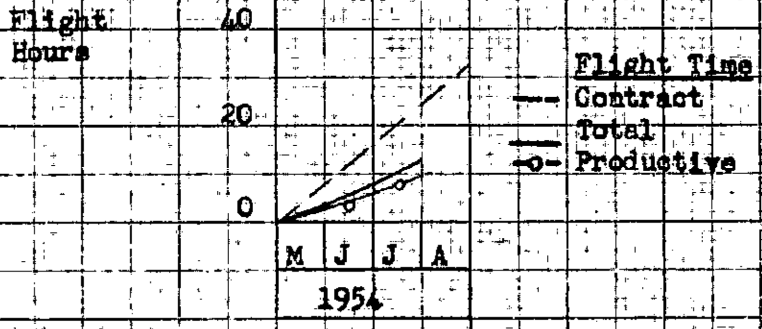
SUMMARY OF ACTIVITY

S/N 1779

Graph No. 11



Flight Time to Date
Contract —— 32.0 Hrs.
Total ——— 12.6 Hrs.
Productive —— 9.7 Hrs.



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

AIRPLANE S/N 53-1781

Assignment

This airplane was assigned to Convair for a period of 14 months which commenced in June 1954. The formal statement of flight test assignment as presented in CCN MSN 5N-4395(159) dated 7 April 1955, is as follows: "To be utilized for Phase III armament firing, functional tests and development, with normal and alternate armament installation." Convair has divided this assignment into five general flight test stages in order to effectively accomplish the required flight test operation. Graph No. 12 presents these stages as originally programmed in Convair Report No. ZC-8-112 and as performed.

Summary of Activity

Armament mechanism operation and structure loads were partially completed in the early part of the test program and later completed in May. The captive missile loads and missile launching program were completed as well as the 2.75 and 2.0 inch rocket programs. The armament tests were generally satisfactory. Flight handling characteristics and stability at combat ceiling with the missile bay doors open was good.

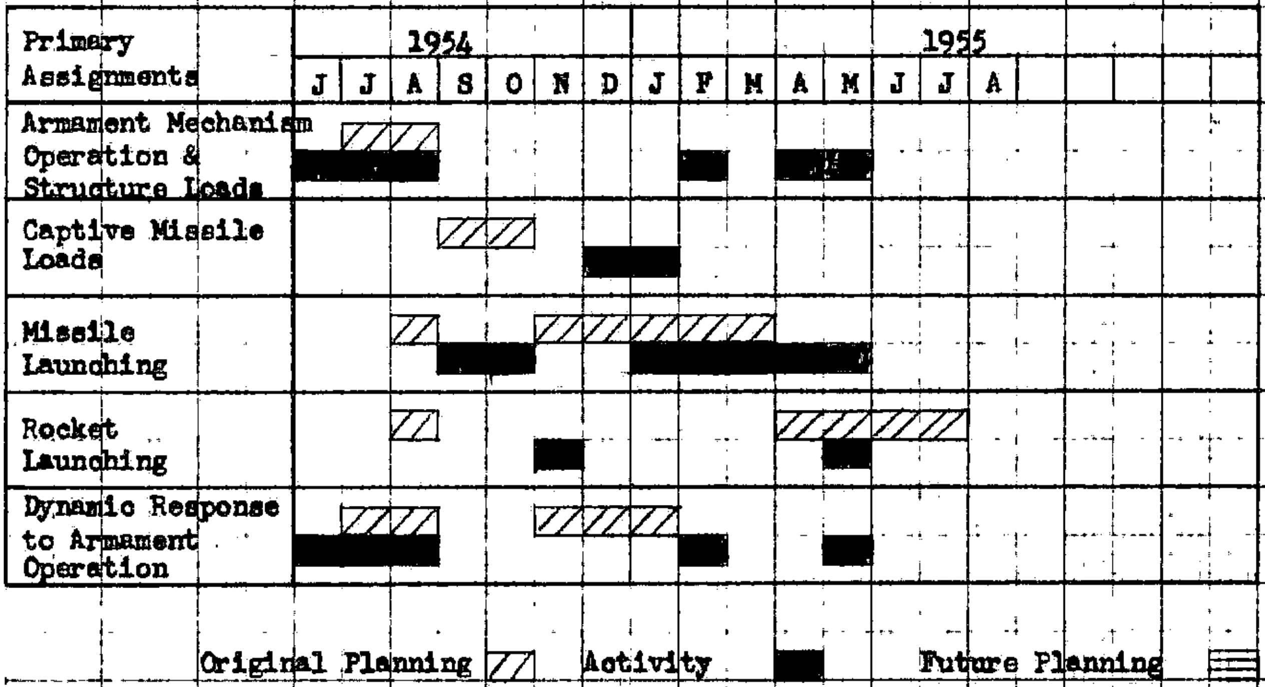
This airplane completed its originally assigned flight test task and is currently being modified at the San Diego Experimental Factory to a Ding Dong launching configuration.

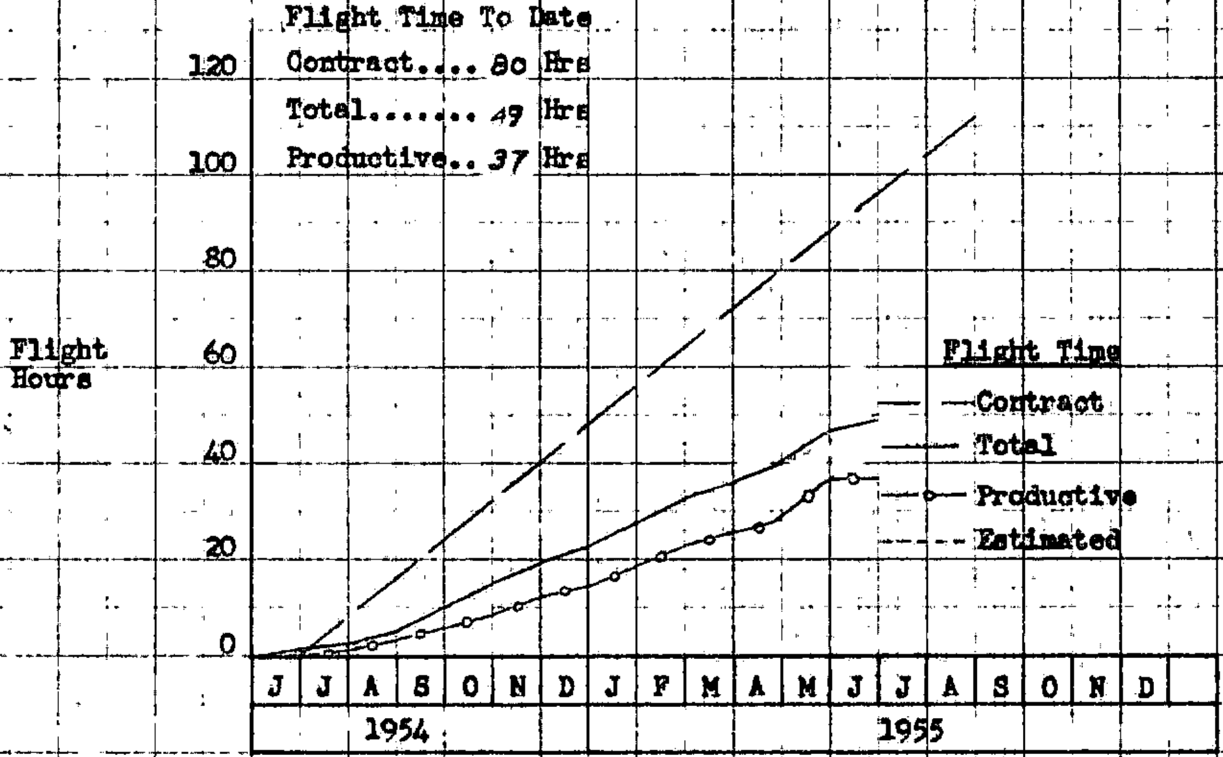
Contributions to the F-102 program obtained from flight tests of S/N 53-1781 are:

A. Information on separation, dispersion and trajectory of rockets and missiles, satisfactory.
B. Airplanes response to rocket and missile firing, satisfactory. Firing full load of rockets at maximum altitude caused no "snuffing".
C. Developed ATM to take care of negative G flight condition.

Summary of Activity **Graph No. 12**

S/N 53-1781




Flight Time To Date

Contract.... 80 Hrs

Total....... 49 Hrs

Productive.. 37 Hrs

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

Airplane S/N 53-1782

Assignment

This airplane was assigned to Convair for a period of one month to demonstrate that the functional characteristics of the production antenna installations were satisfactory for communication and navigation equipment for the F-102 airplane. The testing was to further correlate flight measurements with antenna model studies made at Convair. This testing was authorized by Air Force Contract No. AF33(600)-5942. The testing undertaken on this airplane is shown on Graph No. 11.

Summary of Activity

Although airplane S/N 53-1782 was scheduled for testing in August 1954, testing commenced 28 June 1954. After the initial shakedown flights, instrument calibrating was the first undertaking. Performance, communications and navigation data were obtained in July. Compressor stall investigations were conducted concurrently with the communications and navigation tests. Satisfactory results were obtained from these functional tests of the equipment.

The test program was completed on 18 August 1954, when the aircraft was delivered to Palmdale, California for modification prior to delivery to Hughes Aircraft Company.

Contributions to the F-102 program obtained from flight tests of S/N 53-1782 are:

A. Determined that all antennas are compatible.
B. Completed part of the F-102 antenna specification compliance testing.
C. Demonstrated operation of the emergency fuel system, ring type dacron drag chute and electrical system power controller.

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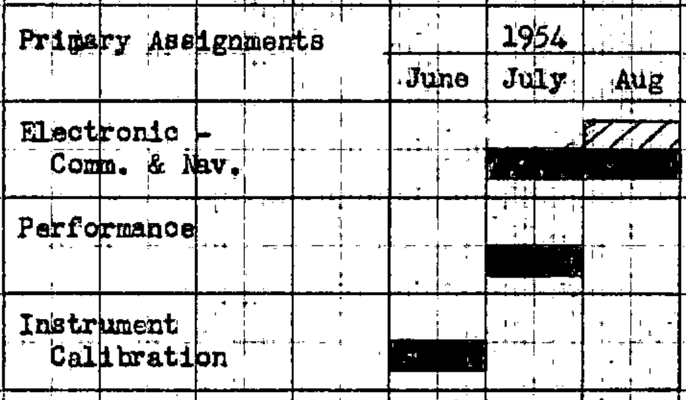
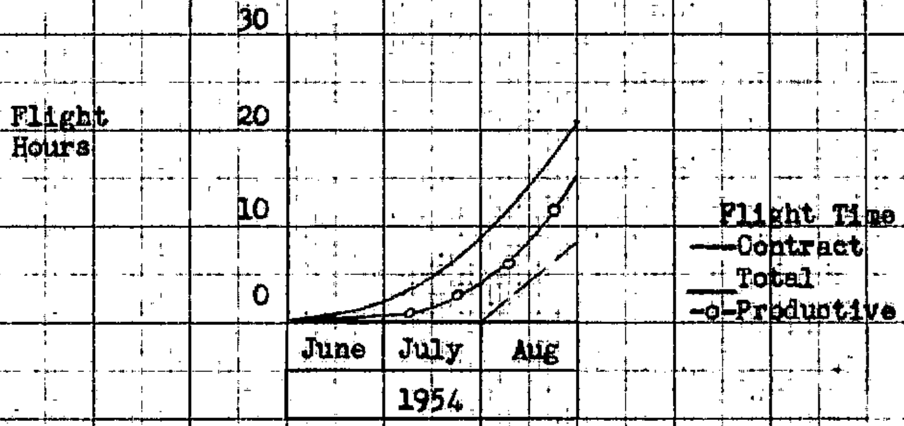
SUMMARY OF ACTIVITY
S/N 1782

Graph No. 11



Original Planning ▨ Activity ■

Flight Time to Date
Contract ---- 8 Hrs.
Total ------ 21.1 Hrs.
Productive-- 15.7 Hrs.



CLEARPRINT PAPER CO. NO. 73B 20 X 20 DIVISIONS PER INCH 190 X 200 DIVISIONS

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

Airplane S/N 53-1783

Assignment

This airplane is assigned to Convair for a period of 10 months which commenced in August 1954. The formal statement of flight test assignment for this airplane as presented in CCN MSN 5N-4395(159) dated 7 April 1955, is as follows: "To be utilized for Phase III tests to include cooling problems. ATM refrigeration unit, cockpit air-conditioning, and correction of deficiencies disclosed during Phase I and II. The aircraft may also be used for Pilot Assist System development in event of difficulties, with airplane S/N 52-7995." Convair has divided this assignment into seven general flight test stages in order to effectively accomplish the required flight test operations. Graph No. 12 presents these six stages as originally programmed in Convair Report ZC-8-114, the periods of flight test activity on each stage, the future periods of activity currently planned on each stage, and the flight time accumulated to date.

Summary of Activity

The original flight test program for S/N 53-1783 was changed considerably. Instrumentation calibration, production items, engine and structural cooling, and pilot assist phases of the test program were deleted to permit delivery of this aircraft to WADC in January 1955. Dry air anti-icing, defogging, and cabin ventilation and pressurization tests were limited to qualitative testing to permit delivery to WADC in January 1955.

An added task of communications and navigation systems testing was completed during the second month of the program. These tests were transferred from S/N 1782.

Air Force pilots evaluated S/N 53-1783 in various icing conditions. In numerous flights, artificial icing conditions were employed behind tanker type aircraft. Preliminary information indicated satisfactory performance of this airplane in the icing conditions encountered. The anti-icing tests were completed on 3 May with Flight No. 82. As a result of the tests in icing conditions without wing and tail anti-icing, the anti-icing equipment has been deleted from the 8-10 airplane with a considerable saving in weight.

S/N 53-1783 was ferried from WADC to EAFB and arrived at Edwards on 15 May. It is now in a standby status at Palmdale, California.

Contributions to the F-102 program obtained from flight tests of S/N 53-1783 are:

A. Determined that the F-102 does not require wing and tail anti-icing.
B. Completed antenna specification compliance testing.
C. Demonstrated instrument flying characteristics.
D. Demonstrated night flying characteristics.

Summary of Activity
S/N 53-1783

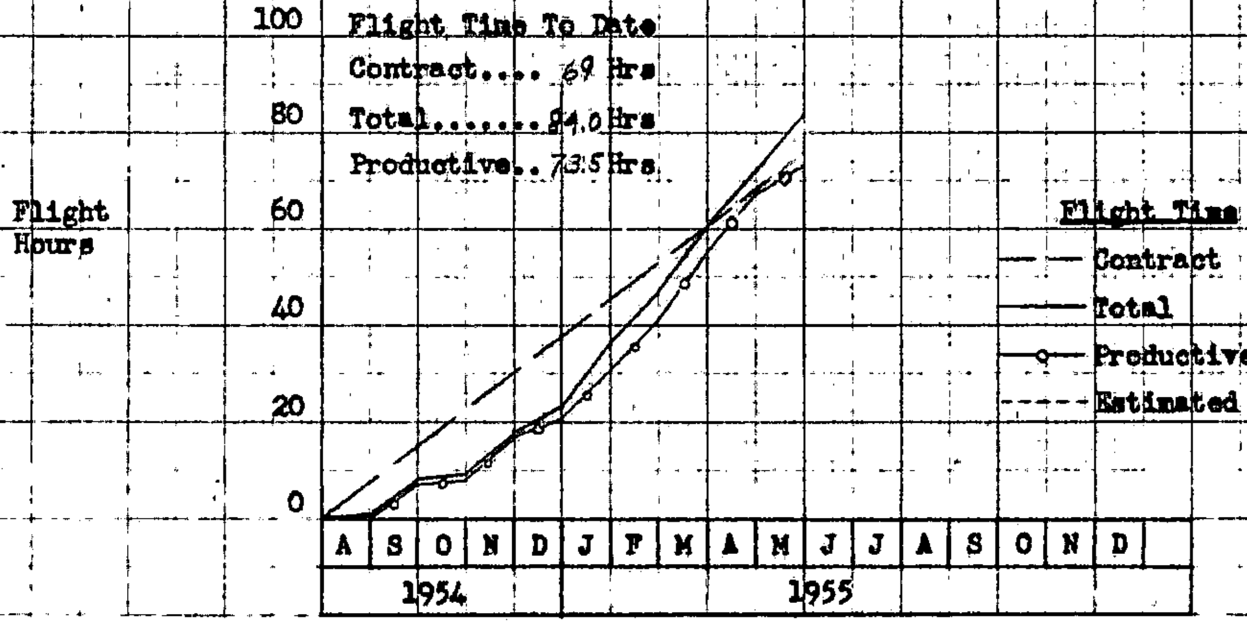
Graph No. 12

Primary Assignments	1954					1955											
	A	S	O	N	D	J	F	M	A	M	J	J	A				
Instrumentation Calibration Production Items	▨			Deleted (A)													
Dry Air Anti-icing, defogging, cabin ventilation & pressurization	▨	▨	▨ ■	▨ ■	▨ ■				■	■			Deleted (B)				
Engine & structural cooling & cooling drag						▨	▨	▨					Deleted (A)				
Dynamic Response			Deleted (A)						▨	▨	▨						
Pilot Assist System			Deleted (A)									▨	▨	▨			
Natural Icing Flights - WADC		Added (C)				■	■	■	■	☰							
Communications & Navigation Systems		■	■			Added (D)											

Convair Proposes Termination ←

Original Planning ▨ Activity ■ Future Planning ☰

(A) Deleted to permit delivery of A/C to WADC Jan. 1955.
(B) Wing & tail anti-icing, cabin pressurization deleted. Cabin vent & defogging limited to qualitative tests to permit delivery to WADC Jan. 1955.
(C) Added to original test program - tests are being flown by WADC personnel.
(D) Transferred from Airplane S/N 53-1782.



Flight Time To Date
Contract.... 69 Hrs
Total....... 84.0 Hrs
Productive.. 73.5 Hrs

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DEFINITIONS

For purposes of this report, the following definitions apply.

(1) Contract Flight Time - Based on 8 Hrs/maintenance month.

(2) Total Flight Time - Airplane total flight time during the period the airplane is assigned to Convair for Flight Testing.

(3) Productive Flight Time - Flight time during which data in furtherance of the development test program was obtained. Air Force Phase II testing is included in this time.

(4) Anticipated Flight Time - Estimate of flight time to be conducted during the coming month.

(5) Convair Test - Flights in which the aircraft has taken off with a Convair pilot, in direct pursuit of fulfilling a portion of the scheduled test program (includes aborts).

Classification Changed to: UNCLASSIFIED

Authorized by: DD 254 Date: 6/16/60

Declassified by: ICB Dept:11 Date: 6/21/60

[redacted]

[redacted]

File
V.Q.

272

COPY NO. 12

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Classification Changed to UNCLASSIFIED
Authorized b : DB 254 Date: 6/10/60
Reclassified by: FCB Dept: 11 Date: 8/11/60

TITLE

F-102 ACCELERATED DEVELOPMENT
TEST PROGRAM
SUMMARY
OF
FLIGHT TEST ACTIVITIES

CONVAIR
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

[redacted]

C O N V A I R
A DIVISION OF GENERAL DYNAMICS CORPORATION
SAN DIEGO

REPORT ZC-8-142
DATE 2 Sept 1955
MODEL F-102

TITLE

F-102 ACCELERATED DEVELOPMENT TEST PROGRAM SUMMARY OF FLIGHT TEST ACTIVITIES

PREPARED BY J. F. Tomaschek

GROUP Engineering Flight Test

REFERENCE

CHECKED BY N W Reed
For: T. L. Maloy

APPROVED BY
for P. M. Prophett

NO. OF PAGES

NO. OF DIAGRAMS

REVISIONS

NO.	DATE	BY	CHANGE	PAGES AFFECTED

INTRODUCTION

This summary report is presented to show the progress to date of the Accelerated Development Test Program for the F-102 airplane. Contractual commitments are compared with actual accomplishments on those airplanes currently being flight tested by Convair.

SUMMARY OF FLIGHT TEST ACTIVITY

The F-102 Accelerated Development Test Program flight operations conducted by Convair are proceeding according to schedule. Convair has completed testing on five of the 17 airplanes, which will eventually be assigned to Convair for flight operations. To date, Convair test airplanes have been flown a total of 439 hours during 654 flights.

The periods of individual airplane flight test utilization and a summary of total flight time is presented in Graphs No. 1 and 1A. A summary of the test program and flight time to date by individual airplanes is presented in Graphs No. 3 through 12. A summary of flight time and the general purpose of the flights accomplished to date on all Convair test airplanes is presented in the following table:

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FLIGHT TEST HOURS

Airplane S/N	Total	Convair	Other Agency	Ferry, Shakedown Familiarization, etc.
* 52-7994	1.8	1.8	0.0	0.0
* 52-7995	105.0	67.2	36.1	1.7
* 53-1779	12.6	9.9	0.8	1.9
53-1780	52.2	45.6	5.3	1.3
* 53-1781	49.4	42.5	0.0	6.9
* 53-1782	21.1	18.4	0.3	2.4
* 53-1783	84.0	23.5	59.8	0.7
53-1787	75.6	48.8	23.1	3.7
53-1788	27.6	25.3	0.9	1.4
53-1791	9.0	8.1	0.0	0.9
** 53-1794	0.5	0.0	0.0	0.5
Totals	438.8	291.1	126.3	21.4

NUMBER OF FLIGHTS

52-7994	6	6	0	0
52-7995	163	113	46	4
53-1779	24	19	1	4
53-1780	75	66	6	3
53-1781	84	72	0	12
53-1782	29	24	1	4
53-1783	94	35	57	2
53-1787	112	79	28	5
53-1788	49	46	1	2
53-1791	17	16	0	1
53-1794	1	0	0	1
Totals	654	486	140	38

* Indicates Convair operations complete, (52-7995 replaced 52-7994).
** Indicates airplane has not yet entered its Flight Test Program.

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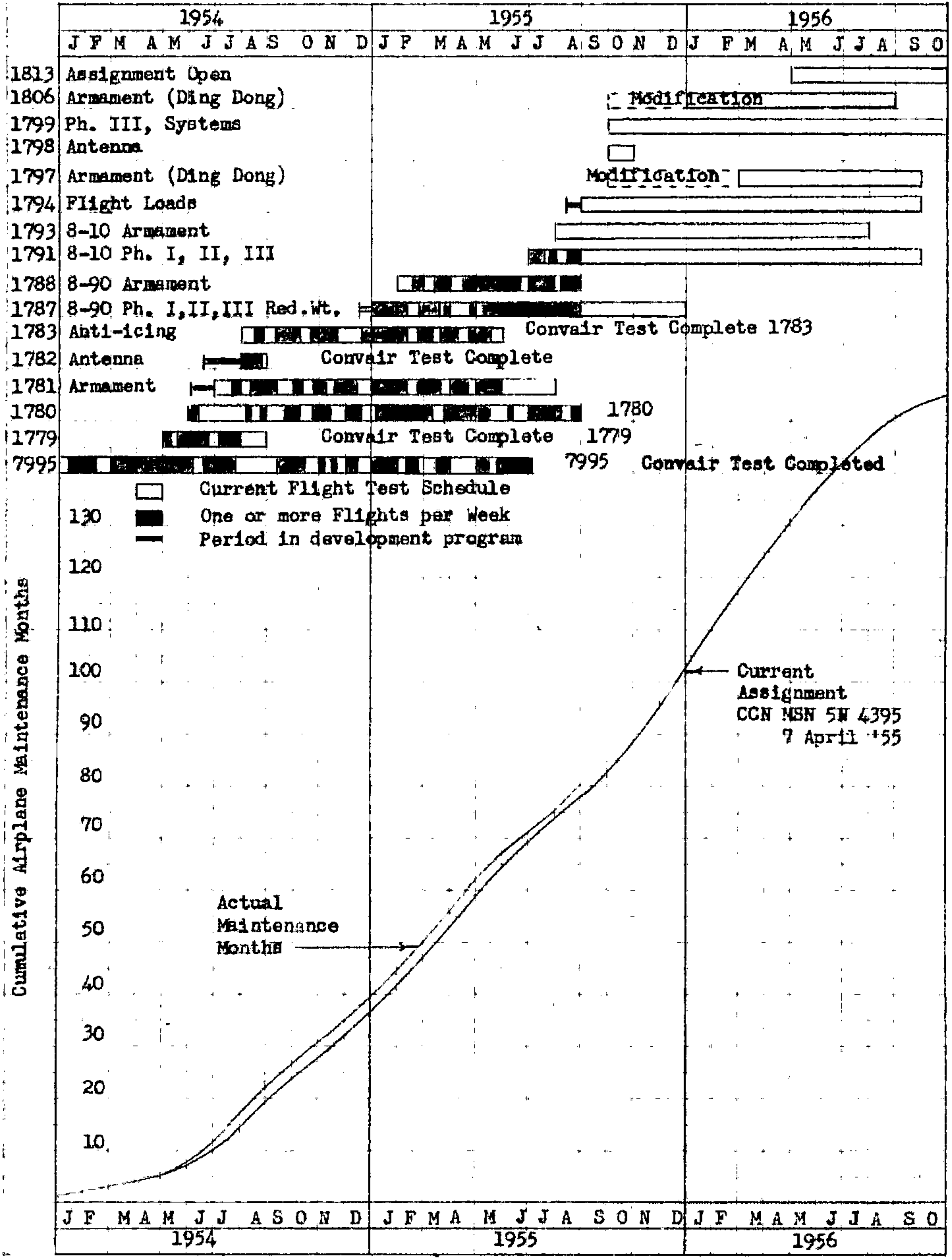
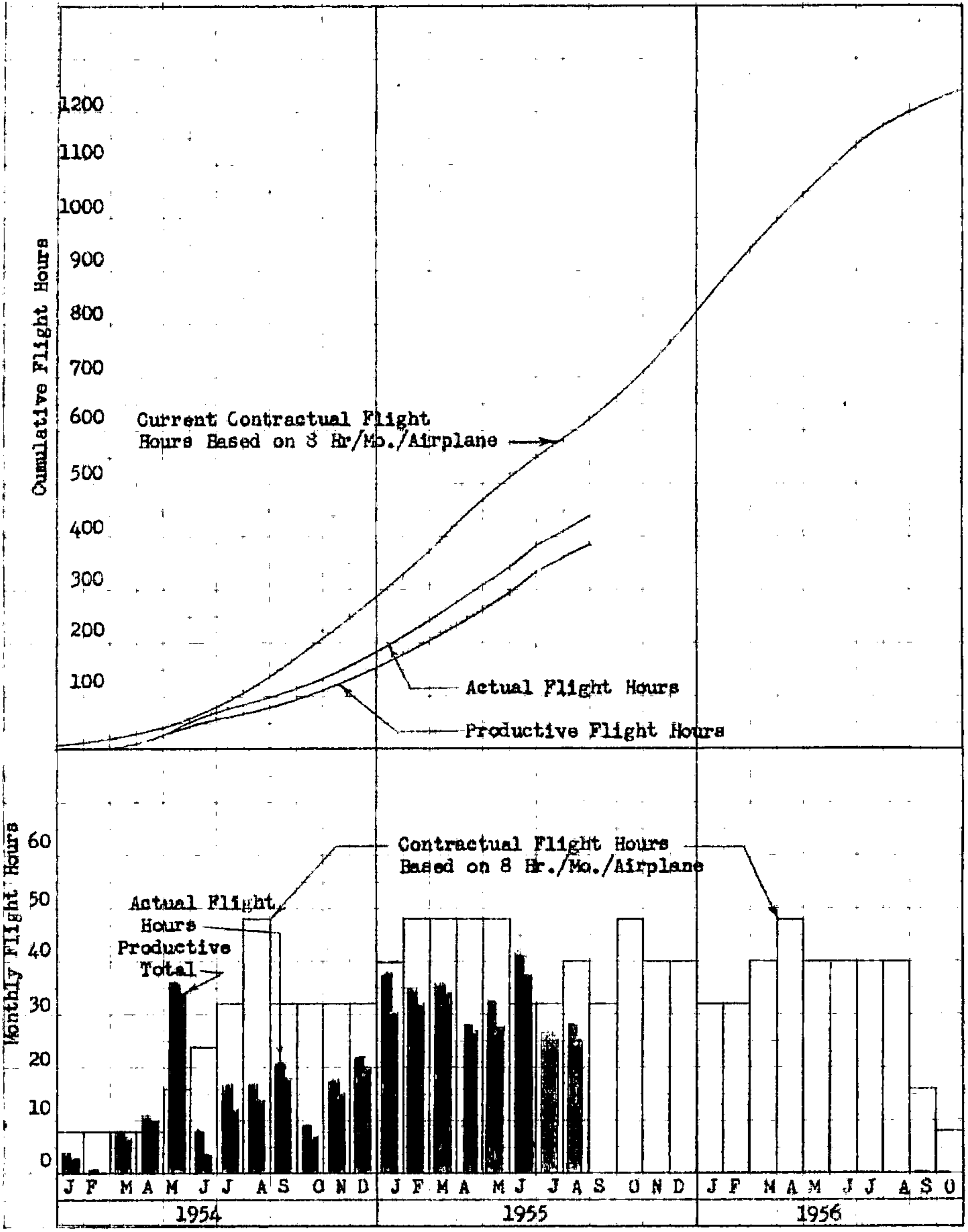
CONVAIR F-102 ACCELERATED DEVELOPMENT PROGRAM
AIRCRAFT ASSIGNMENT SCHEDULE AND ACTIVITY STATUS

Graph No. 1



Graph No. 1A

CONVAIR F-102 ACCELERATED DEVELOPMENT PROGRAM
ESTIMATED FLIGHT TIME AND FLIGHT HOURS FLOWN



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SUMMARY OF FLIGHT TEST ACTIVITIES FOR AUGUST 1955

Convair test airplanes flew 28.1 hours in 45 flights during August 1955. Of this total 25.1 hours in 38 flights were flown towards the accomplishment of the Convair Development Test Program. The average productive flight time of the four airplanes being tested during August was 6.4 hours per airplane. Convair flew 44 flights and the Air Force flew one.

Flight test activities of special interest were:

(a) Airplane S/N 53-1787 flew Flight No. 102 on 5 August with shock wedges installed forward of the engine inlet ducts. The wedges proved unsatisfactory. A Ram Air Turbine test was performed on 18 August during Flight No. 107 and a transonic and supersonic tuft study was completed on Flight No. 110 on 24 August.

(b) Airplane S/N 53-1788 flew the first flight of the HAC 2.00 Inch Rocket Program on 15 August.

(c) Airplane S/N 53-1791 flew Flight No. 20 on 18 August with an eleven inch inlet duct ramp extension which eliminated the buzz condition below Mach 1.3.

(d) Airplane S/N 53-1794 completed its first shakedown flight on 25 August. This airplane is scheduled to begin its actual Flight Test Program on 1 December.

The current test airplane assignment and test facility assignment is as follows:

Serial No.	Location	Assignment
53-1780	Edwards AFB	Controls Development
53-1787	Edwards AFB	Stability & Control, Reduced Wt.
53-1788	Holloman AFB	Armament
53-1791	Edwards AFB	Phase I
53-1794	Edwards AFB	Instrumentation for Test Program.

A Summary of Convair flight operations during August is as follows:

Serial Number	Total Hours	Prod. Hours	Number of Flights	Days Flown	Major Ground Work Items
53-1780	5.3	5.3	8	7	Periodic A & E Inspection
53-1787	9.6	8.3	15	12	Shock Wedge Installation & Inertia Coupling Instrumentation.
53-1788	9.8	9.1	16	11	Rocket Tube Mod. & Eng.Shroud Work
53-1791	2.9	2.4	5	5	Inlet Duct Extension Installation, Buzz Instrumentation & Pilot Assist Activation.
53-1794	0.5	0	1	1	
Totals	28.1	25.1	45	36	

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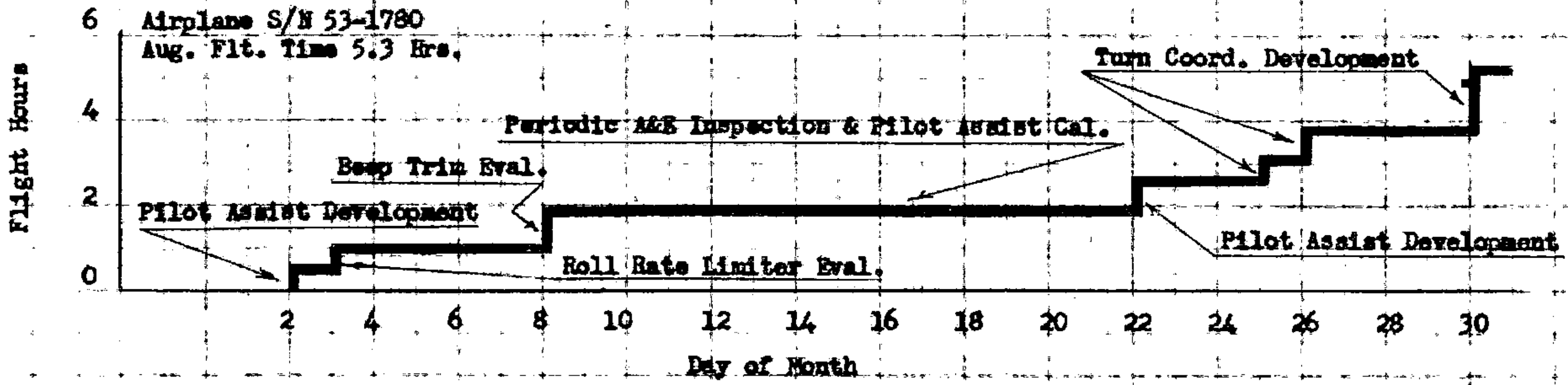
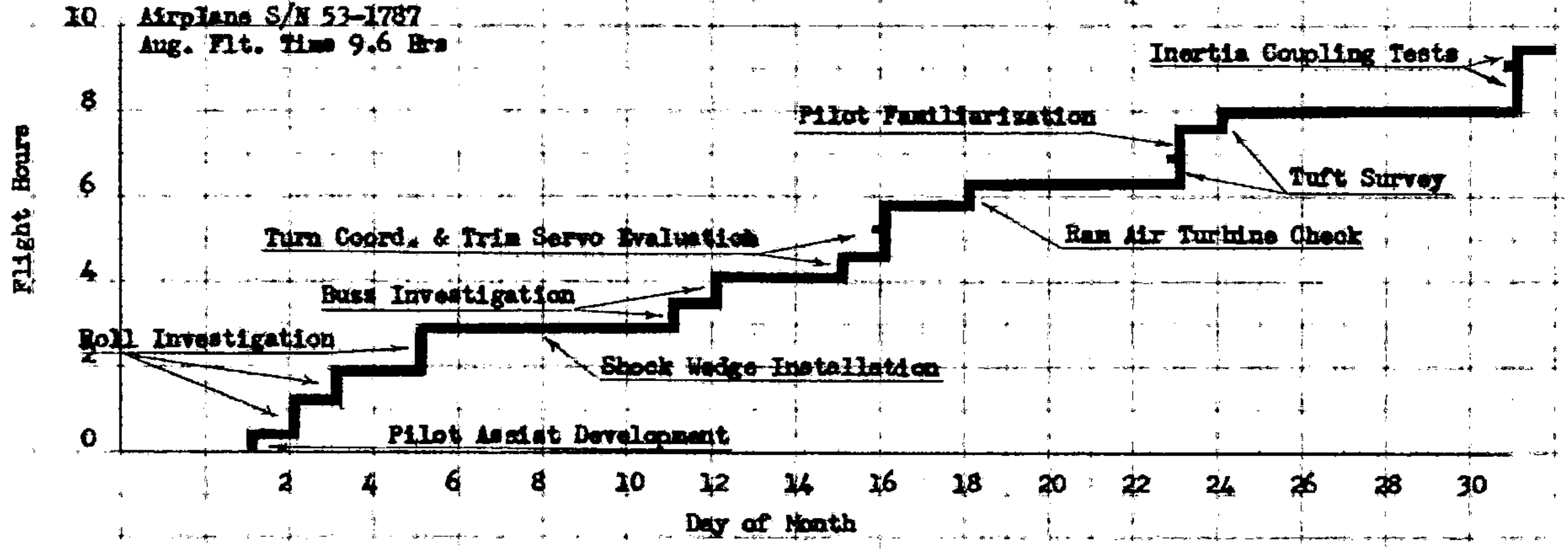
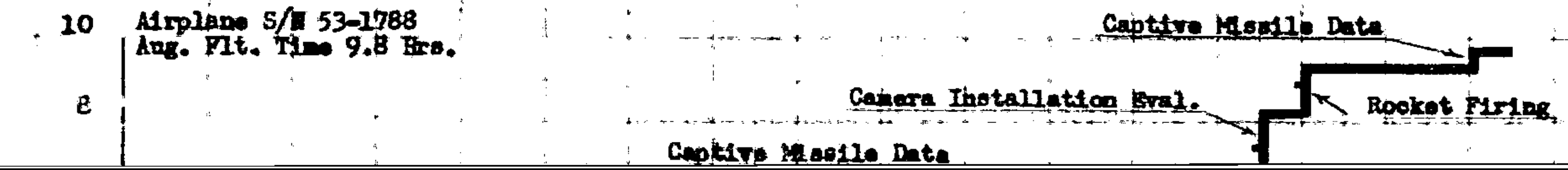
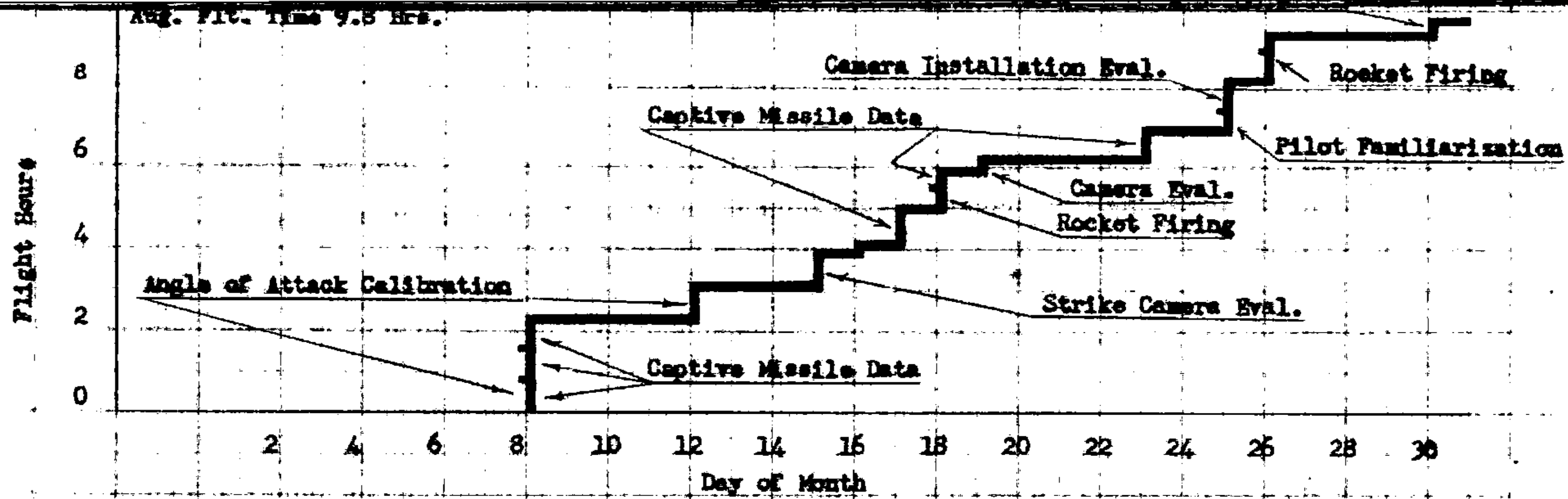
Graph No. 2

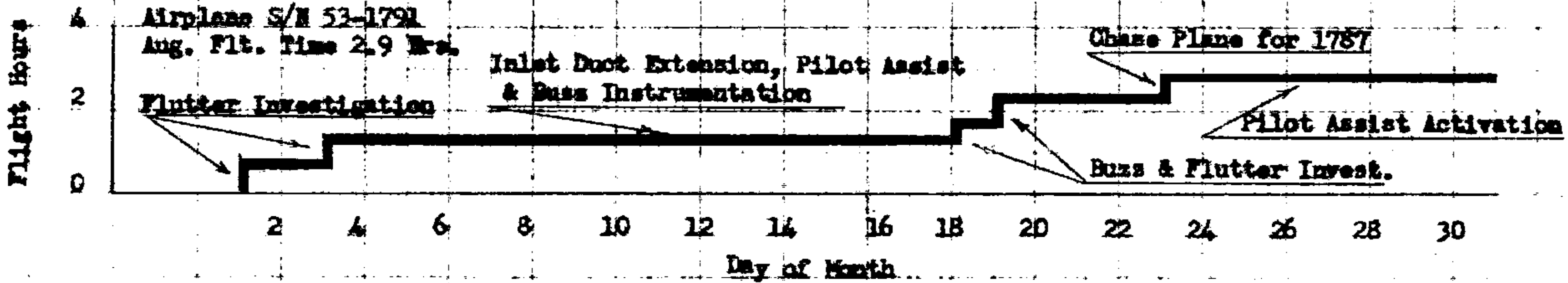
FLIGHT TEST ACTIVITY
August 1955

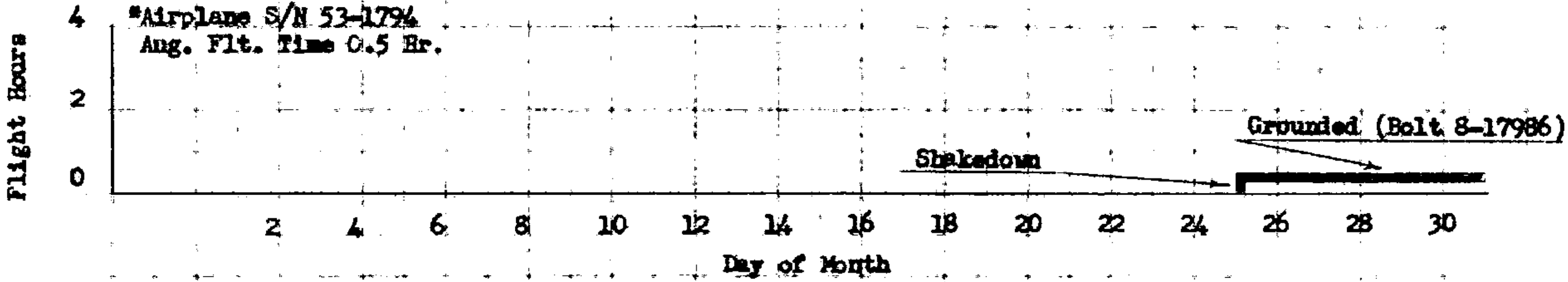












* S/N 53-1794 has not yet begun the actual Flight Test Program.

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY

Airplane S/N 53-1780

Assignment

This airplane is assigned to Convair for a period of 15 months which commenced in July 1954. The formal statement of flight test assignment for this airplane as presented in CCN MSN 5N-4395(159) dated 7 April 1955, is as follows: "To be utilized for Phase III testing of components sub-systems (pneumatic, hydraulic, electrical, and control) tests and development." Convair has divided this assignment into seven general flight test stages in order to effectively accomplish the required flight test operation. Graph No. 4 presents these seven stages as originally programmed in Convair Report No. ZC-8-112, the periods of flight test activity on each stage, the future period of activity currently planned on each stage, and the flight time accumulated to date.

Summary of Activity

Stability and control and dynamic response were the primary considerations in the early phases of this test program. This Phase III airplane has been untilized primarily for testing and refinement of the control, damping and pilot assist systems which will be incorporated in the 8-10 version. Along with this testing, dynamic response data has been obtained for development of the Hughes Aircraft Company fire control system.

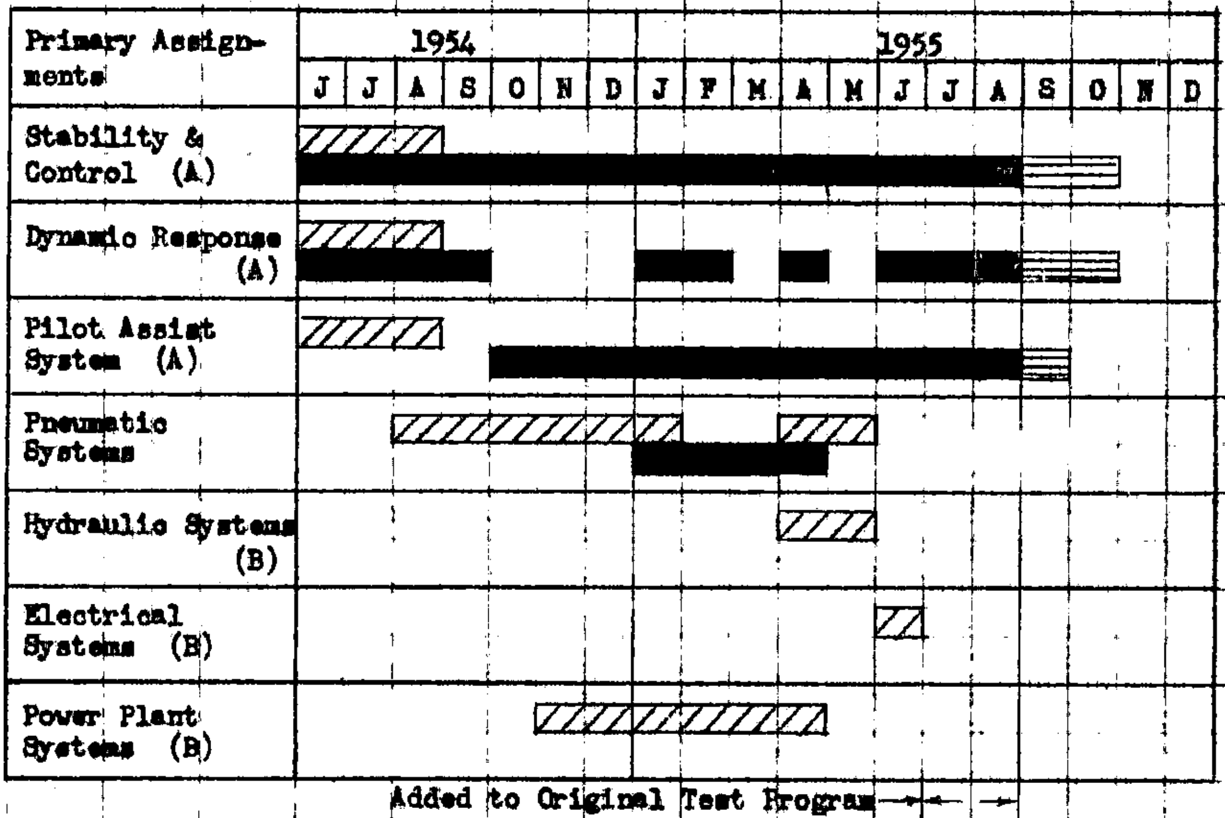
Some testing of the pneumatic system ATM power and cooling have been conducted along with the pilot assist development tests.

Control system testing continued on S/N 53-1780 during the month of August. Emphasis was placed on the optimization of the damper and pilot assist systems and turn coordinator evaluation over a greater variety of flight conditions for the purpose of establishing initial settings for the F-102A systems. Concurrently, intensive developmental testing has continued on the pneumatic roll rate limiter and on the electronic pitch "g" limiter. Environmental vibration data also was obtained as part of the program to improve rate gyro reliability.

Future testing now scheduled for S/N 53-1780 will include further development of the roll rate and pitch "g" limiters as well as an evaluation of reduced force lockouts in the elevon system. Pitch damper and pilot assist gain optimization investigations and turn coordinator development are also expected to continued. Following this, a development program for the improvement of the 8-80 and 8-82 longitudinal stability characteristics through the substitution of 8-10 elevons and fuel system components is tentatively scheduled.

Summary of Activity
S/N 53-1780

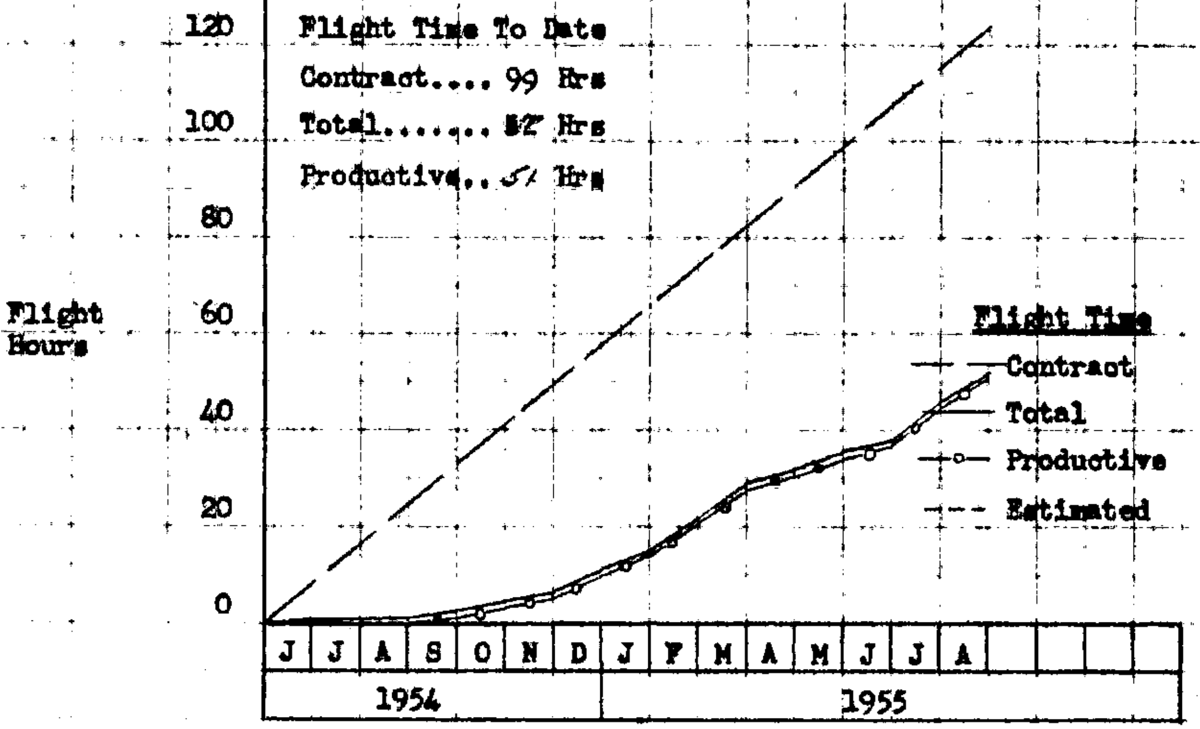
Graph No. 4



Original Planning ▨ Activity ■ Future Planning ☰

(A) Test Assignments Transferred from Airplane S/N 53-1779.

(B) Proposed Cancellation - Specification Compliance Items.



INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY
(Continued)

Airplane S/N 53-1787

Assignment

This airplane is assigned to Convair for a period of 12 months which commenced in January 1955. The formal statement of flight test assignment for this airplane as presented in CCN MSN 5N-4395(159) dated 7 April 1955, is as follows: "Prototype tests of reduced weight aircraft. Phase I, II and III." Convair has divided this assignment into four general flight test stages in order to effectively accomplish the required flight test operations. Graph No. 5 presents these four stages as programmed in Convair Report No. ZC-8-118, the periods of flight test activity on each stage, and the flight time accumulated to date. In the event the original program is revised, the future periods of activity will be indicated in addition to the above information.

Summary of Activity

Phase I performance and preliminary flight characteristics have been accomplished, preceded by qualitative evaluation of airplane systems and airspeed calibrations. Roll rate data have been obtained to substantiate analog computer studies for the purpose of establishing roll rate limitations for the subject airplane. Roll rate tests are being continued for the purpose of determining the effects of various entry conditions and the effects of increased static stability (via sideslip signal feedback to the rudder) on the roll coupling problem.

Much of the testing on this airplane has been devoted to an investigation of high speed vibration or "buzz". Many changes in airplane configuration have been made in search of the source of this disturbance. The primary buzz investigation program has now been shifted to S/N 53-1791. S/N 53-1787 will support the investigation on a secondary basis.

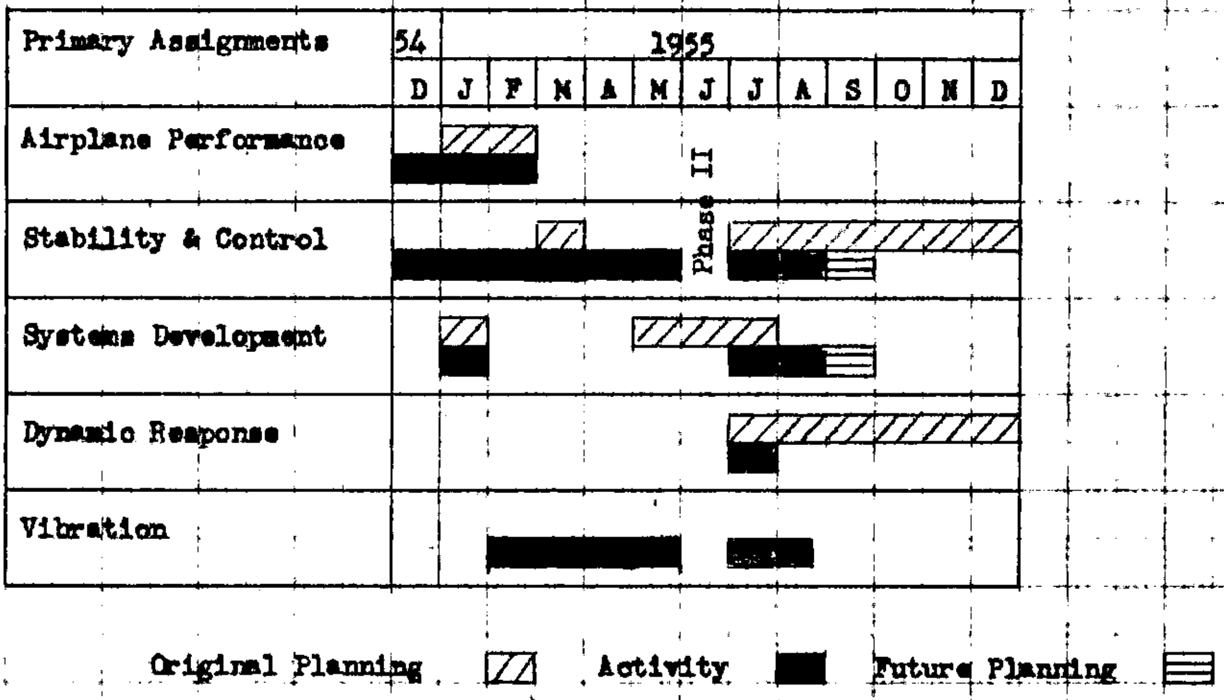
During the month of June Air Force Phase II testing was completed, and the buzz studies continued.

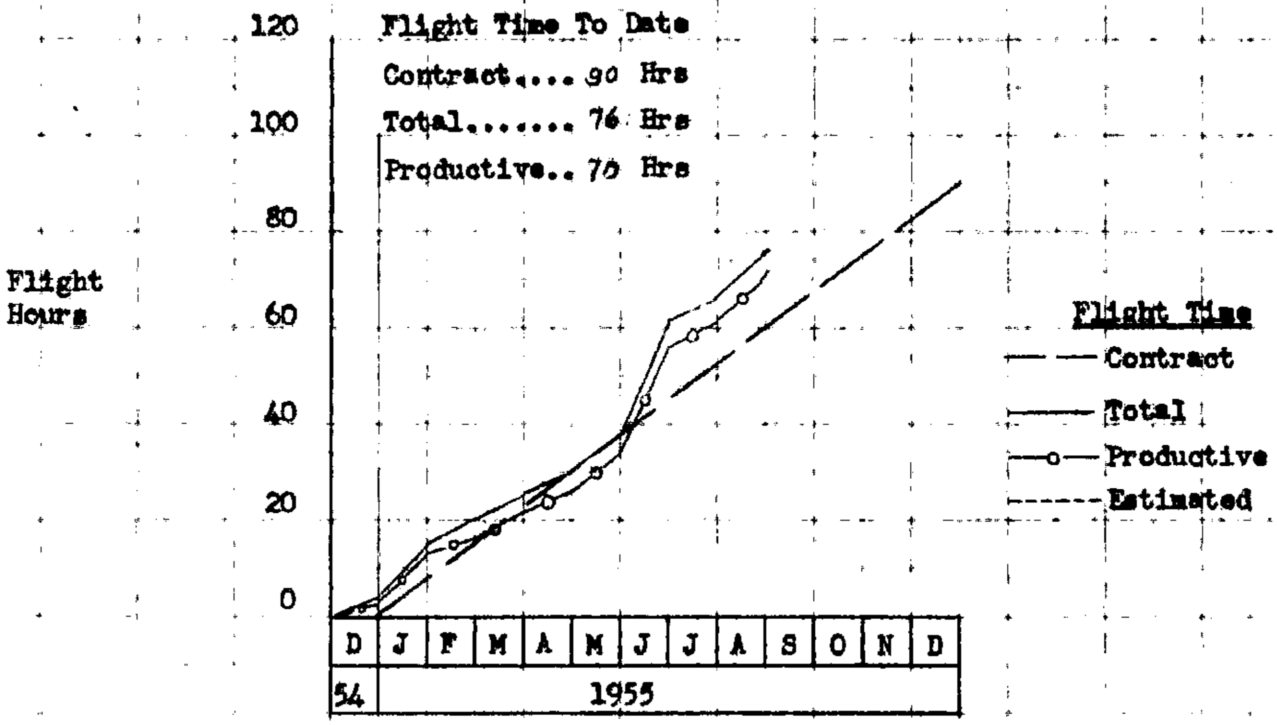
This airplane confirmed the performance predictions for the 8-10 airplane. The manual mode system has been developed to a satisfactory degree, and tests for refinement of the trim servo and Pilot Assist systems are being continued. During the month of August an operational check of the Ram Air Turbine Unit also was completed.

A program to establish the buffet boundaries has been assigned for future activity.

Graph No. 5

Summary of Activity
S/N 53-1787





Flight Time To Date
Contract.... 90 Hrs
Total....... 76 Hrs
Productive.. 75 Hrs

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY
(Continued)

Airplane S/N 53-1788

Assignment

This airplane was assigned to Convair for a period of 5 months commencing in February 1955. The formal statement of flight test assignment for this airplane as presented in CON MCN 5N-4395(159) dated 7 April 1955, is as follows: "To be utilized for subsonic verification of armament tests and supersonic tests and functional development of armament including hardware and dispersion studies. These tests will be conducted up to combat altitude and maximum speed. Utilize for systems test of MX-1179 after completion of armament tests above." Convair has divided this assignment into six general test stages in order to effectively accomplish the required flight test operations. Graph No. 6 presents these six stages as originally programmed in Convair Report No. ZC-8-119, the periods of flight test activity on each stage, the future periods of activity presently planned on each stage, and the flight time accumulated to date.

Summary of Activity

Airplane S/N 53-1788 was ferried to Holloman Air Force Base on 22 February 1955. It commenced the actual Flight Test Program on 14 March 1955.

CVAC missile and rocket firing programs have been conducted and are now considered complete. The missile and rocket separation characteristics appeared satisfactory under the test conditions and no effect on engine performance was encountered.

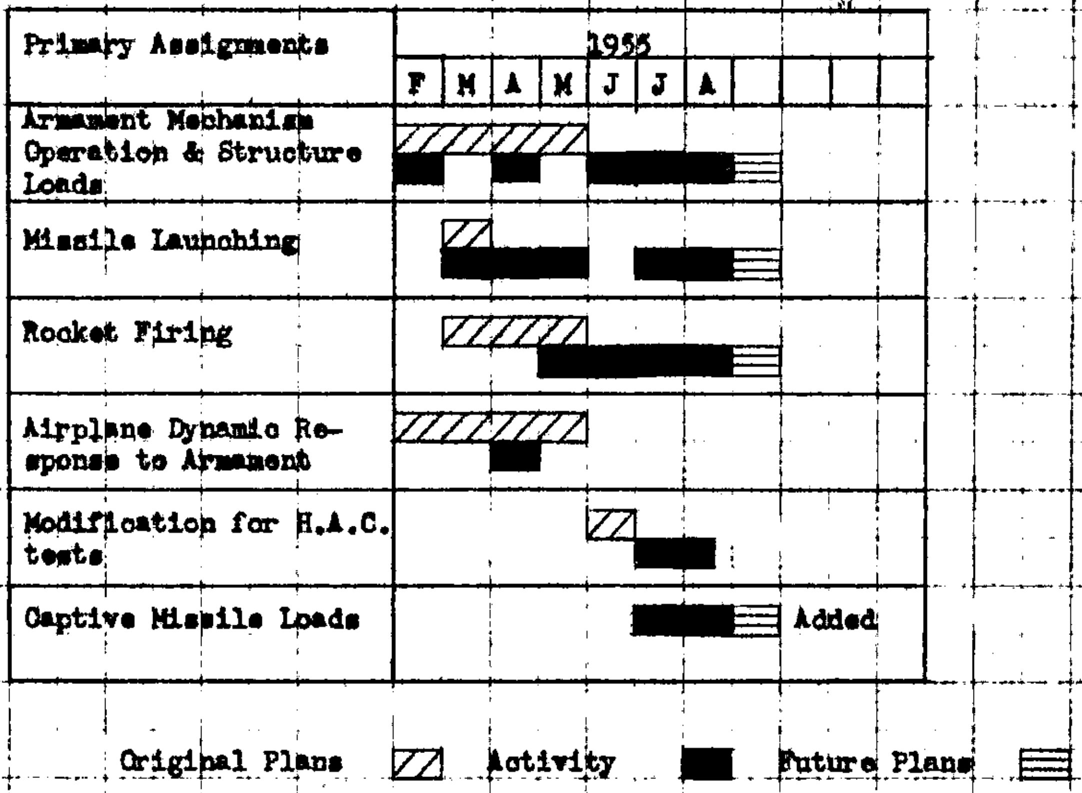
Data which is to be used to eliminate the present speed restrictions at which the armament mechanism may be used has been obtained.

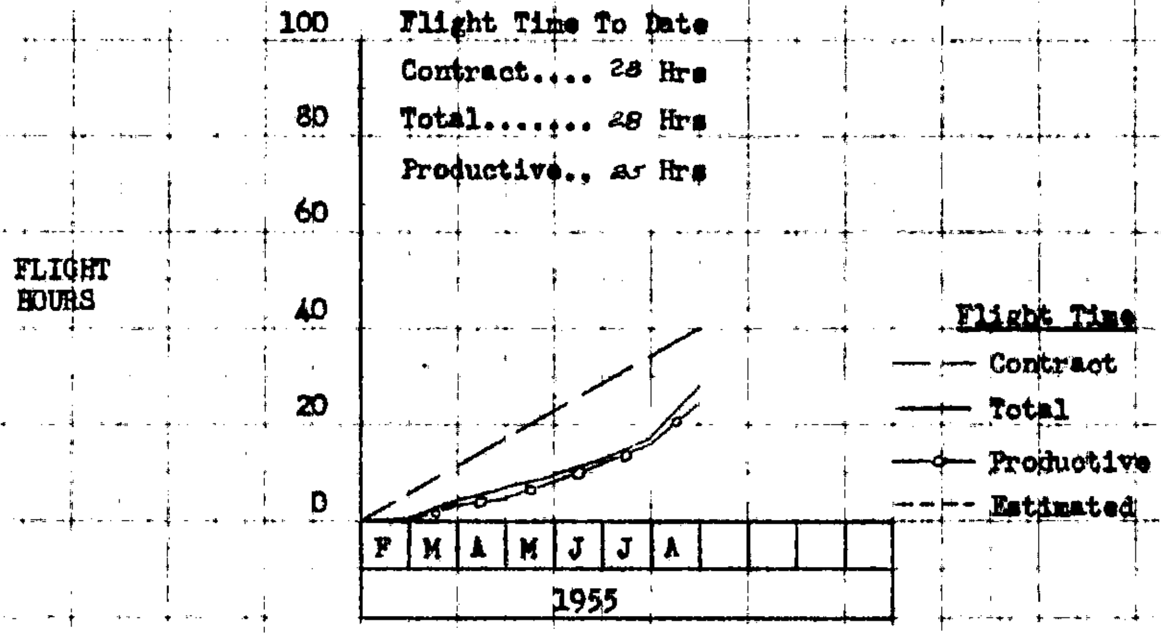
Fast acting armament gear checks and captive missile air loads tests have been conducted. From the preliminary analysis it appears that the results are within the limits expected.

The Strike Camera installation evaluation and the collection of data for the HAC Captive Missile Program took place in August and are scheduled to continue. Ground engine starts without external electrical sources and angle of attack calibrations are scheduled to be performed before the end of the test period.

Summary of Activity
S/N 53-1788

Graph No. 6





Flight Time To Date
Contract.... 28 Hrs
Total....... 28 Hrs
Productive.. 25 Hrs

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY
(Continued)

Airplane S/N 53-1791

Assignment

This airplane is assigned to Convair for a period of 15 months commencing June 1955. The formal statement of flight test assignment as presented in CCN MSN M101047 is: "To be utilized for recheck of Phase I, II and III plus cooling as it applied to the first production light weight model." Convair has divided this assignment into seven general flight test stages, in order to effectively accomplish this flight test operation. Graph No. 7 presents these stages as originally programmed in Convair Report No. ZC-8-120, the periods of flight test activity on each stage, the future periods of activity currently planned on each stage, and the flight time accumulated to date.

Summary of Activity

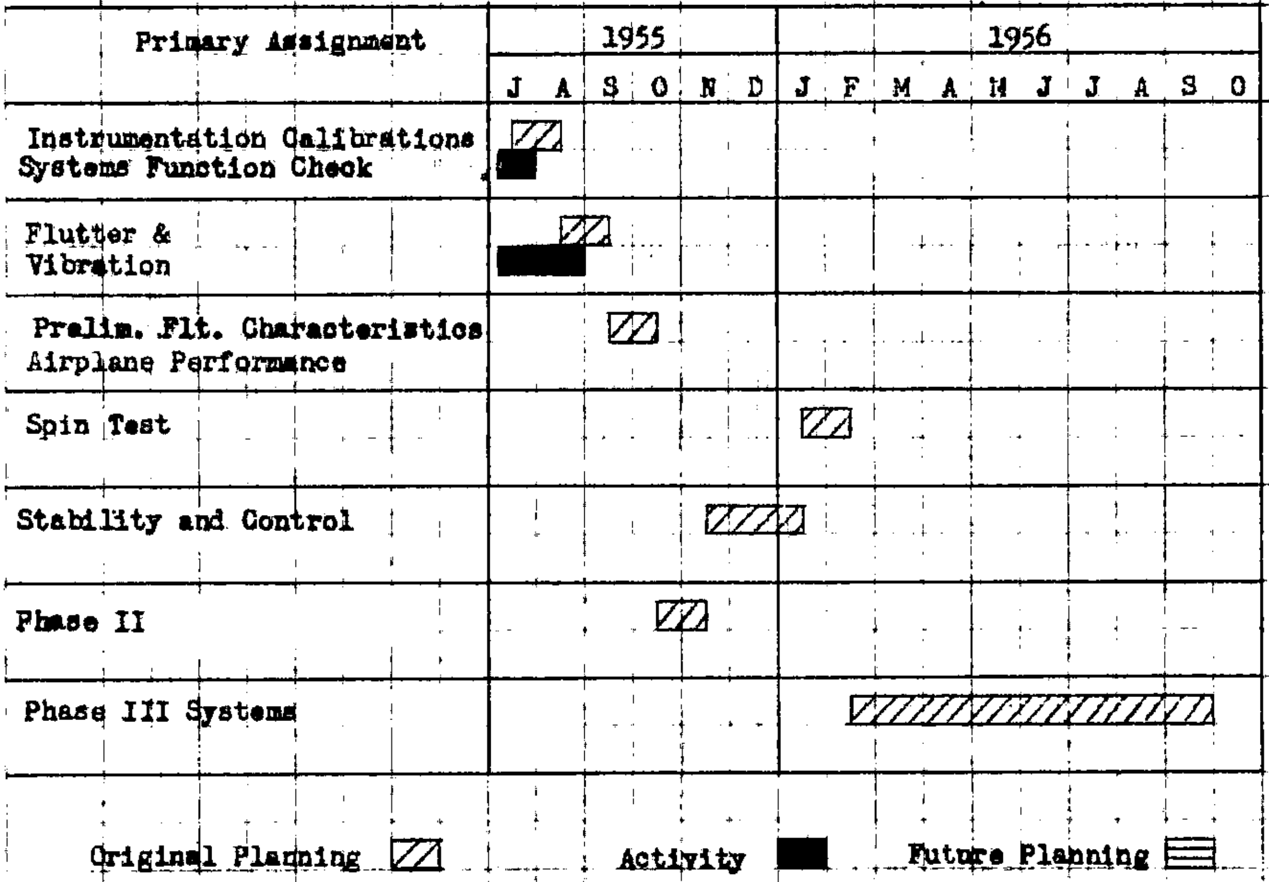
Airplane S/N 53-1791 arrived at Edwards Air Force Base on 21 June 1955. After five shakedown flights, it commenced its flight test on 30 June. During the first month of the Test Program in flight flutter tests and low speed vibration studies were conducted. The low speed low frequency vibration study was considered complete on July 27, when the source of the vibration was determined to be a regulator valve.

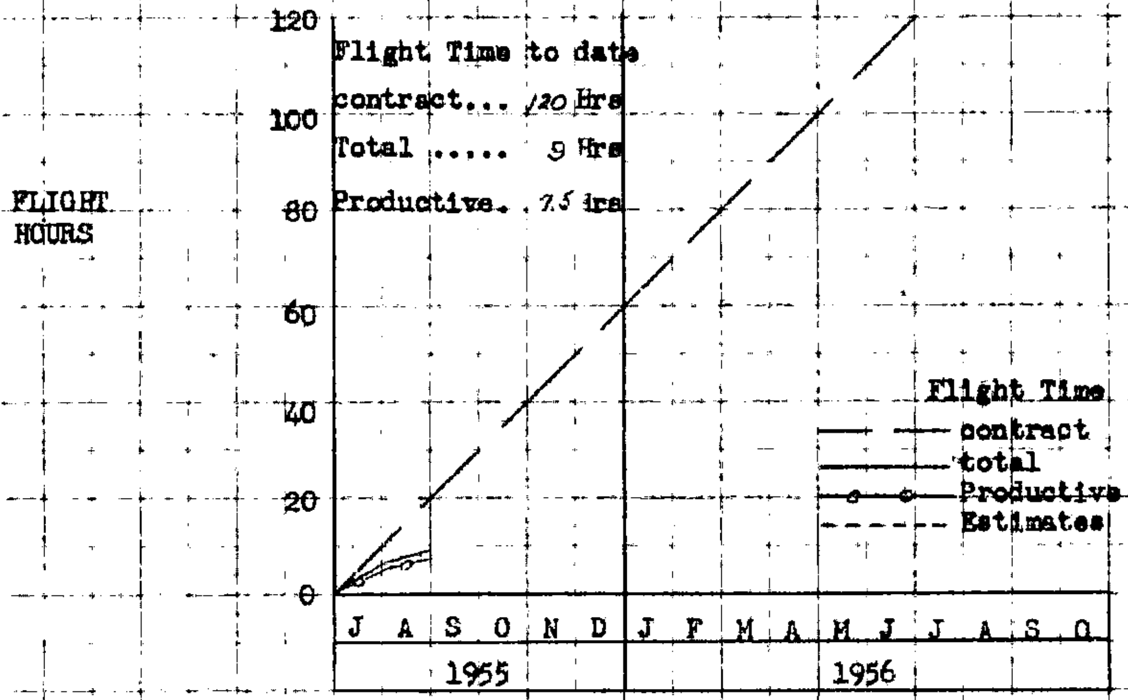
The flight flutter tests were continued through August along with buzz investigations. An extension added to the engine inlet duct ramp resulted in increasing the Mach number at which buzz was first encountered.

Flight tests of flutter and buzz investigations with a longer extension of the inlet ramp are scheduled to be continued through September. Investigation of Pilot Assist and turn coordination, dive brake effectiveness, performance with wing fences removed, nose wheel door air loads, and afterburner nozzle actuator temperatures are some of the studies scheduled to be carried out in the future.

Summary of Activity
S/N 53-1791

Graph No. 7





Flight Time to date
contract... 120 Hrs
Total ..... 9 Hrs
Productive. 7.5 Hrs

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES
(Continued)

Airplane S/N 53-1794

Assignment

This airplane was assigned to Convair for a period of 13 months. The formal statement of test assignment for this airplane as presented in CCN MSN 5N-4395(159) dated 7 April 1955 was as follows: "To be utilized for flight loads determination, operation of high pressure pneumatic system and structural integrity demonstration." Convair divided this program into six general flight test stages in order to effectively accomplish the required flight test operation. Graph No. 8 presents the six stages as programmed.

Summary of Activity

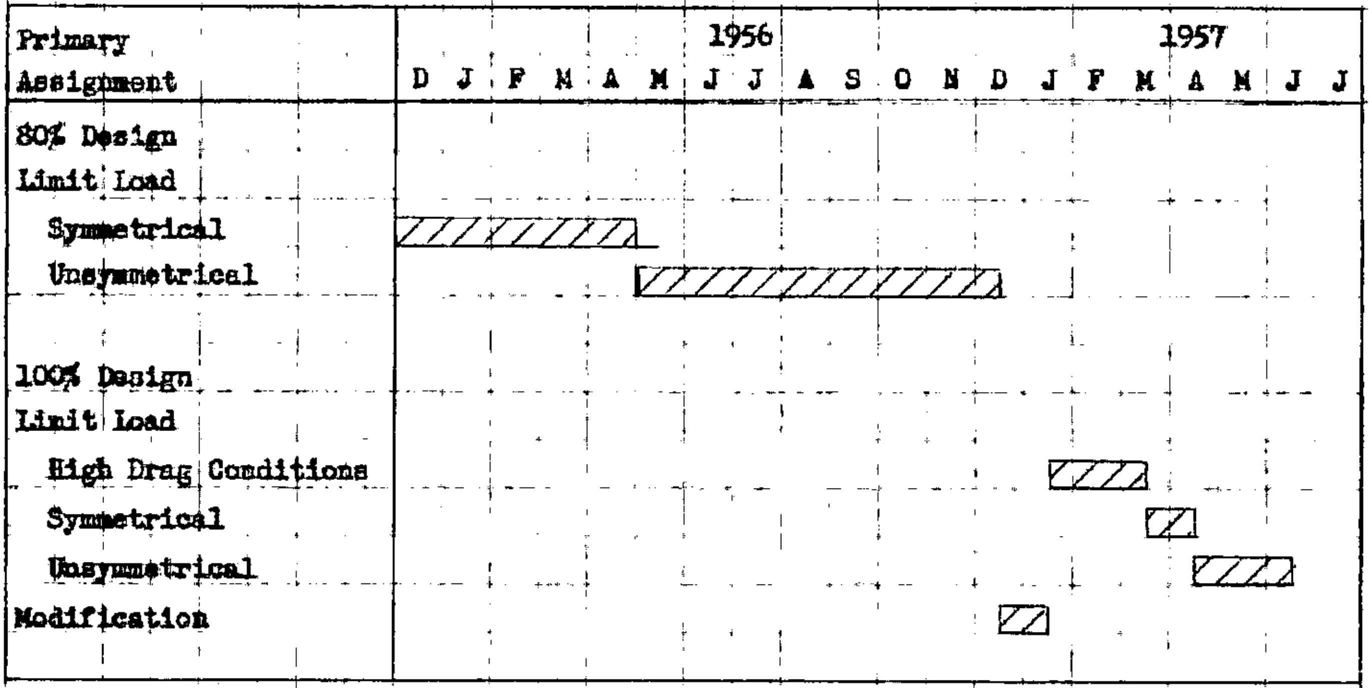
This airplane arrived at EAFB on 16 August 1955. The first shakedown and Air Force acceptance flight was completed on 25 August.

The first flights of the Flight Test Program are scheduled for early December. Prior to that time installation of test equipment and instrumentation check flights will be performed.

SUMMARY OF ACTIVITY

S/N 53-1794

Graph No. 8

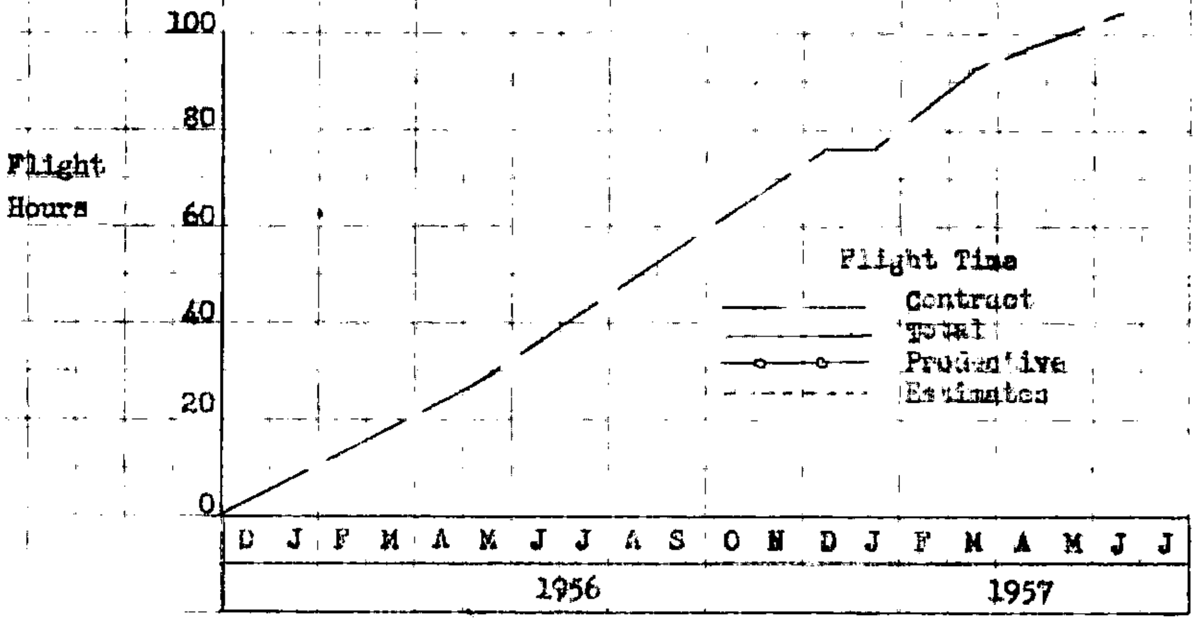


Original Planning ▨ Activity ■

Flight Time to date

Contract - - 100 Hours

Total - - - Hours



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AIRPLANES CURRENTLY

NOT

ASSIGNED

TO A

TEST PROGRAM

CONFIDENTIAL

INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY
(Continued)

Airplane S/N 52-7994

Assignment

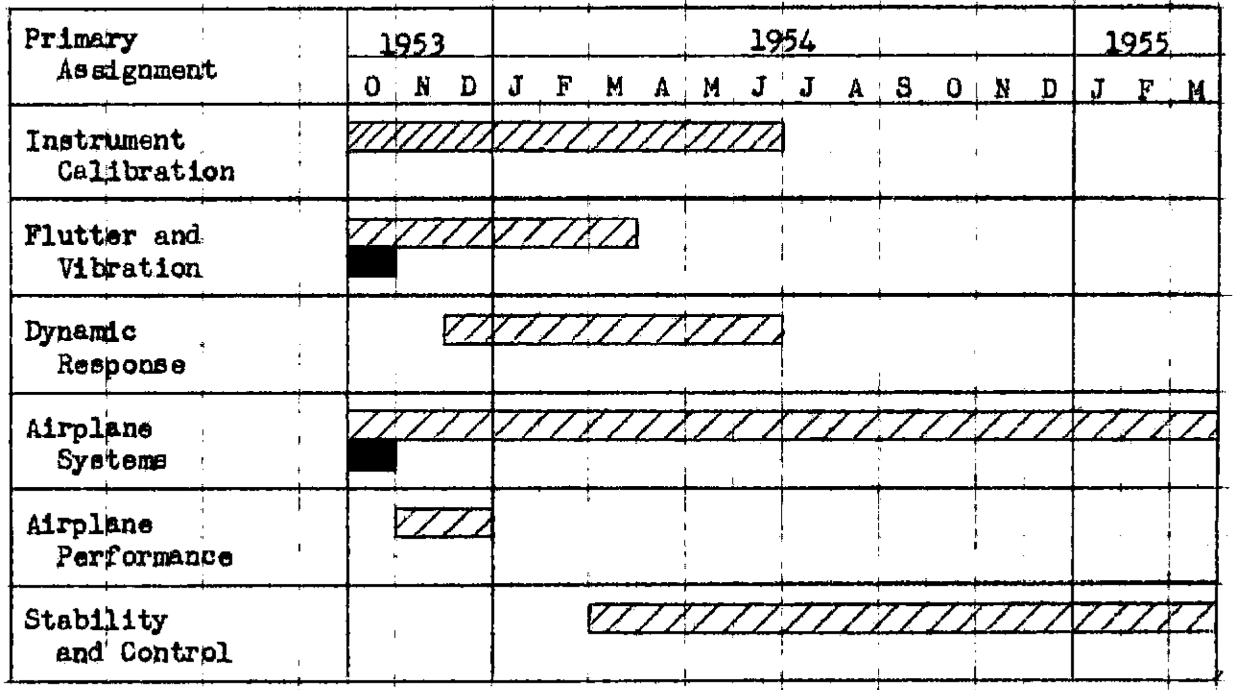
This airplane was assigned to Convair for a period of 18 months which commenced in October 1953. The test assignment of this airplane was to demonstrate and evaluate the basic flying and operating qualities of the YF-102 airplane. This program was authorized by Air Force Contract No. AF33(600)-5942, Supplemental Agreement No. 3. Graph No. 10 presents the six flight test stages that were outlined in Convair Report No. ZC-8-101.

Summary of Activity

Airplane S/N 52-7994 began its test program on schedule. No major mechanical malfunctions were noted. A slight modification to the landing gear was made and flutter tests were in progress. On 2 November 1953, on Flight No. 7, the airplane crashed and was removed from the test program. This flight test program was then transferred to airplane S/N 52-7995, the No. 2 YF-102 airplane.

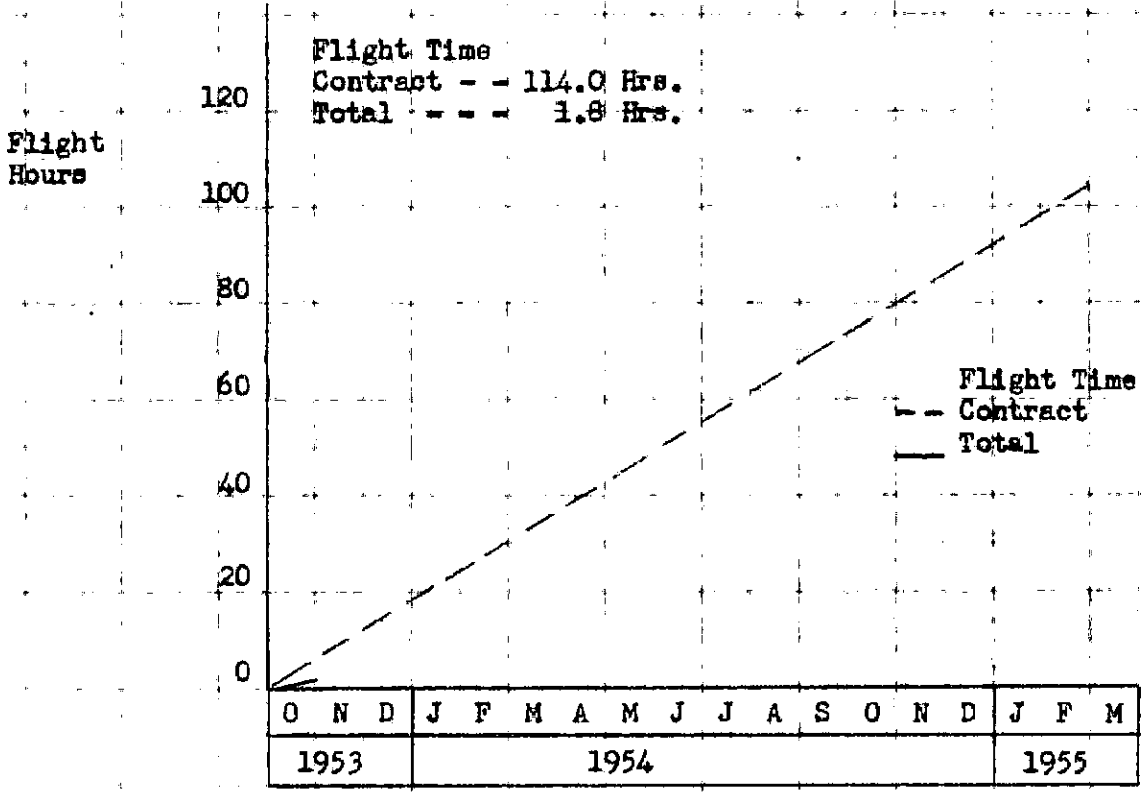
Graph No. 10

SUMMARY OF ACTIVITY
S/N 52-7994



Airplane Crashed 2 November 1953

Original Planning ▨ Activity ■



INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITIES
(Continued)

Airplane S/N 52-7995

Assignment

This airplane was assigned to Convair for a period of 17 months which commenced in January 1954. The formal statement of test assignment for this airplane as presented in CCN MSN 5M-4395(159) dated 7 April 1955, was as follows: "To be utilized for safety of flight tests, initial power plant performance, cooling, pitch and yaw rate damper systems, and pilot relief development. Assume minimum flutter investigation and Phase II, which was not accomplished on No. 1 prototype." Convair divided this assignment into six general flight test stages in order to effectively accomplish the required flight test operation. Graph No. 3 presents the six stages as originally programmed in Convair Report No. ZC-8-103, and the periods of flight test activity on each stage.

Summary of Activity

Preliminary airplane performance and airplane flight characteristics including flutter and vibration tests were completed in the early months of this flight test program. Stability and control tests continued throughout the program as a result of control system changes made to develop the 8-10 versions. Roll rate studies were accomplished in connection with flight characteristics and computer analysis.

This airplane was the second experimental model and performed its early flights with uncambered leading edges and straight wing tips. These were changed to cambered leading edges with reflexed tips and later the tips were changed to full camber without reflex. From a qualitative point of view, the cambered wing tips have not had an appreciable effect on the performance or stability characteristics of the aircraft.

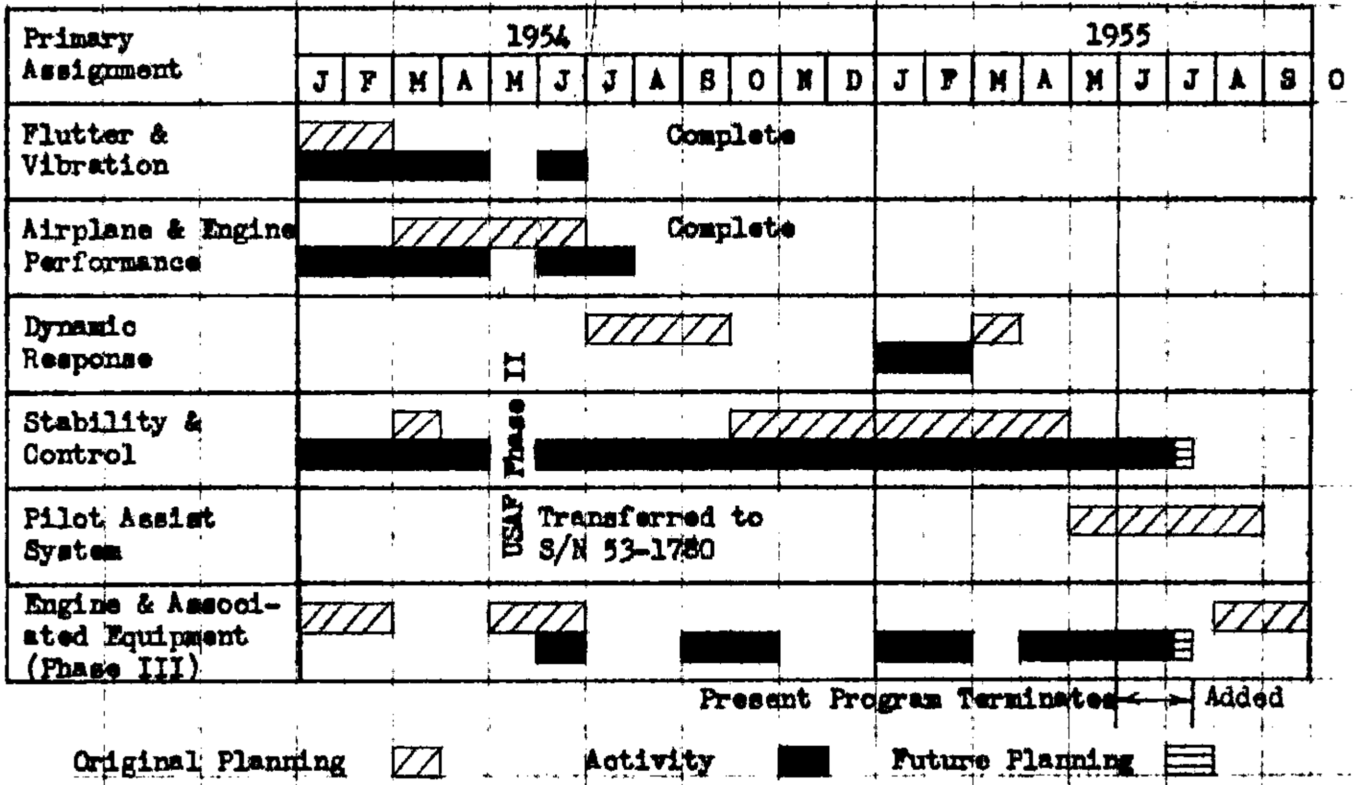
An extended nose was evaluated and several tail cone configurations were compared, the enlarged elevon fairing being one of them. Some variations were made to the elevon fairing during the tests and this along with the extended nose were incorporated into the production models of the airplane.

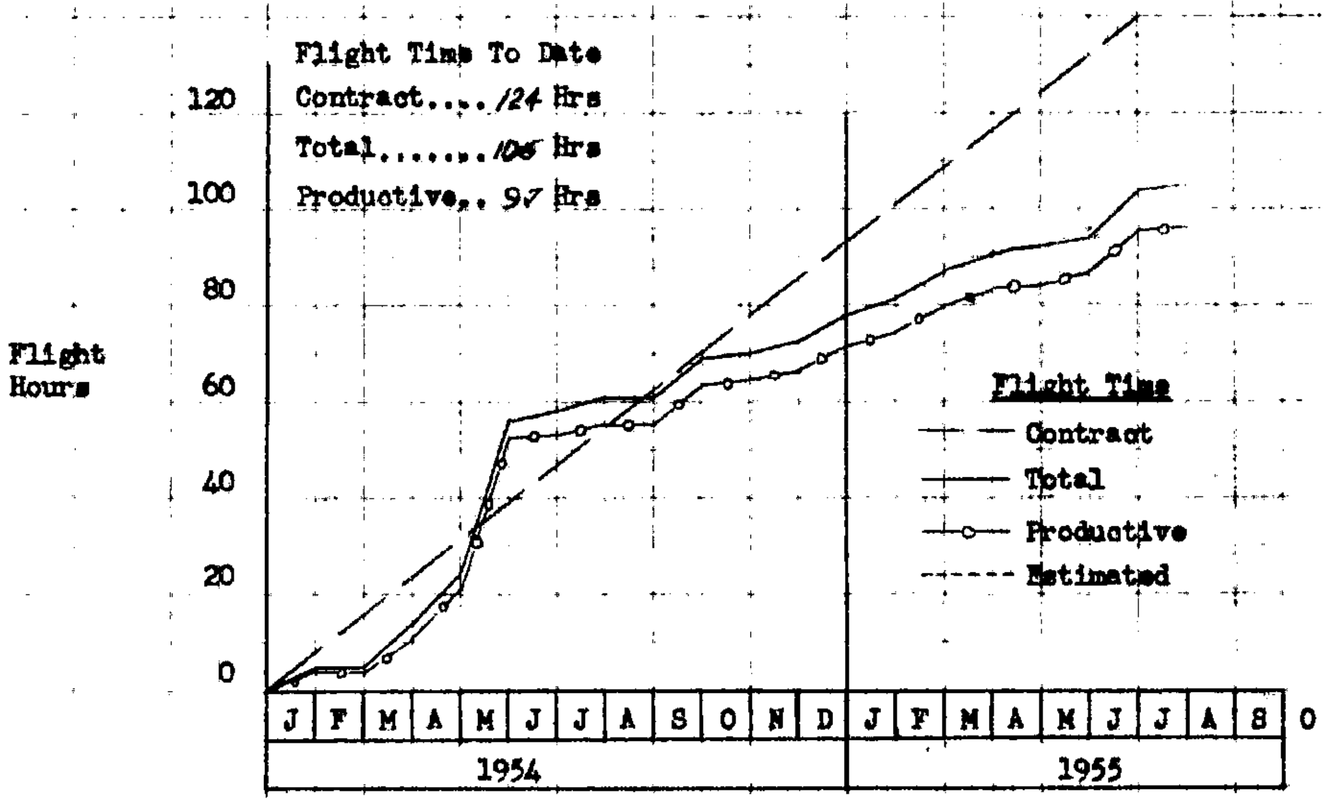
Noise levels at crew stations during ground operations, cockpit noise, radome glycol distribution, performance and stability tests, wing fence evaluation, and many development tests of the engine components and systems were conducted during the period of 19 months of field operations.

The final flight of the test program was flown on 1 July and the airplane was ferried to San Diego on 8 July pending assignment. It completed 163 flights during the test programs.

Summary of Activity **Graph No. 3**

S/N 53-7995





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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY
(Continued)

Airplane S/N 53-1779

Assignment

This airplane was assigned to Convair for a period of 4 months which commenced in May 1954. The primary purpose of Convair testing was the development testing of the pitch and yaw damper systems. This test program was authorized by Air Force Contract AF33(600)-5942, Supplemental Agreement No. 3. Graph No. 11 presents the three flight test stages which were outlined in Convair Report No. ZC-8-108.

Summary of Activity

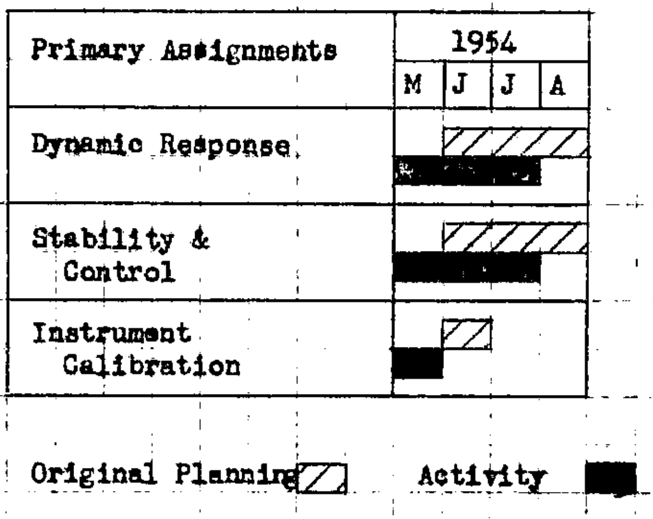
The first test flight of this program was conducted on 18 May 1954. The flight testing proceeded on schedule with dynamic response and stability and control, as related to the pitch and yaw damper systems, being the prime considerations.

Lateral stability tests proved satisfactory and the optimum yaw damper gain settings were obtained. The pitch damping system in its configuration at the end of the development program did not produce satisfactory damping at all airspeeds at both high and low altitudes with a constant gain setting.

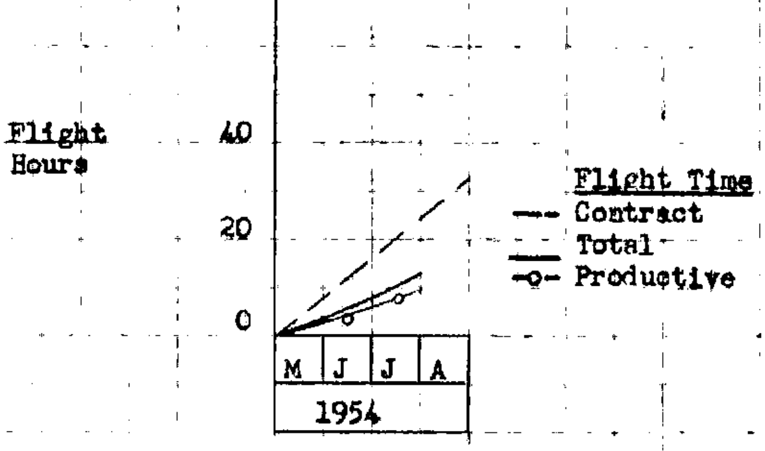
The final flight of this program was conducted on 30 July when this aircraft was delivered to Palmdale, California, for modification prior to delivery to Hughes Aircraft Company.

SUMMARY OF ACTIVITY
S/N 1779

Graph No. 11



Flight Time to Date
Contract —— 32.0 Hrs.
Total ——— 12.6 Hrs.
Productive —— 9.7 Hrs.



INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY
(Continued)

Airplane S/N 53-1781

Assignment

This airplane was assigned to Convair for a period of 14 months which commenced in June 1954. The formal statement of flight test assignment as presented in CCN MSN 5N-4395(159) dated 7 April 1955, is as follows: "To be utilized for Phase III armament firing, functional tests and development, with normal and alternate armament installation." Convair has divided this assignment into five general flight test stages in order to effectively accomplish the required flight test operation. Graph No. 12 presents these stages as originally programmed in Convair Report No. ZC-8-112.

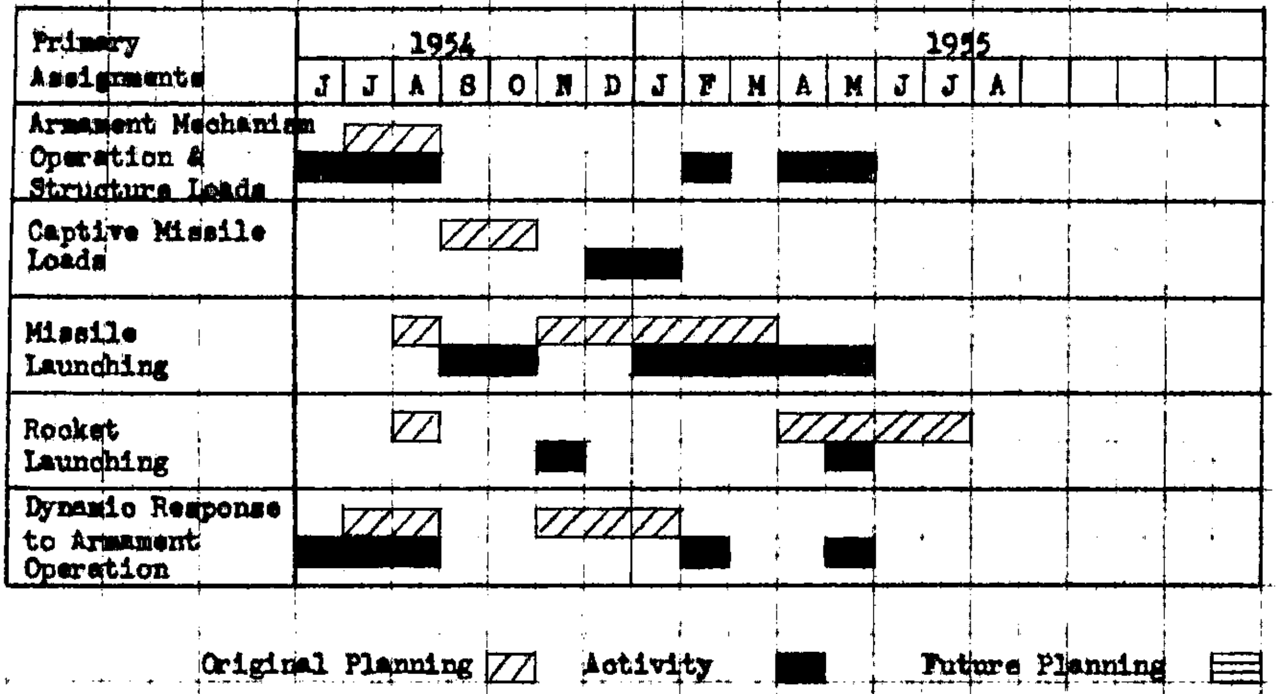
Summary of Activity

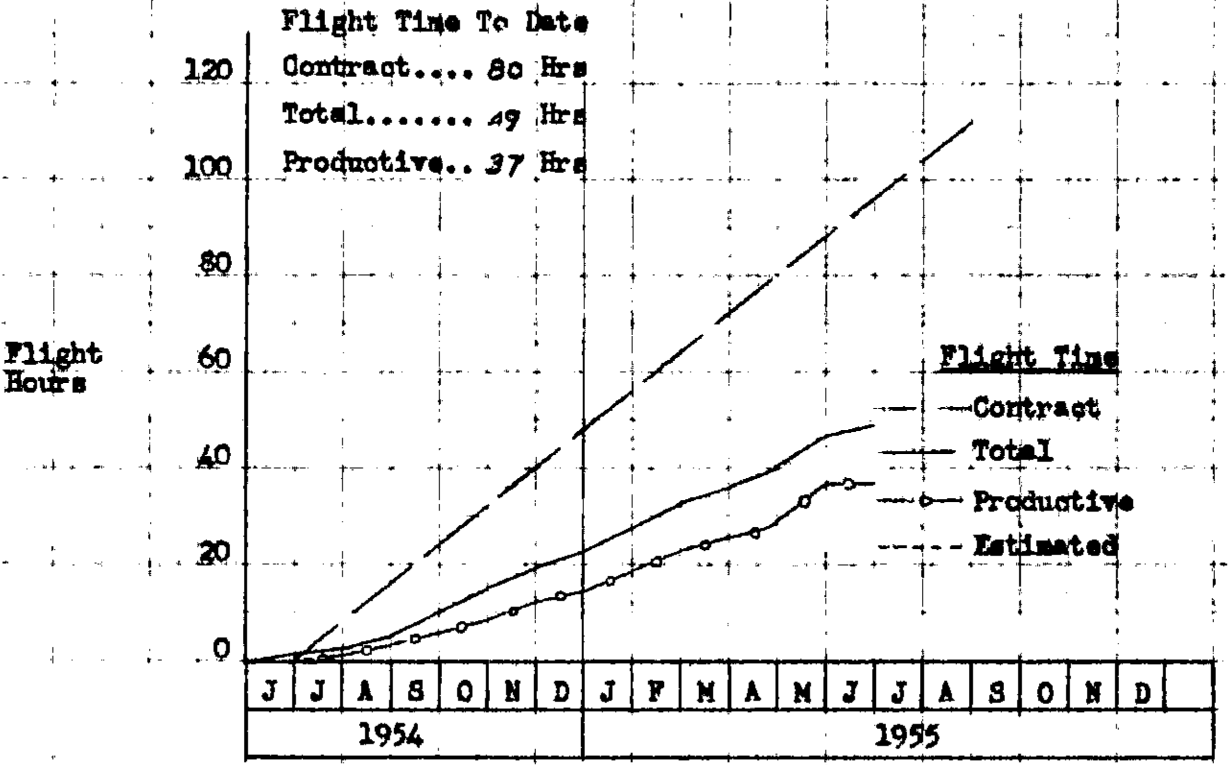
Armament mechanism operation and structure loads were partially completed in the early part of the test program and later completed in May. The captive missile loads and missile launching program were completed as well as the 2.75 and 2.0 inch rocket programs. The armament tests were generally satisfactory. Flight handling characteristics and stability at combat ceiling with the missile bay doors open was good.

This airplane completed its originally assigned flight test task and is currently being modified at the San Diego Experimental Factory to a Ding Dong launching configuration.

Summary of Activity **Graph No. 12**

S/N 53-1781




Flight Time To Date
Contract.... 80 Hrs
Total....... 49 Hrs
Productive.. 37 Hrs

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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY
(Continued)

Airplane S/N 53-1782

Assignment

This airplane was assigned to Convair for a period of one month to demonstrate that the functional characteristics of the production antenna installations were satisfactory for communication and navigation equipment for the F-102 airplane. The testing was to further correlate flight measurements with antenna model studies made at Convair. This testing was authorized by Air Force Contract No. AF33(600)-5942. The testing undertaken by this airplane is shown on Graph No. 11.

Summary of Activity

Although airplane S/N 53-1782 was scheduled for testing in August 1954, testing commenced 28 June 1954. After the initial shakedown flights, instrument calibrating was the first undertaking. Performance, communications and navigation data were obtained in July. Compressor stall investigations were conducted concurrently with the communications and navigation tests. Satisfactory results were obtained from these functional tests of the equipment.

The test program was completed on 18 August 1954, when the aircraft was delivered to Palmdale, California for modification prior to delivery to Hughes Aircraft Company.

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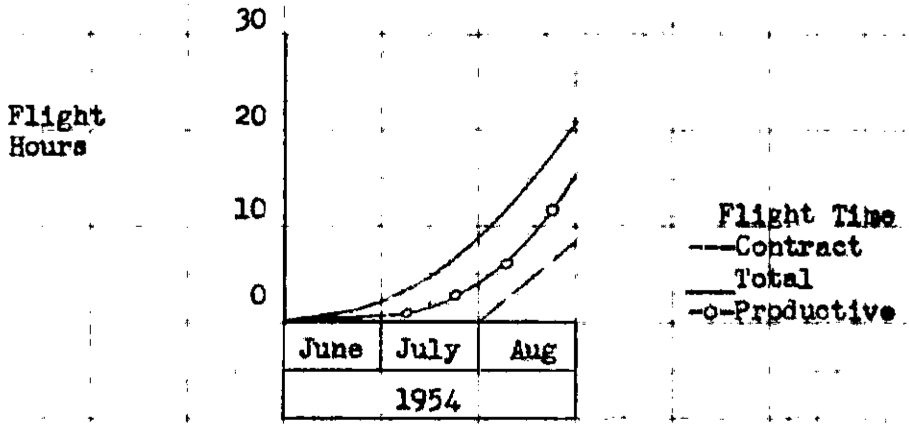
SUMMARY OF ACTIVITY
S/N 1782

Graph No. 11

Primary Assignments	1954		
	June	July	Aug
Electronic - Comm. & Nav.		■	▨ ■
Performance		■	
Instrument Calibration	■		

Original Planning ▨ Activity ■

Flight Time to Date
Contract ---- 8 Hrs.
Total ------ 21.1 Hrs.
Productive-- 15.7 Hrs.



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INDIVIDUAL AIRPLANE SUMMARY OF ACTIVITY
(Continued)

Airplane S/N 53-1783

Assignment

This airplane is assigned to Convair for a period of 10 months which commenced in August 1954. The formal statement of flight test assignment for this airplane as presented in CCN MSN 5N-4395(159) dated 7 April 1955, is as follows: "To be utilized for Phase III tests to include cooling problems. ATM refrigeration unit, cockpit air-conditioning, and correction of deficiencies disclosed during Phase I and II. The aircraft may also be used for Pilot Assist System development in event of difficulties with airplane S/N 52-7995." Convair has divided this assignment into seven general flight test stages in order to effectively accomplish the required flight test operations. Graph No. 12 presents these six stages as originally programmed in Convair Report ZC-8-114, the periods of flight test activity on each stage, the future periods of activity currently planned on each stage, and the flight time accumulated to date.

Summary of Activity

The original flight test program for S/N 53-1783 was changed considerably. Instrumentation calibration production items, engine and structural cooling, and pilot assist phases of the test program were deleted to permit delivery of this aircraft to WADC in January 1955. Dry air anti-icing, defogging, and cabin ventilation and pressurization tests were limited to qualitative testing to permit delivery to WADC in January 1955.

An added task of communications and navigation systems testing was completed during the second month of the program. These tests were transferred from S/N 1782.

Air Force pilots evaluated S/N 53-1783 in various icing conditions. In numerous flights, artificial icing conditions were employed behind tanker type aircraft. Preliminary information indicated satisfactory performance of this airplane in the icing conditions encountered. The anti-icing tests were completed on 3 May with Flight No. 82. As a result of the tests in icing conditions without wing and tail anti-icing, the anti-icing equipment has been deleted from the 8-10 airplane with a considerable saving in weight.

S/N 53-1783 was ferried from WADC to EAFB and arrived at Edwards on 15 May. It is now in a standby status at Palmdale, California.

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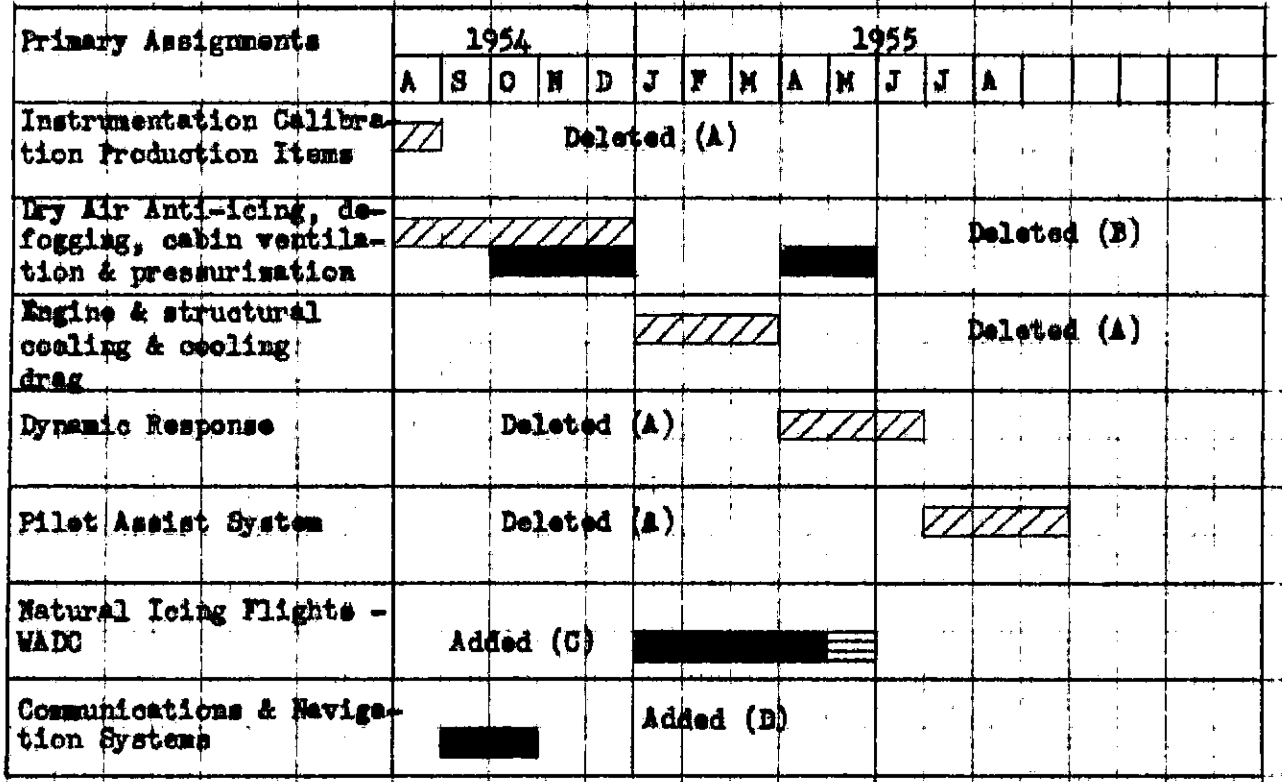
Summary of Activity
S/N 53-1783

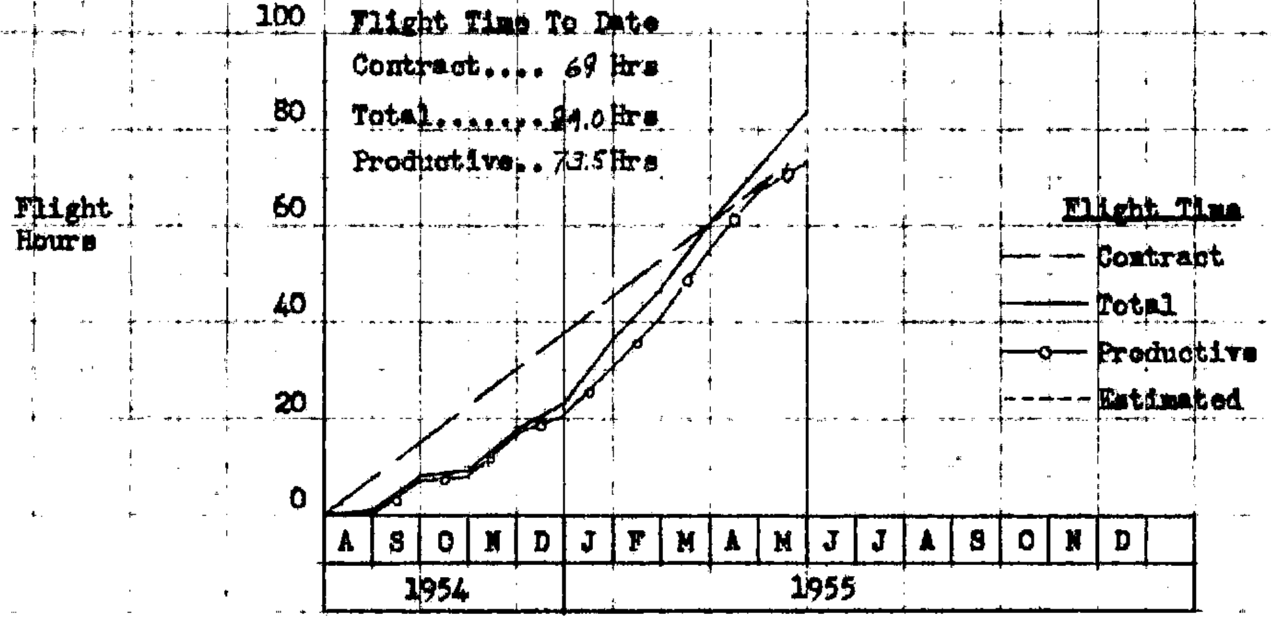
Graph No.12



Convair Proposes Termination ←

Original Planning ▨ Activity ■ Future Planning ☰

(A) Deleted to permit delivery of A/C to WADC Jan. 1955.
(B) Wing & tail anti-icing, cabin pressurization deleted. Cabin vent & defogging limited to qualitative tests to permit delivery to WADC Jan. 1955.
(C) Added to original test program - tests are being flown by WADC personnel.
(D) Transferred from Airplane S/N 53-1782.



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REPORTS PUBLISHED DURING AUGUST 1955

Weekly Flight Operations Reports

Report No. ZC-8-512-22 covering week ending 31 July.
Report No. ZC-8-512-23 covering week ending 7 August.
Report No. ZC-8-512-24 covering week ending 14 August.
Report No. ZC-8-512-25 covering week ending 21 August.
Report No. ZC-8-512-26 covering week ending 28 August.

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DEFINITIONS

For purposes of this report, the following definitions apply.

(1) Contract Flight Time - Based on 8 Hrs/maintenance month.

(2) Total Flight Time - Airplane total flight time during the period the airplane is assigned to Convair for Flight Testing.

(3) Productive Flight Time - Flight time during which data in furtherance of the development test program was obtained. Air Force Phase II testing is included in this time.

(4) Anticipated Flight Time - Estimate of flight time to be conducted during the coming month.

(5) Convair Test - Flights in which the aircraft has taken off with a Convair pilot, in direct pursuit of fulfilling a portion of the scheduled test program (includes aborts).

Classification Changed to: UNCLASSIFIED

Authorized by: [illegible] 214 Date: 6/20/60

Reclassified by: FCB Dept:31 Date: 8/11/60

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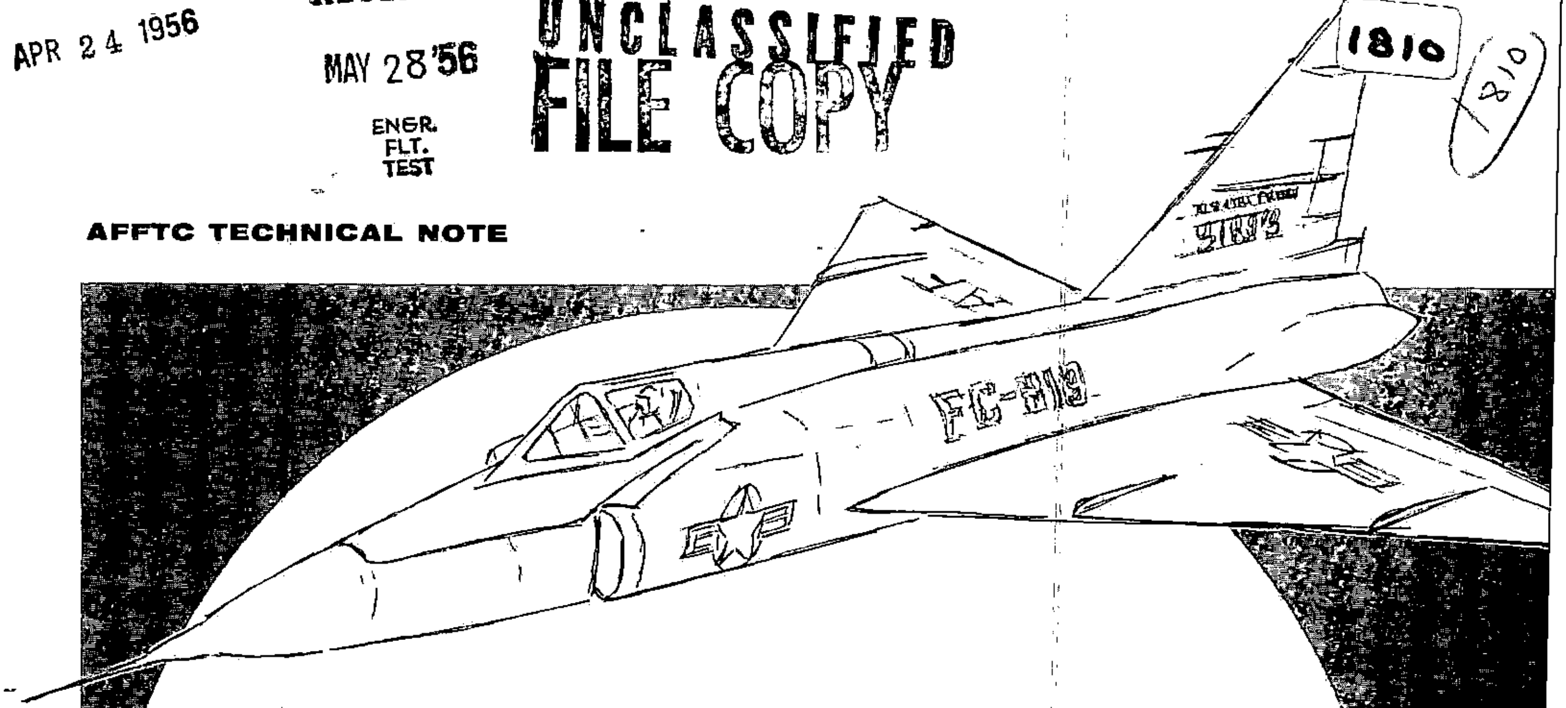

ROLL EVALUATION OF THE F-102A WITH AN ENLARGED VERTICAL STABILIZER

TITLE UNCLASSIFIED

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Authorized by: SAAMA Date 1[illegible]
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Reclassified by:
L. G. Avitt
Dept.
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MARION H. YANCY, JR.
Project Engineer

MAJOR ROBERT L. STEPHENS
Project Pilot

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EDWARDS AIR FORCE BASE, CALIFORNIA
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE

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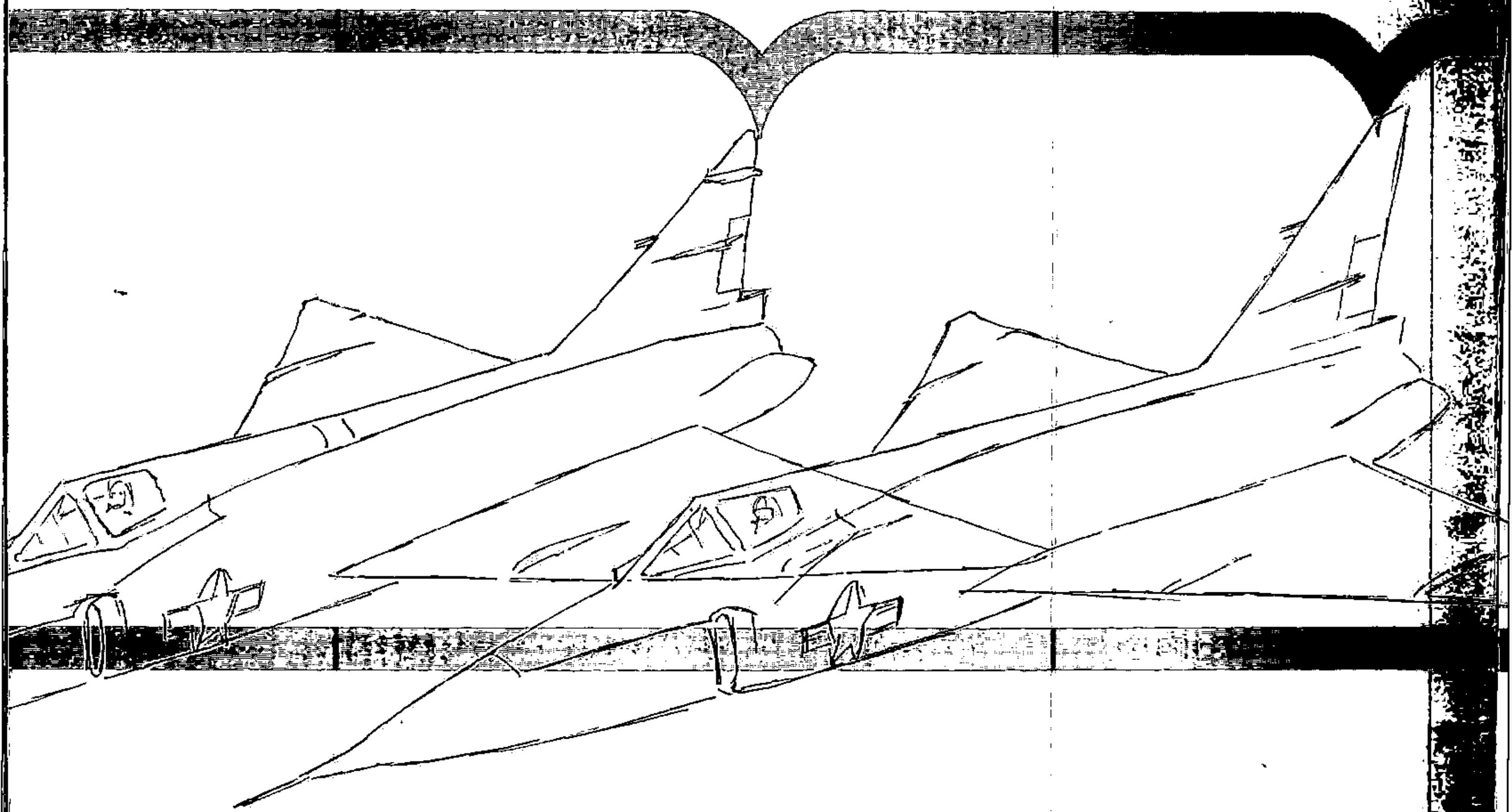

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AFFTC-TN-56-15 APRIL 1956

ROLL EVALUATION OF THE F-102A WITH AN ENLARGED VERTICAL STABILIZER

MARION H. YANCY, JR.
Project Engineer

ROBERT L. STEPHENS
Major, USAF
Project Pilot

Classification Changed to:
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Authorized by: SAAMA Date 12-12-66
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Reclassified by: Dept. Date
L.G. Avritt 130-1 4-4-67

[redacted]

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this technical note has been reviewed and approved*

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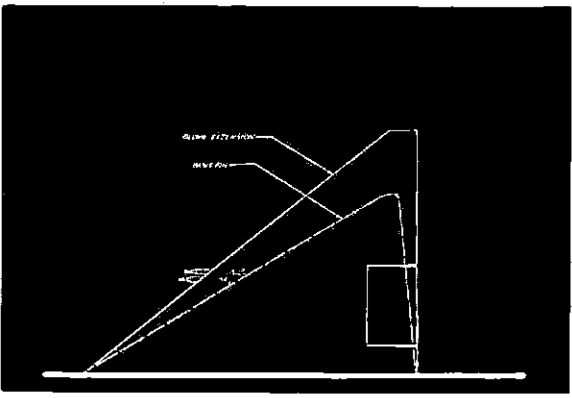
H. A. HANES
Colonel, USAF
Director, Flight Test

J. S. Holtoner
J. S. HOLTONER
Brigadier General, USAF
Commander

*3 APRIL 1956



abstract

The inherent stability provided by the 40% increase in vertical tail area is sufficient to eliminate any roll restrictions because of inertial coupling dangers. These roll tests show that the test airplane has less inertial coupling effects and better handling and general flying qualities than the standard F-102A with the sideslip feed-back operating. The 100% reliability of the large tail is much more desireable than the small tail with a sideslip feed-back which does not incorporate a satisfactory safety device. The performance of the airplane has not been perceptibly changed. An aileron restriction of 10 degrees (total deflection) brought about structural considerations is adequate for operational requirements of the clean airplane. Aileron requirements for spin recovery and asymmetric load conditions, however, were not determined.

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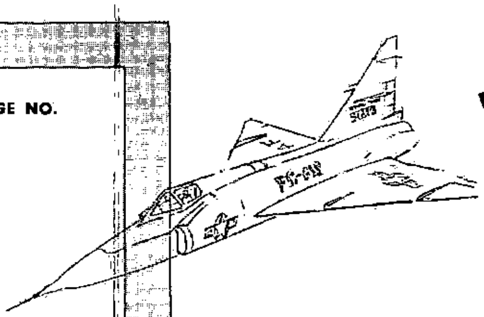

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introduction

This Technical Note presents an evaluation of the rolling characteristics of the F-102A with a 40% enlarged vertical stabilizer. These flight tests were conducted under the authority of the Commander, Air Force Flight Test Center, at the request of the Air Research and Development Command. The tests were accomplished at Edwards Air Force Base from 11 January to 24 February 1956. Fifteen flights totaling 12 hours and 5 minutes were required in the F-102A airplane USAF No. 53-1813 to complete the evaluation. The evaluation pilots were: J. S. Holtoner, Brig. General, USAF; H. A. Hanes, Colonel, USAF; R. M. Baker, Colonel, USAF; F. K. Everest, Lt. Colonel, USAF; R. L. Stephens, Major, USAF; S. R. Childs, Major, USAF; R. M. White, Captain, USAF.

The F-102A airplane USAF No. 53-1813 is a standard production F-102A except that it has a "glove fit" vertical stabilizer which increased the area from 68.33 to 95.14 square feet. The test airplane does not have a production "yoke" stick but the total aileron available is 14 degrees, the same as a production airplane. The test airplane was powered by a J57-P-41 trimmed-up to simulate a J57-P-23 engine. All other F-102A aircraft have J57-P-23 engines installed. The engine start gross weight of the airplane was 27,640 pounds with a c g of 29.6% MAC. The enlarged vertical stabilizer is quoted as increasing the airplane gross weight by 166 pounds. The control system is a standard F-102A control system which includes a yaw damper, and a turn coordinator. The pitch damper was installed but inoperative during these tests. Telemetering facilities were used to obtain airplane response and control position data and for ground monitoring during the test flights. Performance data was obtained by photopanel.

The limitations for this evaluation described by the contractor are as follows:

ROLL LIMITATIONS:

- **Abrupt roll maneuvers are limited to altitudes in excess of 12,000 feet.**
- **Roll maneuvers may be performed with maximum available aileron and to 2 to 3 degrees of critical elevator.**

AIRSPEED LIMITATIONS:

- **Maximum airspeed, clean configuration: 450 knots IAS or 1.25 Mach number, whichever occurs first.**
- **Maximum airspeed, gear down: 240 knots IAS.**

LOAD LIMITATIONS:

- **Normal load factor — Symmetrical +3.5g −1.8g — Asymmetrical +3.0g −1.0g.**
- **Lateral load factor −0.5g (This restriction may be lifted as data is obtained and analyzed).**

test results

roll evaluation

Inertial coupling effects in rolls have been reduced in the F-102A airplane with the enlarged vertical tail area so that no divergence in sideslip is encountered. Several two and three turn continuous rolls were made and the data shows the sideslip angle did not diverge during these rolls. The effect of elevator deflections were checked during the rolls and the increase in sideslip angles was negligible. The maximum sideslip angle encountered during these tests was 6.5 degrees at 15,000 feet and 0.62 Mach number.

The directional stability provided by the enlarged vertical tail is better than that of the standard F-102A with sideslip feed-back operating. At 35,000 feet and 0.77 Mach the test airplane rolled 185 degrees per second with 13 degrees of aileron and maximum sideslip of 6.0 degrees. The standard F-102A under similar flight conditions, with sideslip feed-back operating, rolled 170 degrees per second with 15 degrees of aileron and maximum sideslip angles of 5 to 9 degrees.

The contractor maintains that the structural loads incurred with a "glove fit" tail approach the limit loads in rolls with full aileron deflection at low altitudes and high airspeeds. An aileron restriction of 10 degrees instead of present 14 degree (total aileron deflection) has been recommended by the contractor to prevent conditions which might result in structural failure. A 10 degree aileron limitation has little effect on the roll capabilities at low airspeeds. In the power approach configuration at 180 knots, there is little decrease in roll rate experienced with decrease in aileron deflection from 14 degrees to 10 degrees. The time required to roll through 30 degrees of bank angle is .70 seconds with 14 degrees aileron and .83 seconds with 9 degrees aileron. The roll capabilities at higher speeds are reduced with a 10 degree aileron limitation so that at 15,000 feet and 0.53 Mach number the maximum roll rate is reduced from 150 degrees per second to 110 degrees per second. At 0.81 Mach number, the maximum roll rate is reduced from 205 degrees per second to 180 degrees per second. The time to roll through 100 degrees of bank angle using 13 degrees of aileron was 0.95 seconds at 15,000 feet and 0.62 Mach number. Under the same conditions of speed and altitude, 1.05 seconds were required for 100 degrees of bank angle using 10.2 degrees of aileron. The amount of aileron obtainable decreases with increase of IAS at low altitudes so that 14 degrees of aileron is available at 180 knots IAS and only 10.5 degrees is available at 480 knots. With a limitation of 10 degrees, the maximum aileron available at high speed would undoubtedly be reduced further. The aileron requirements for spin recovery or asymmetrical load conditions are unknown.

Presently being considered by the contractor is non-linear aileron force-deflection gradient such that the gradient is essentially constant to 10° aileron deflection and jumps to a higher constant gradient for the range of 10 to 14 degree total deflection.

Rolls were accomplished at 180 knots IAS with gear down at 12,000 feet. These rolls were made through approximately 120 degrees change in bank angle (60 to 60 degrees). The yaw damper and turn coordinator were operating during these rolls. In these rolls the roll rate tends to a constant value at large aileron angles possibly because of increasing sideslip angles (approaching five degrees) and loss of aileron effectiveness. The data is presented in Figure 1, Appendix I, and the roll data is listed in Table 1 of Appendix I.

Rolls were accomplished in the cruise configuration at 15,000 feet in the speed range of 0.53 to 0.95 Mach number. Rolls were made through 120, 360 and 720 degrees change in bank angle. The 720 degree rolls were made to determine if any adverse effects occurred in continuous rolls. The roll time histories in Figures 6 through 8, Appendix I show that the sideslip angle reached a peak in approximately 1 second and did not reach a steady state response at time of recovery. In most 360 degree rolls the sideslip angle reached a peak in 1 second and a second slightly higher peak occurred during recovery. The maximum sideslip angle obtained during all of the roll tests was 6.5 degrees in a two turn roll with 175 degrees per second rate of roll at 0.62 Mach number and 15,000 feet altitude. Elevator deflections of 2.5 degrees up to 1.25 degrees down were made during the rolls. The effects on sideslip angle were not noticeable. The roll characteristics are presented in Figure 2, Appendix I. The sideslip angles are negligible in rolls using up to 10 degrees of aileron deflection and vary from 1 to 6.5 degrees with 10 to 14 degrees of aileron deflection. These rolls are presented in tabular form in Table 2, Appendix I.

Rolls were accomplished at 0.77, 0.92 and 1.12 Mach number at 35,000 feet from wings level through 360 degrees or more. Time histories of some typical rolls are presented in Figures 9 through 13, Appendix I. These rolls show the characteristic of the sideslip angle is not divergent during one to four continuous rolls. This data shows that the inherent stability of the aircraft is sufficient to be free of inertial coupling dangers. Elevator deflections of 2.5 degrees up (A/C nose up) to 1.4 degrees down were made during these tests and the data did not show any appreciable increase in sideslip angle. The roll characteristics are presented in Figures 3 through 5, and Table 3, Appendix I.

dynamic lateral-directional stability

The dynamic lateral-directional stability characteristics of this airplane are excellent with the yaw damper and turn coordinator operating. It requires 1.0 to 1.5 cycles for the oscillations to completely damp out. With the yaw damper off the oscillations damp out very slowly and do not meet Air Force Specifications, although the conditions encountered are not dangerous. The tests were accomplished at 35,000 feet at 0.77, 0.92 and 1.12 Mach numbers. The data is presented as time histories in Figures 14 through 16, Appendix I.

static directional stability

The airplane has satisfactory static directional stability. It requires right rudder and left aileron to produce a left sideslip. The static directional gradient of this airplane with the enlarged tail is about 2 degrees of rudder per 1 degree of sideslip and the rudder area is 1130 square inches. The YF-102A had a rudder area of 1383 square inches and 1 degree of rudder produced 1 degree of sideslip. The rudder area of the standard F-102A is 1070 square inches.

The static directional tests were conducted at altitudes of 12,000 and 35,000 feet with the yaw damper and turn coordinator operating. The pilot attempted to obtain full rudder deflection during the tests but was not able to because the "q" box is scheduled to limit rudder deflection with "q". The data is presented in Figures 17 and 18, Appendix I. These plots show the rudder effectiveness and the maximum rudder deflections for the speeds presented.

performance evaluation

The climb performance of the test aircraft is essentially the same as obtained on the Phase II YF-102A airplane. The effects of the large tail on climbs are within the limit of test accuracy. Two check climbs at maximum power were made to altitudes above 53,000 feet. The data obtained below 45,000 feet is not valid for comparison purposes because the speed schedule as flown at lower altitudes was much greater than the recommended climb speed schedule. The climb data is corrected to the same standard thrust used for the Phase II tests (AFFTC-TR-55-31). The data is presented in Figure 19, Appendix I, with the Phase II data included as dashed lines for comparison.

Maximum level flight speeds obtained at 37,000, 44,000 and 51,000 feet indicated a very slight change from those of the aircraft with a small tail. The differences, however, are within the limits of test accuracy. This data is presented in Figures 20 and 21 as drag versus Mach number and drag coefficient versus Mach number.

The engine used for these tests was a J57-P-41 trimmed-up to simulate a J57-P-23 engine, as was done in the Phase II program. The static thrust run indicates slightly less thrust at maximum and military power for static conditions than was obtained during the Phase II program. The thrust coefficient is also less than the earlier test and is presented in Figure 22, Appendix I.

conclusions

It is concluded that:

- **No roll limitations are required because of inertial coupling effects.**
- **The roll capabilities with the increased area tail meet the roll requirements to perform operational and non-operational type maneuvers.**
- **The directional and roll-coupling stability of the aircraft has been improved; however, the requirement for the yaw damper and turn coordinator still exists.**
- **The performance of the airplane has not been appreciably affected as there is negligible change in ceiling and maximum speed.**
- **A restriction of 10 degrees (total) aileron deflection has been requested by the contractor to prevent structural failure of the enlarged tail. This restriction is acceptable for normal operation providing the pilot is able to obtain 14 degrees (total) aileron deflection for emergency conditions. At present the aileron requirements for spin recovery and asymmetric load conditions are unknown.**

recommendations

It is recommended that:

- **The increased area tail be included on all F-102A aircraft.**
- **This change be incorporated prior to delivery of any F-102A aircraft to using agencies.**
- **Spin requirements and asymmetric conditions be investigated as soon as possible to determine the aileron requirements for all flight conditions.**
- **An investigation be made to determine the aileron restriction with "q" and Mach number to satisfy structural limits.**

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discussion

The elevons of the F-102A airplane perform a dual function by operating differentially as ailerons and in unison as elevators. Total aileron was computed by taking the algebraic summation of the elevon positions with down elevon considered negative. The direction of aileron (left or right) was chosen according to the direction it tends to make the aircraft roll. The time history plots in this Appendix have been presented as one half of the total aileron displacement. It is not the trace of a single elevon.

Elevator displacement is the average of the two elevons with up deflection tending to cause the aircraft to pitch up. The elevator deflections presented in the time history plots in this Appendix show the elevator deflections from trim and are not absolute values of elevator position.

The sideslip angle, as presented, indicates the opposite of yaw angle as measured with a yaw vane installed on the nose boom.

The climb and level flight performance data were corrected to standard conditions in the same manner as outlined in AFFTC-TR-55-31, the Phase II report on the YF-102A. The engine used for these tests (also for the YF-102A) was a J57-P-41, trimmed-up to simulate a J57-P-23 engine.

FIGURE 1 AILERON ROLL CHARACTERISTICS

POWER APPROACH CONFIG.
ALTITUDE 12,000 FT.

⊙ 180 KNOTS, IAS

60 TO 60 DEGREE ROLLS

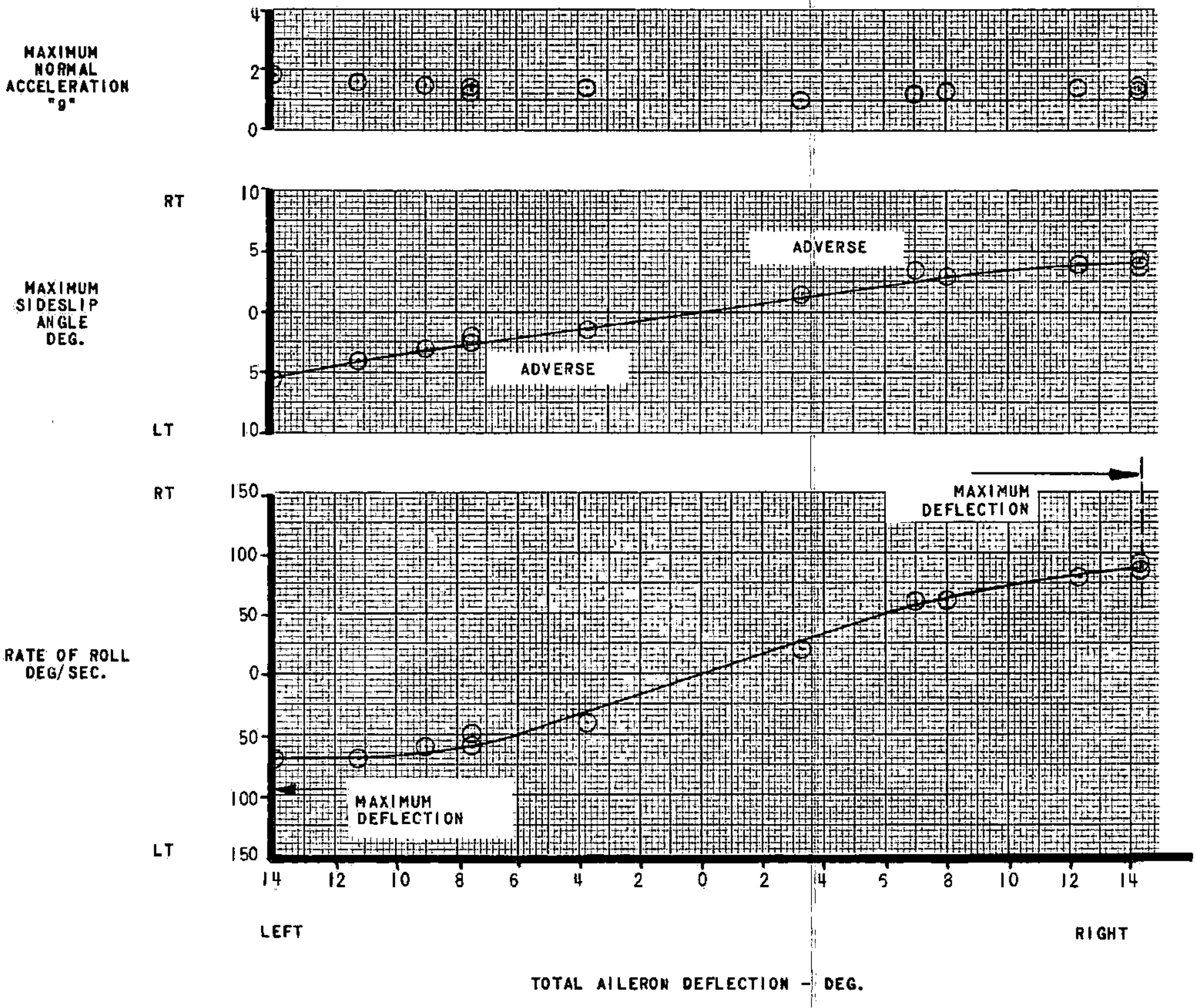


FIGURE 2 **AILERON ROLL CHARACTERISTICS**

CRUISE CONFIGURATION
ALTITUDE 15,000 FT.

	V_c - KNOTS	MACH
◇	492	.95
⊙	415	.81
⊡	314	.62
△	266	.53

1 TURN ROLL - DOT IN SYMBOL
2 TURN ROLL - VERTICAL LINE THRU SYMBOL

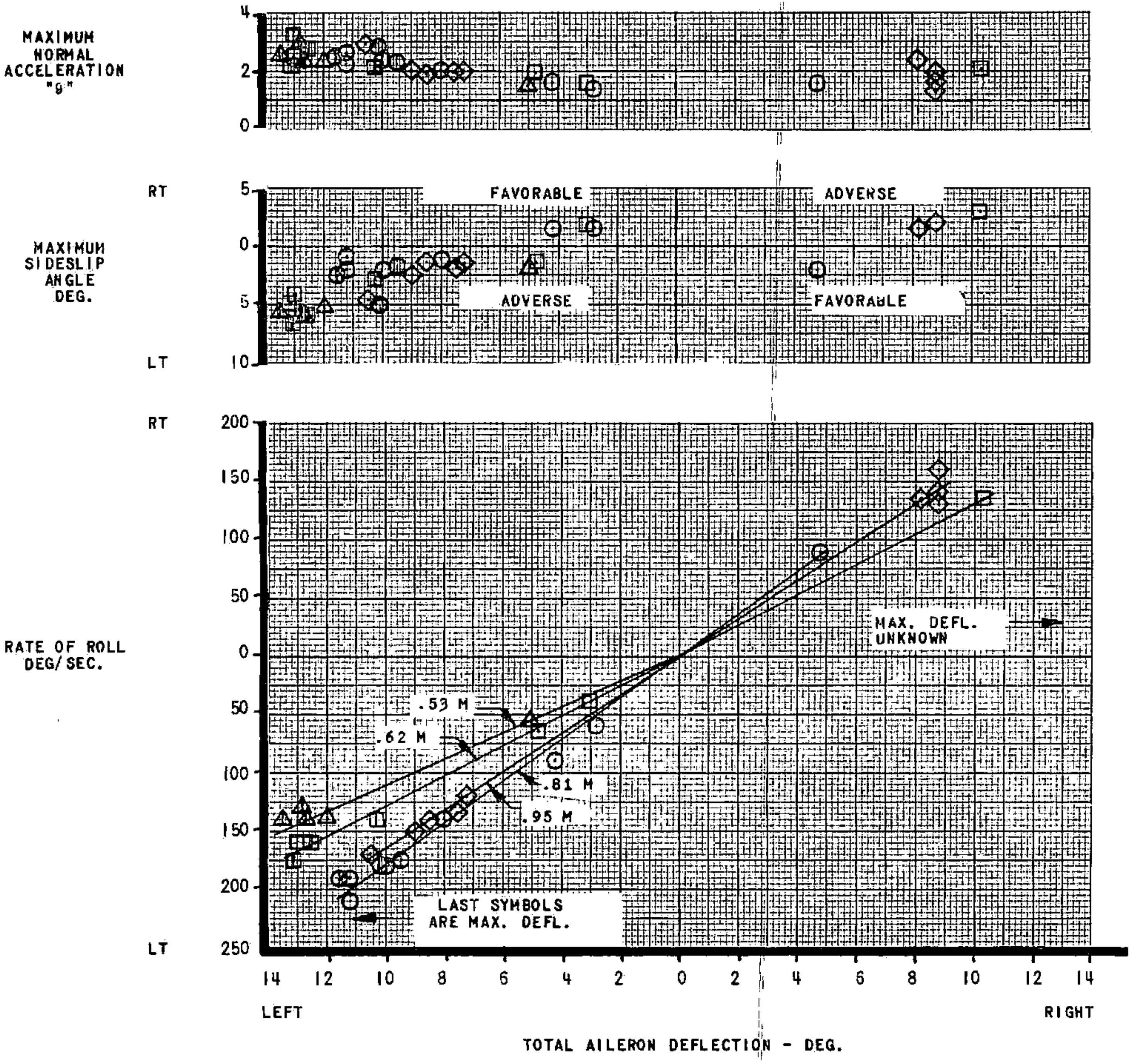


FIGURE 3

AILERON ROLL CHARACTERISTICS

CRUISE CONFIGURATION

⊙ 1 TURN ROLL
⦶ 2 TURN ROLL
⊕ 3 TURN ROLL
● 4 TURN ROLL

MACH NO. 77
ALTITUDE 35,000 FT.
V_c 261 KNOTS

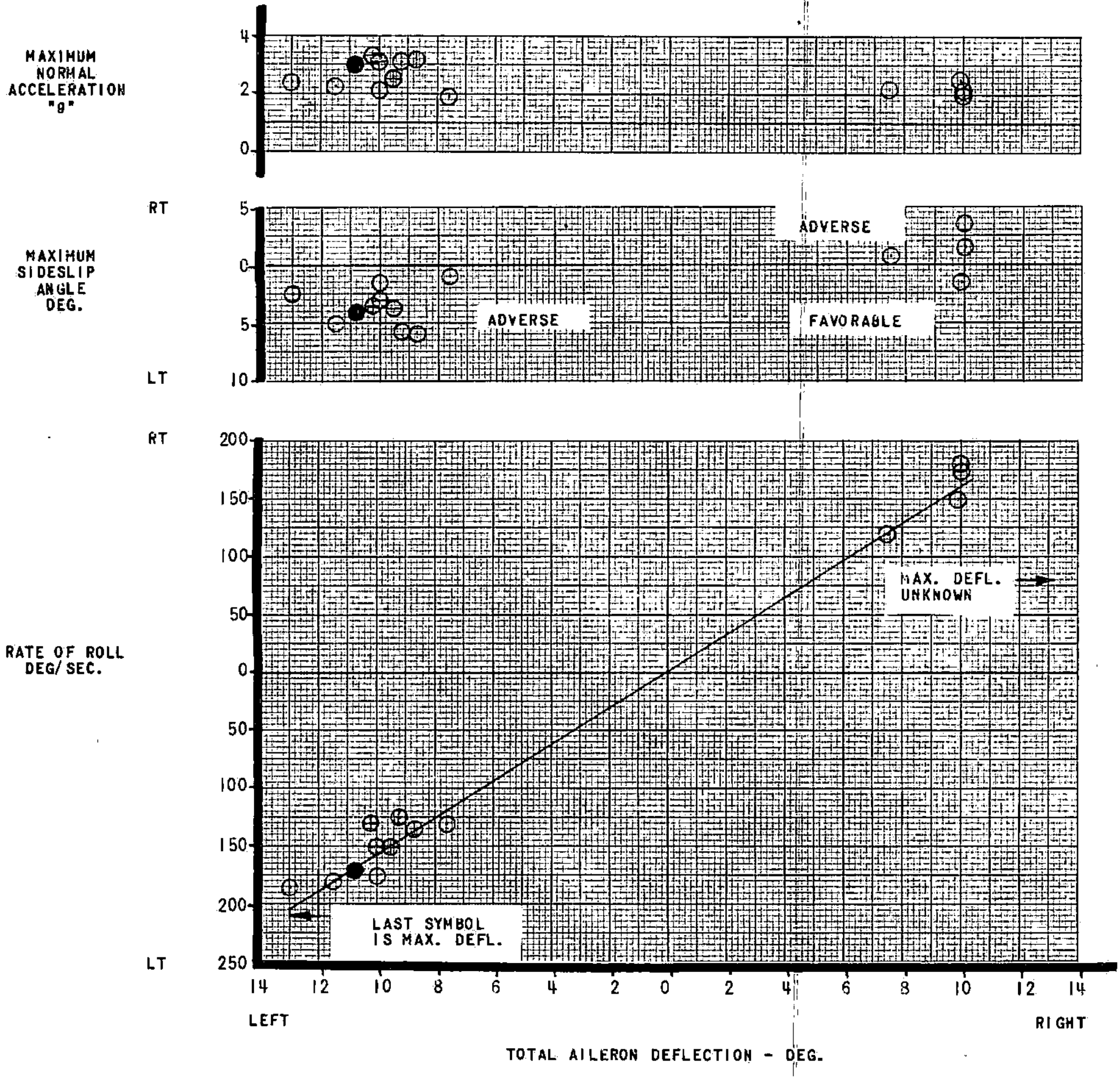


FIGURE 4 AILERON ROLL CHARACTERISTICS

CRUISE CONFIGURATION

⊙ 1 TURN ROLL
⊕ 3 TURN ROLL

MACH. NO. 92
ALTITUDE 35,000 FT.
V_c 320 KNOTS

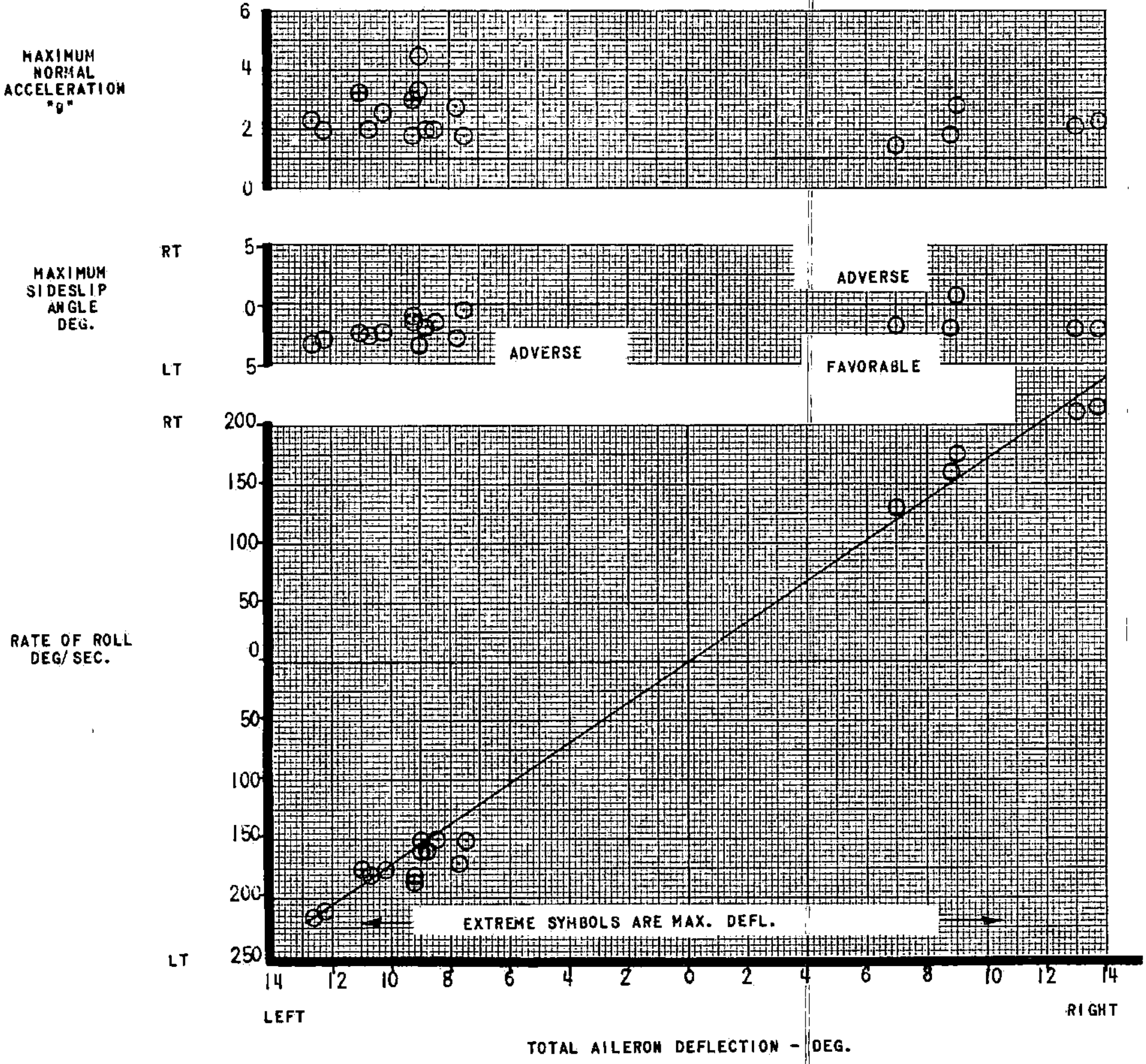


FIGURE 5

AILERON ROLL CHARACTERISTICS

COMBAT CONFIGURATION

⊙ 1 TURN ROLL
⦶ 2 TURN ROLL

MACH. NO. 1.12
ALTITUDE 35,000 FT.
V_c 400 KNOTS

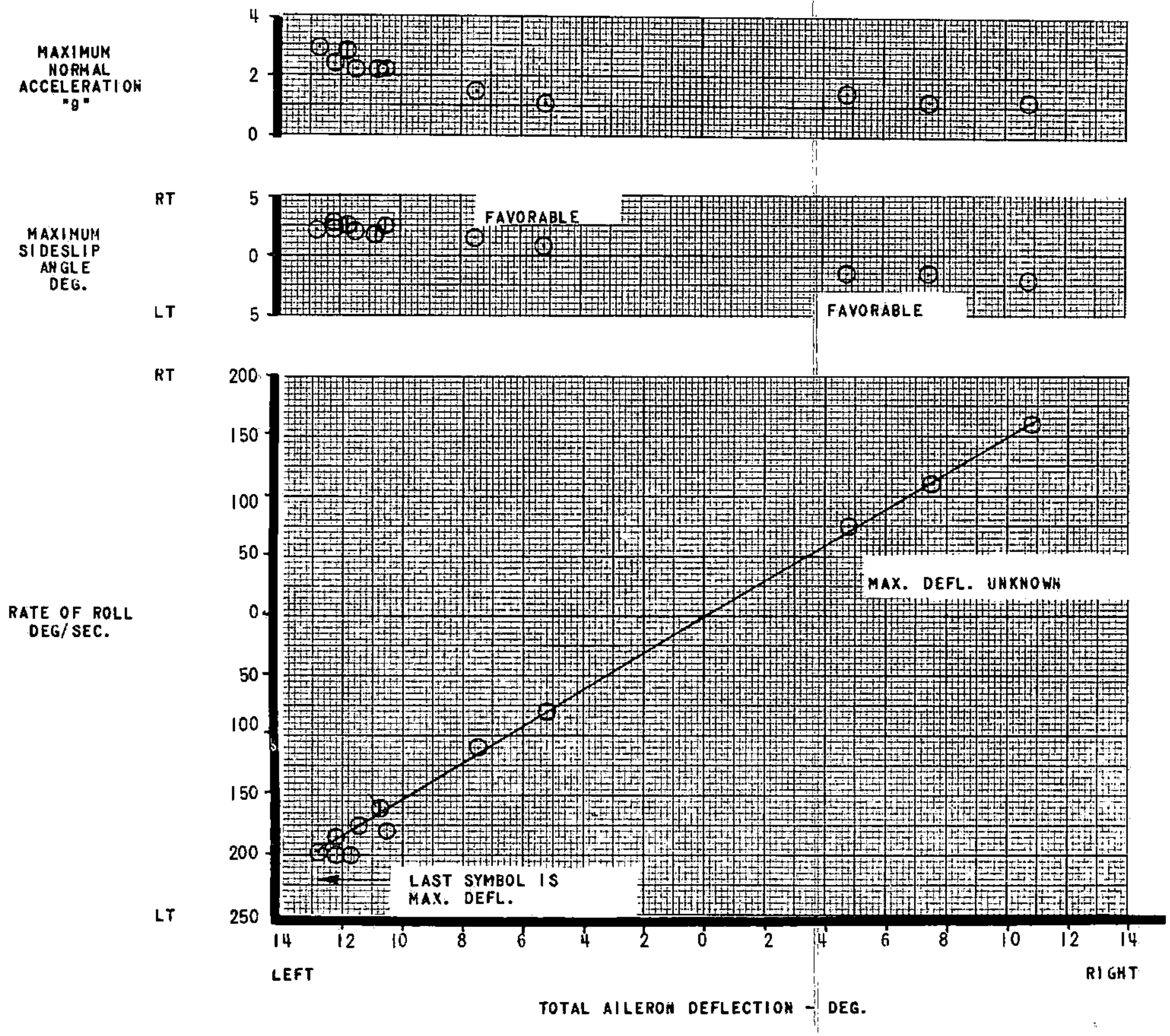
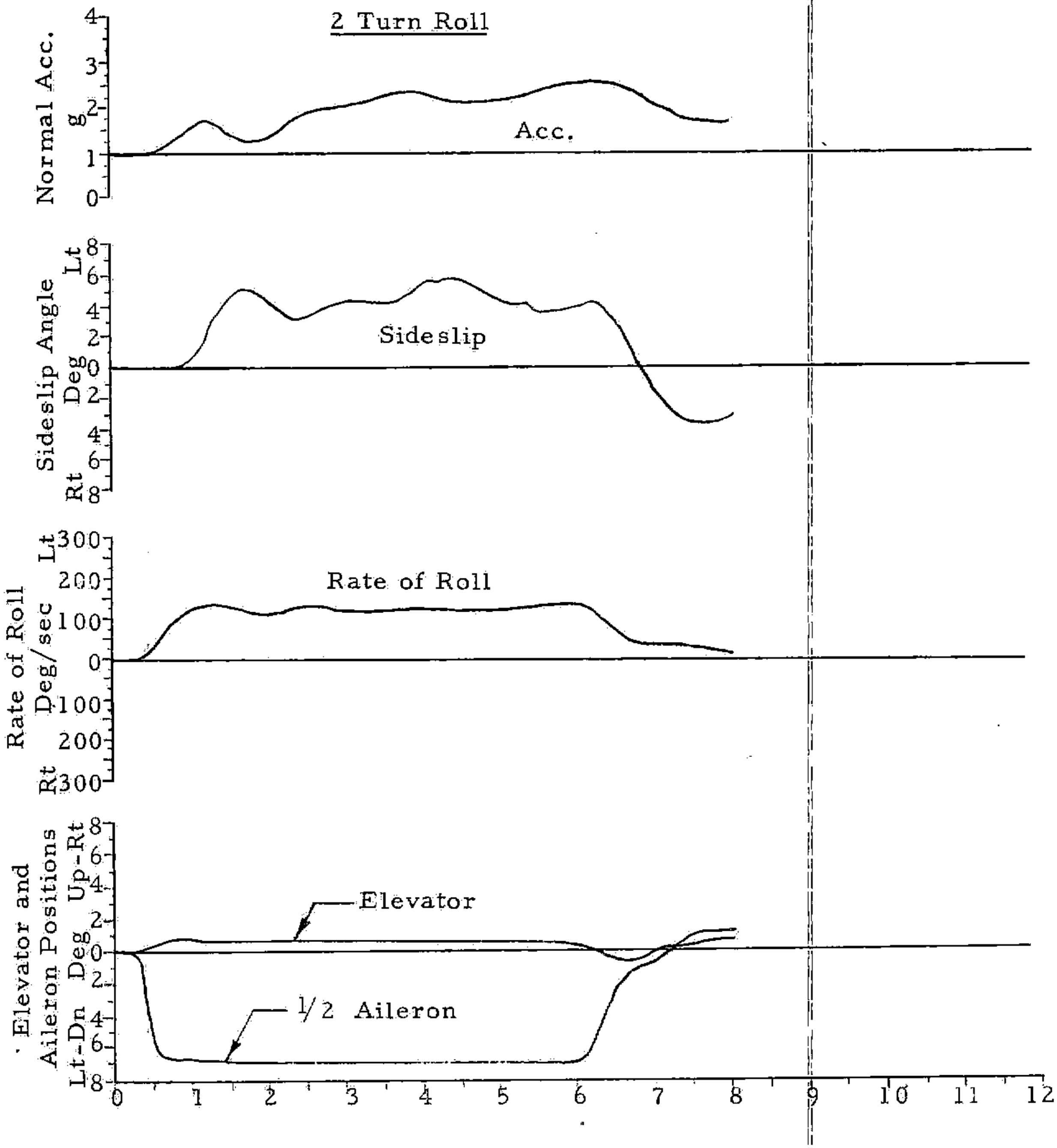


Figure No. 6
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.50
Alt 14,800 Ft.
CAS 253 Knots
Yaw Damper On



Time seconds

Figure No. 7
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.50
Alt 14,800 Ft.
CAS 253 Knots
Yaw Damper On

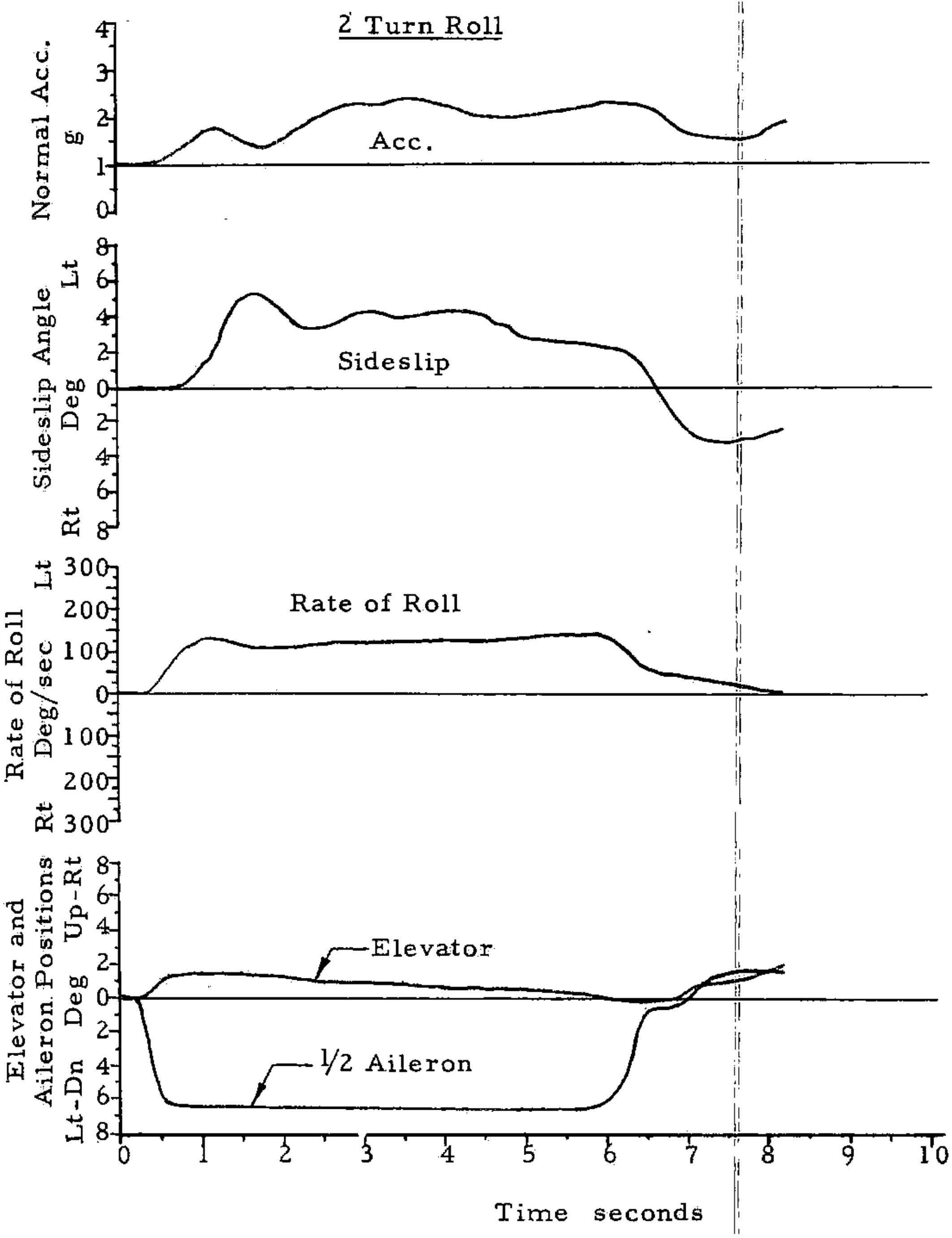


Figure No. 8
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.82
Alt 14,500 Ft.
CAS 410 Knots
Yaw Damper On

Mach No. 0.61
Alt 14,100 Ft.
CAS 309 Knots
Yaw Damper On

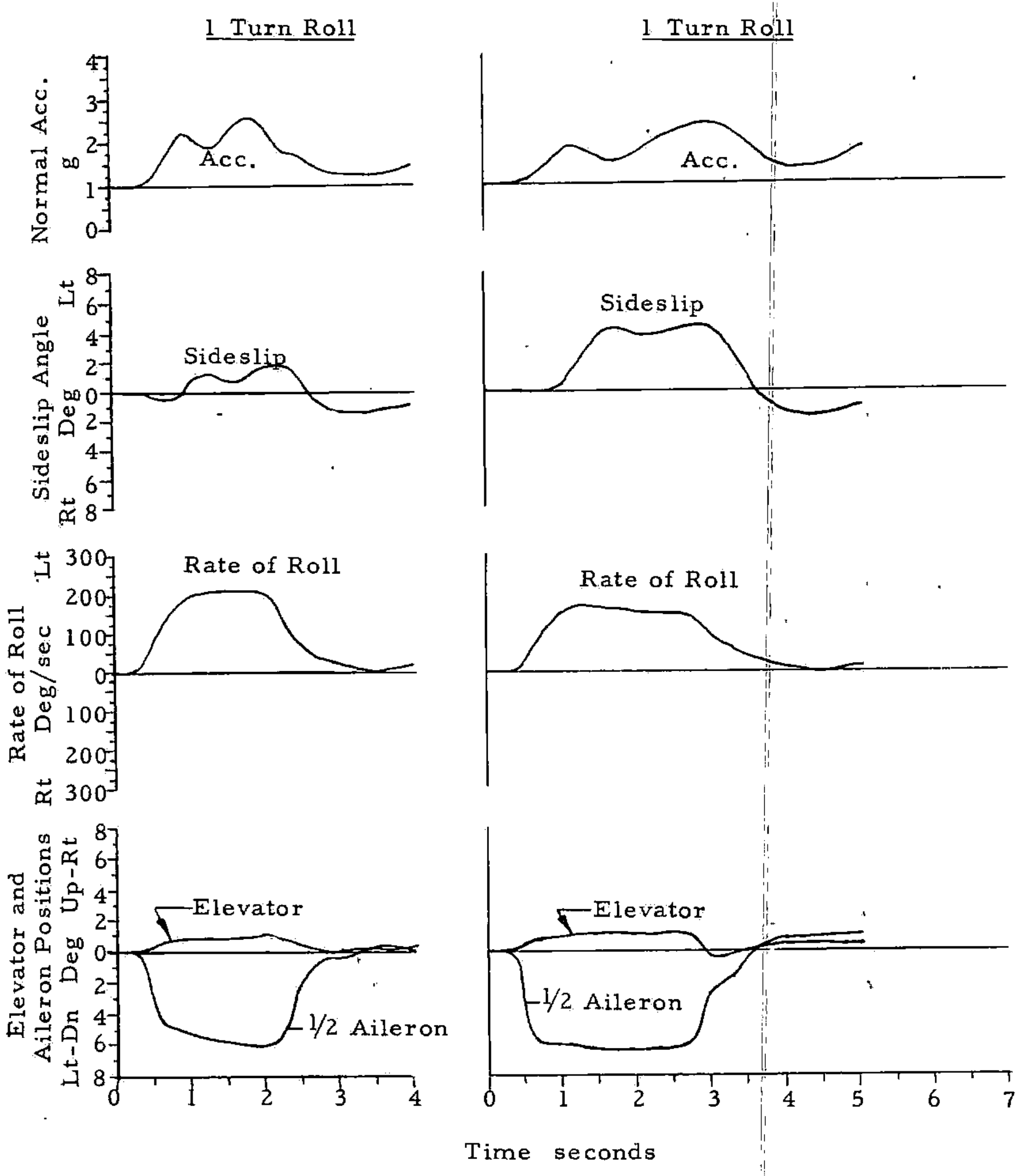


Figure No. 9
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.77
Alt 34,100 Ft.
CAS 265 Knots
Yaw Damper On

Mach No. 0.77
Alt 33,600 Ft.
CAS 269 Knots
Yaw Damper On

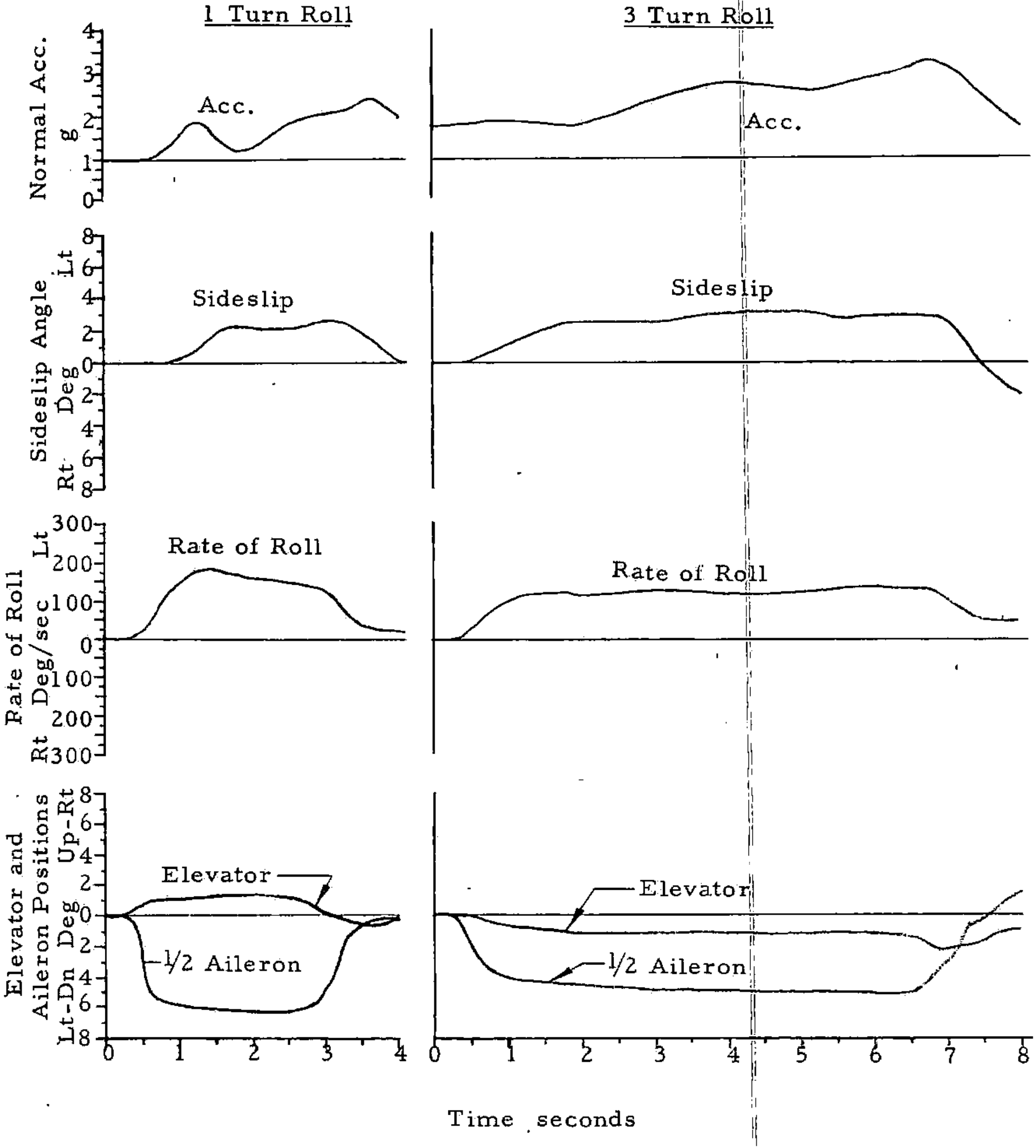


Figure No. 10
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.77
Alt 34,900 Ft.
CAS 260 Knots
Yaw Damper On

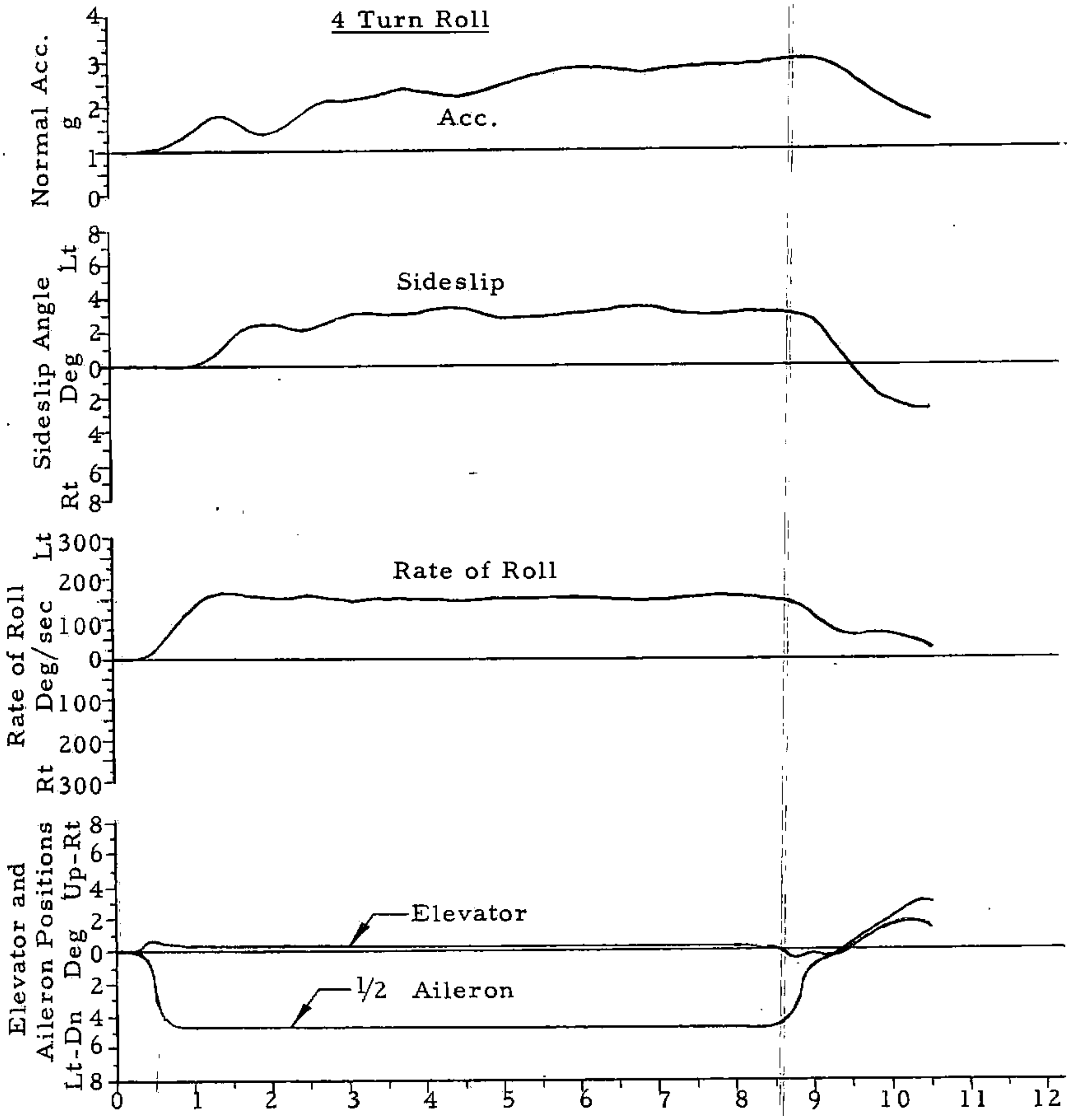
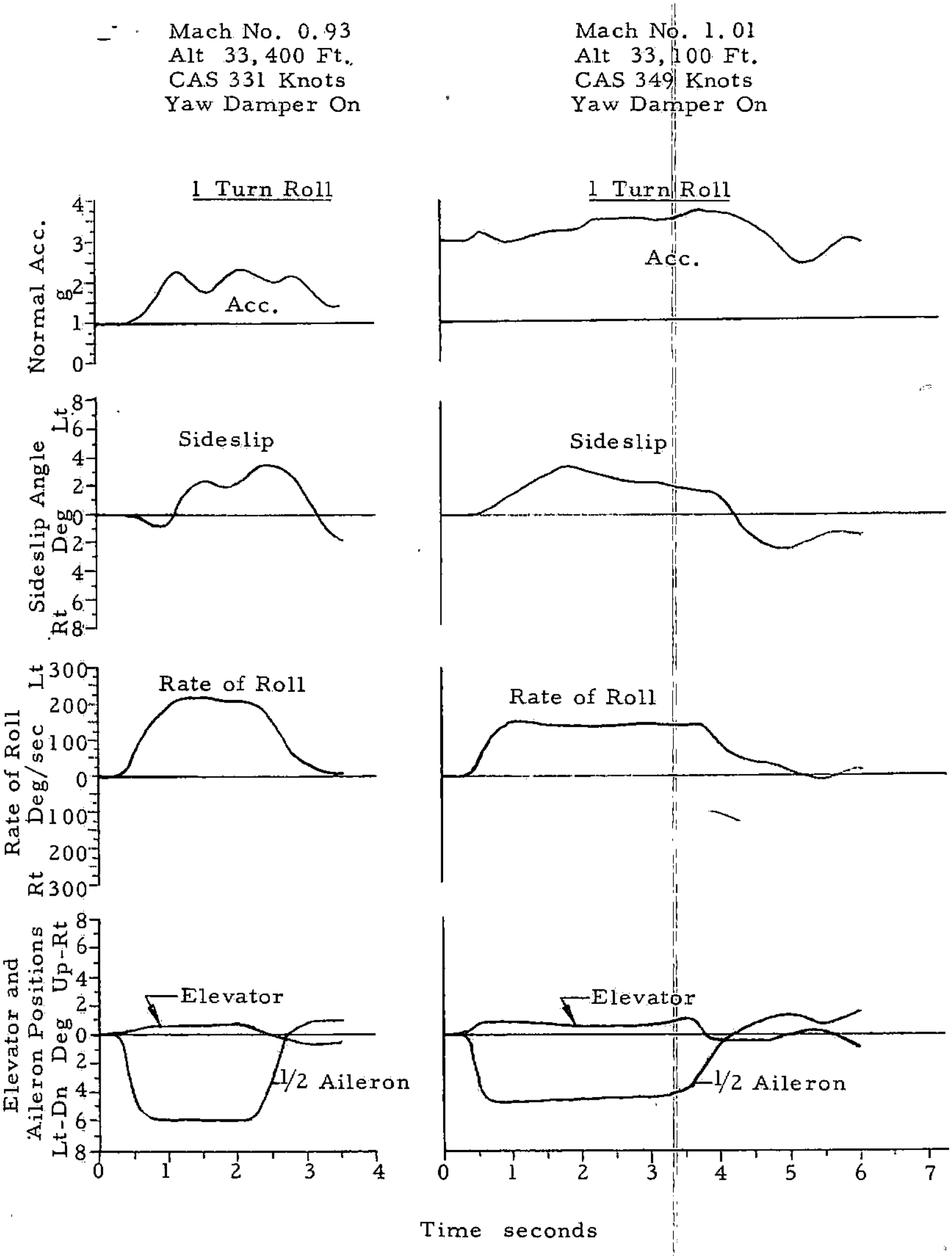


Figure No. 11
AILERON ROLLS
F-102A USAF No. 53-1813
Cruise Configuration



Mach No. 0.93
Alt 33,400 Ft.
CAS 331 Knots
Yaw Damper On

Mach No. 1.01
Alt 33,100 Ft.
CAS 349 Knots
Yaw Damper On

Figure No. 12
AILERON ROLLS
F-102A USAF No. 53-1813
Combat Configuration

Mach No. 1.13
Alt 34,600 Ft.
CAS 409 Knots
Yaw Damper On

Mach No. 1.15
Alt 34,600 Ft.
CAS 417 Knots
Yaw Damper Off

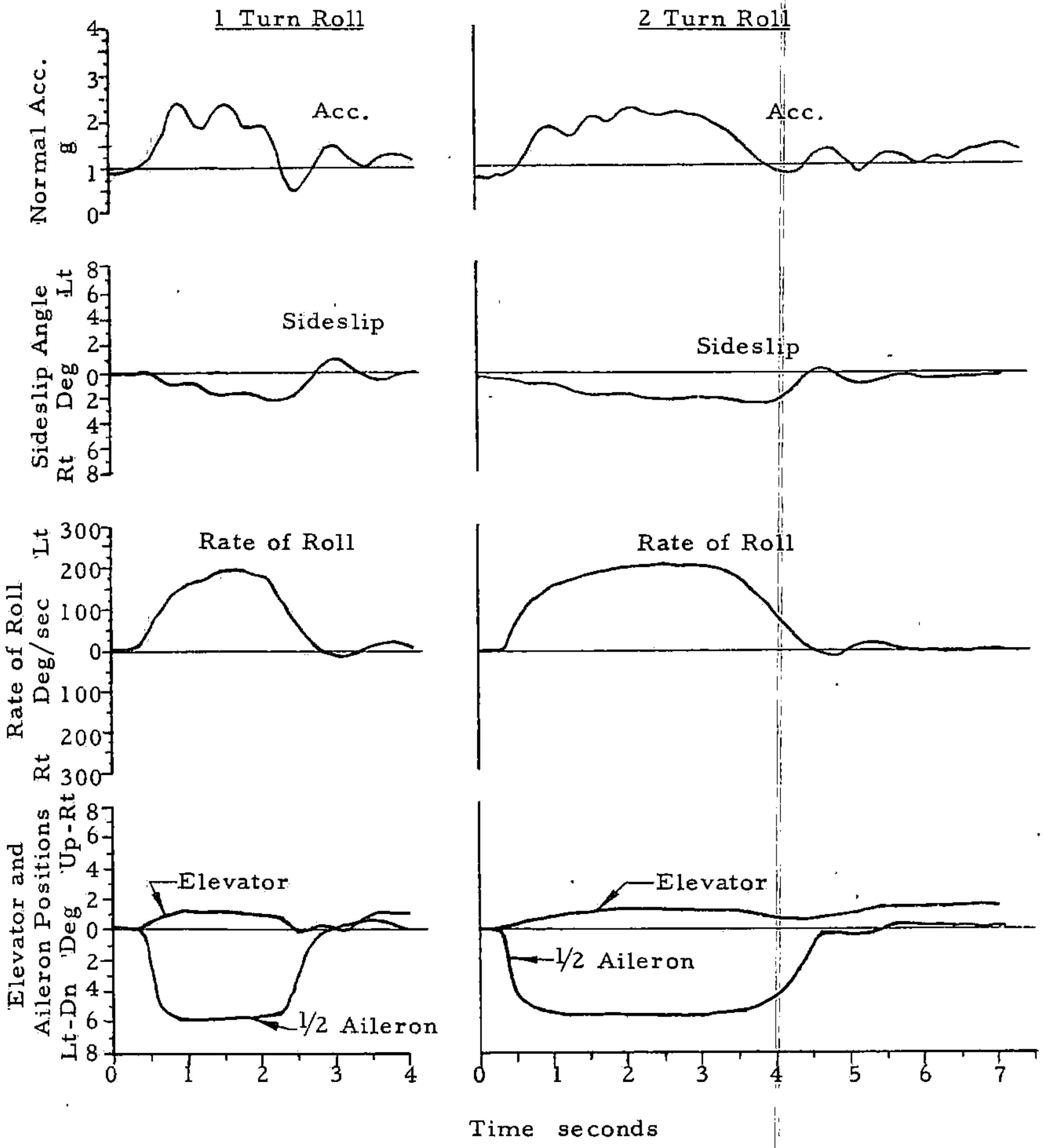
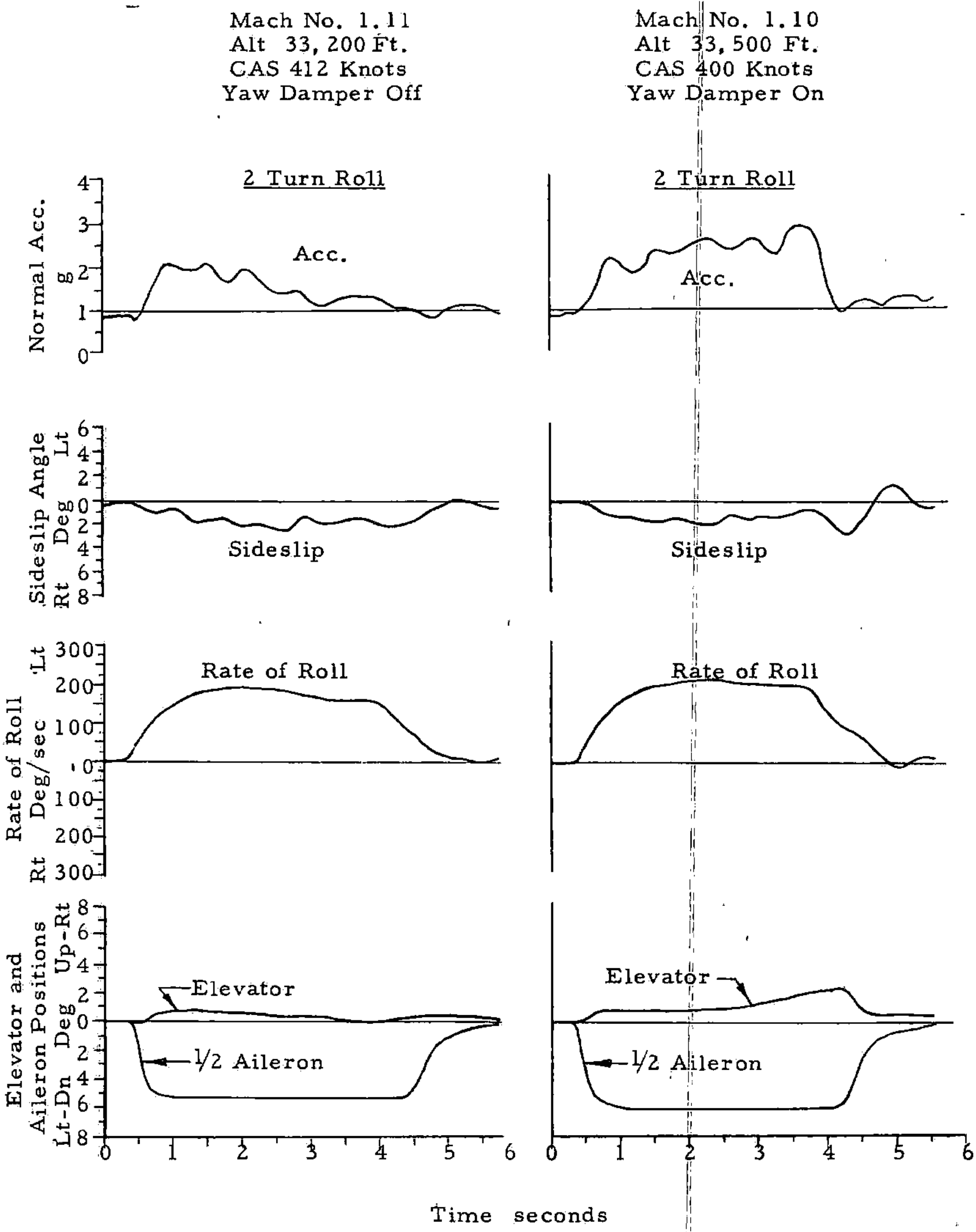


Figure No. 13
AILERON ROLLS
F-102A USAF No. 53-1813
Combat Configuration



Mach No. 1.11
Alt 33,200 Ft.
CAS 412 Knots
Yaw Damper Off

Mach No. 1.10
Alt 33,500 Ft.
CAS 400 Knots
Yaw Damper On

Figure No. 14
DYNAMIC LATERAL-DIRECTIONAL STABILITY
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.77 Altitude 35,000 Ft.

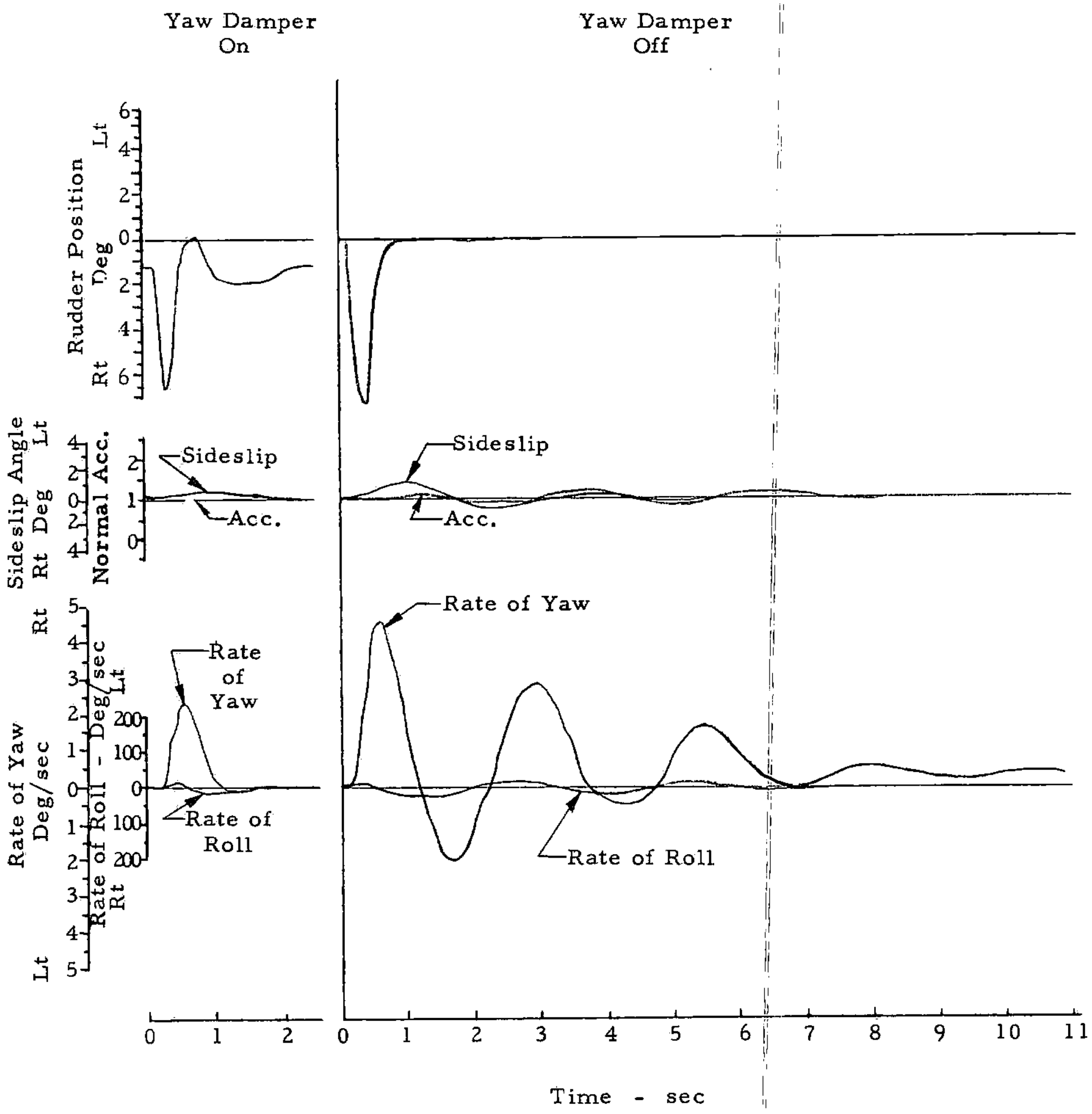


Figure No. 15
DYNAMIC LATERAL-DIRECTIONAL STABILITY
F-102A USAF No. 53-1813
Cruise Configuration

Mach No. 0.93 Altitude 35,000 Ft.

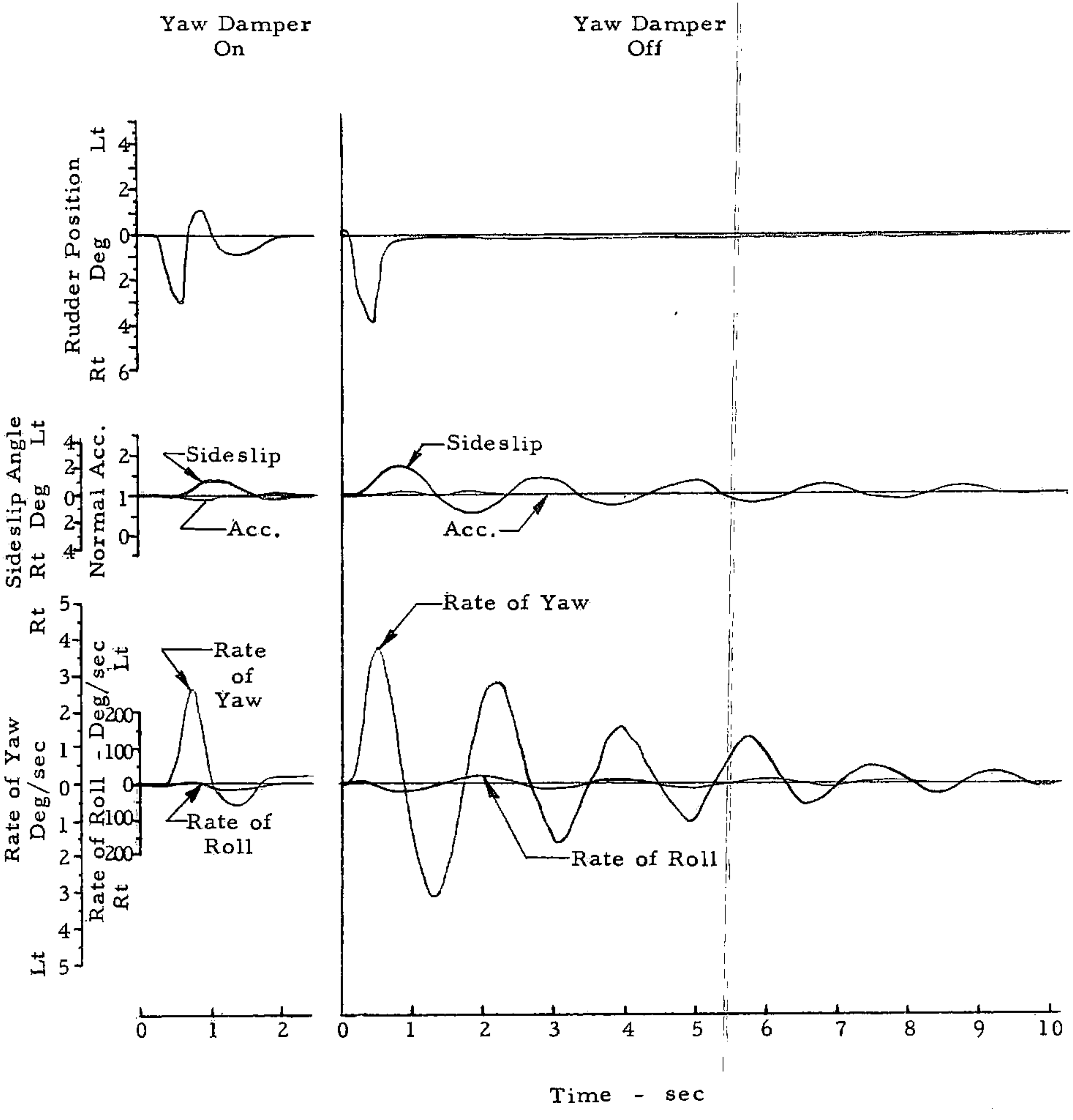


Figure No. 16
DYNAMIC LATERAL-DIRECTIONAL STABILITY
F-102A USAF No. 53-1813
Combat Configuration

Mach No. 1.16 Altitude 35,000 Ft.

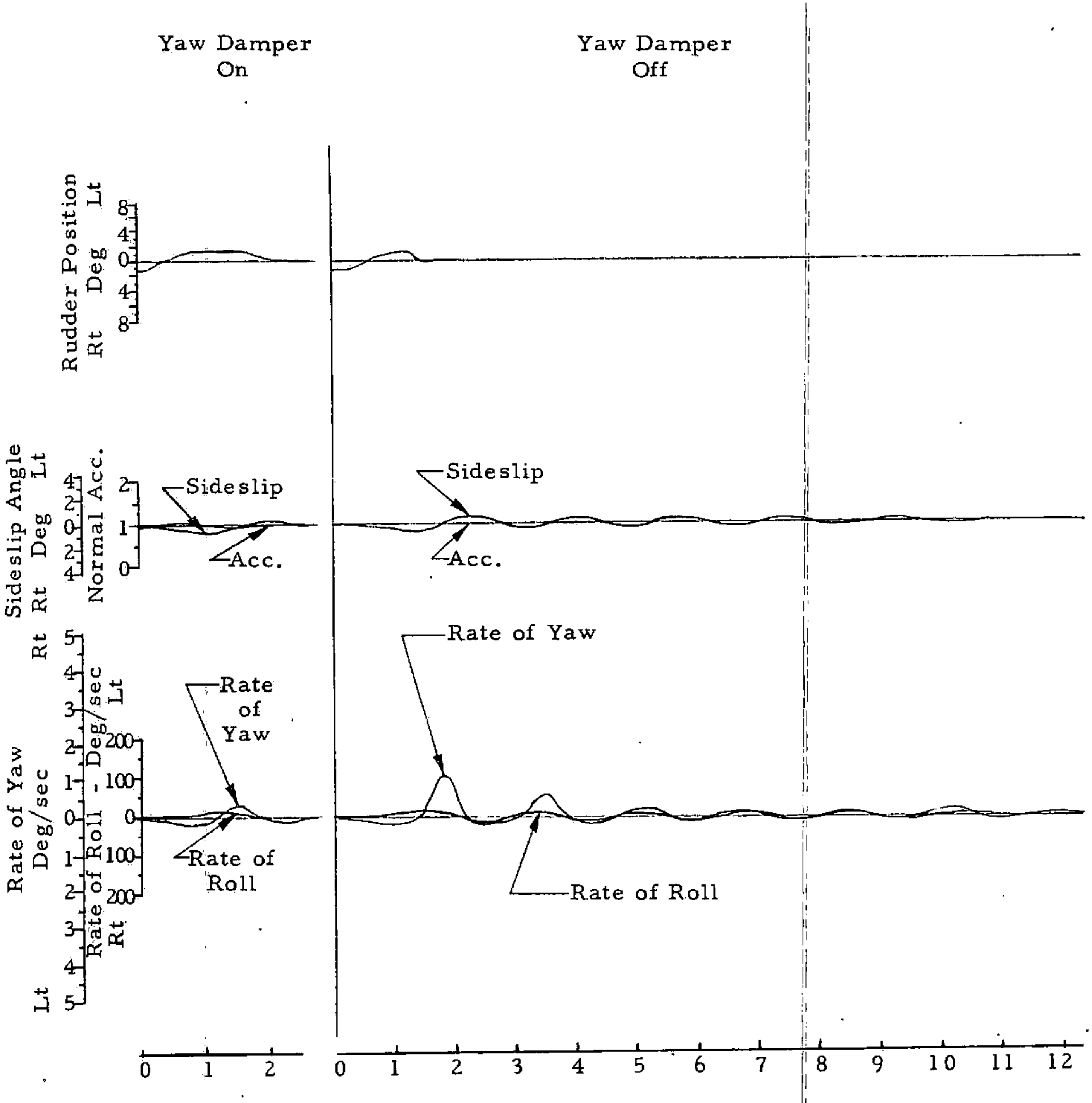


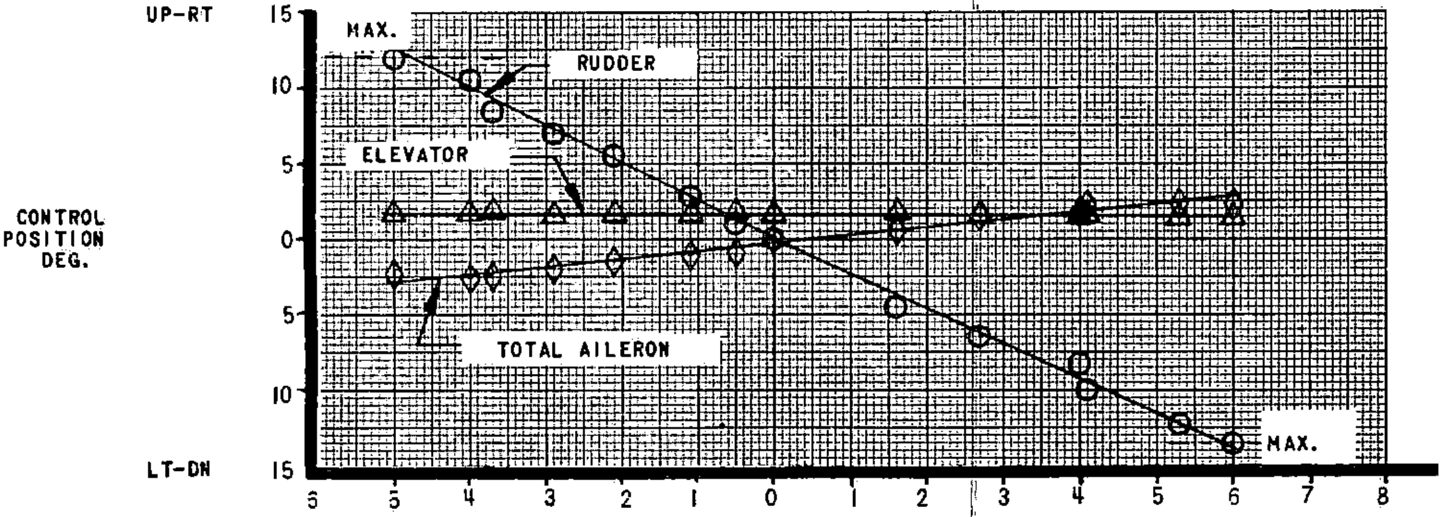
FIGURE 17

STATIC DIRECTIONAL STABILITY

TURN COORDINATOR AND YAW DAMPER ON

CRUISE CONFIG.
IAS 255 KNOTS
ALT. 34,700 FT.
MACH NO. 0.77

NOTE: MAXIMUM RUDDER DEFLECTIONS ATTAINABLE WERE LIMITED BY A "q" BOX



POWER APPROACH CONFIG.
IAS 180 KNOTS
ALT. 12,000 FT.

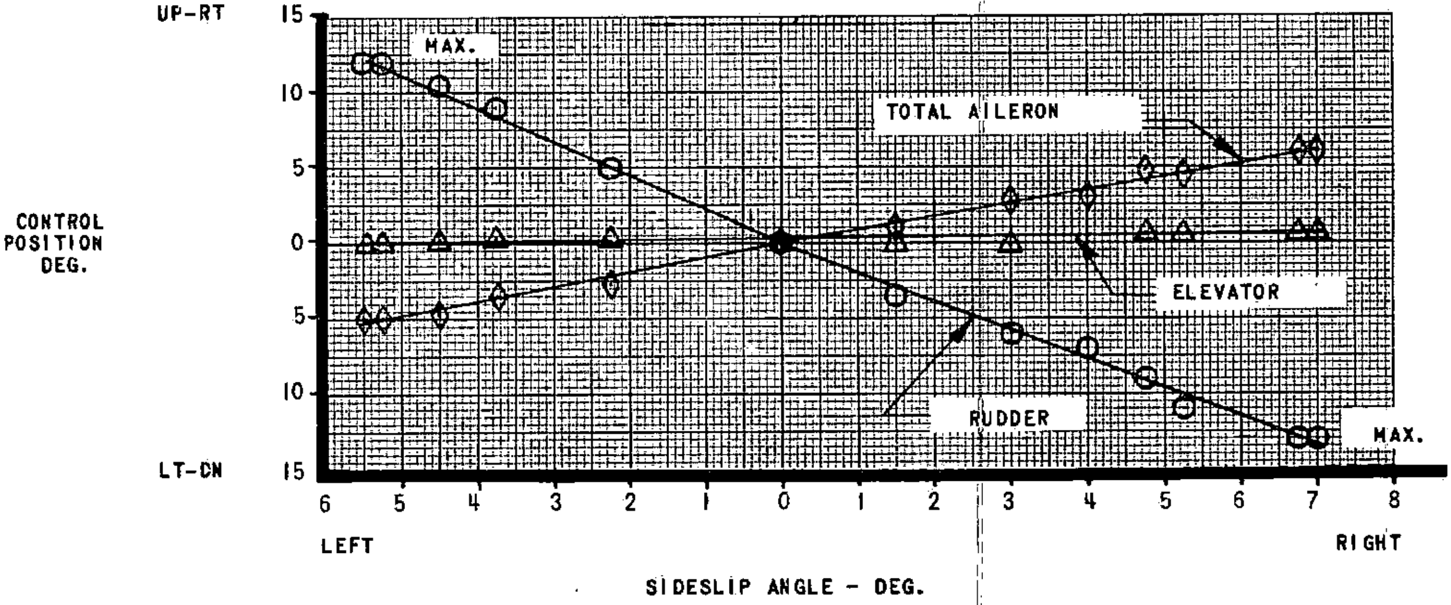
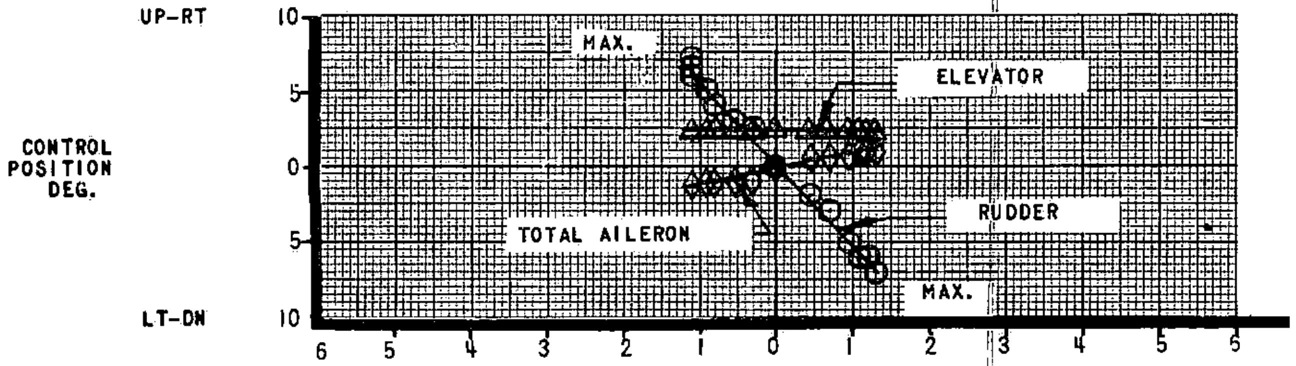


FIGURE 18 STATIC DIRECTIONAL STABILITY

TURN COORDINATOR AND YAW DAMPER ON

COMBAT CONFIG.
IAS 426 KNOTS
ALT. 35,300 FT.
MACH. NO. 1.18

NOTE: MAXIMUM RUDDER DEFLECTIONS ATTAINABLE WERE LIMITED BY A "q" BOX



CRUISE CONFIG.
IAS 312 KNOTS
ALT. 35,000 FT.
MACH. NO. 0.91

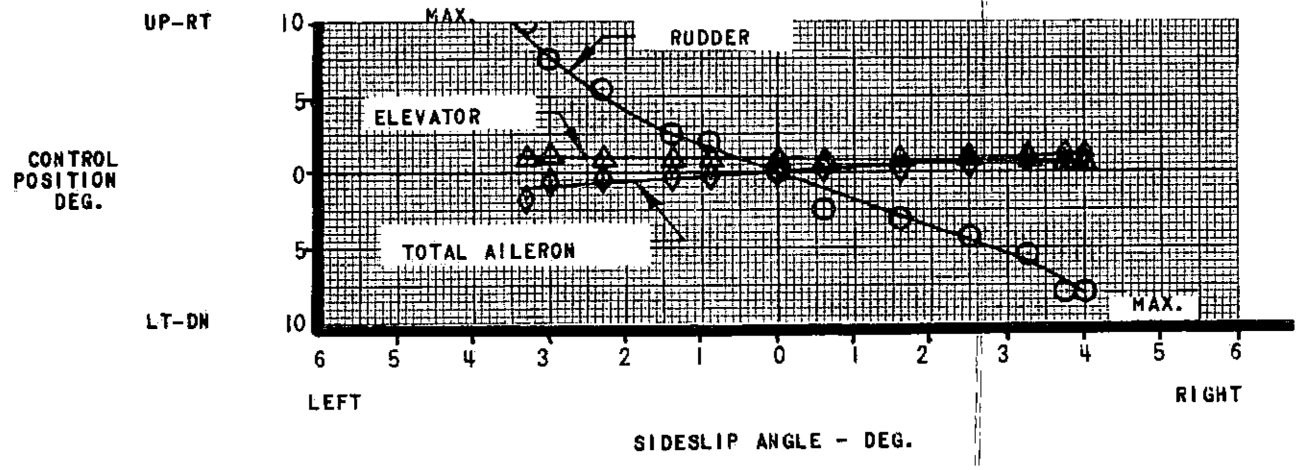


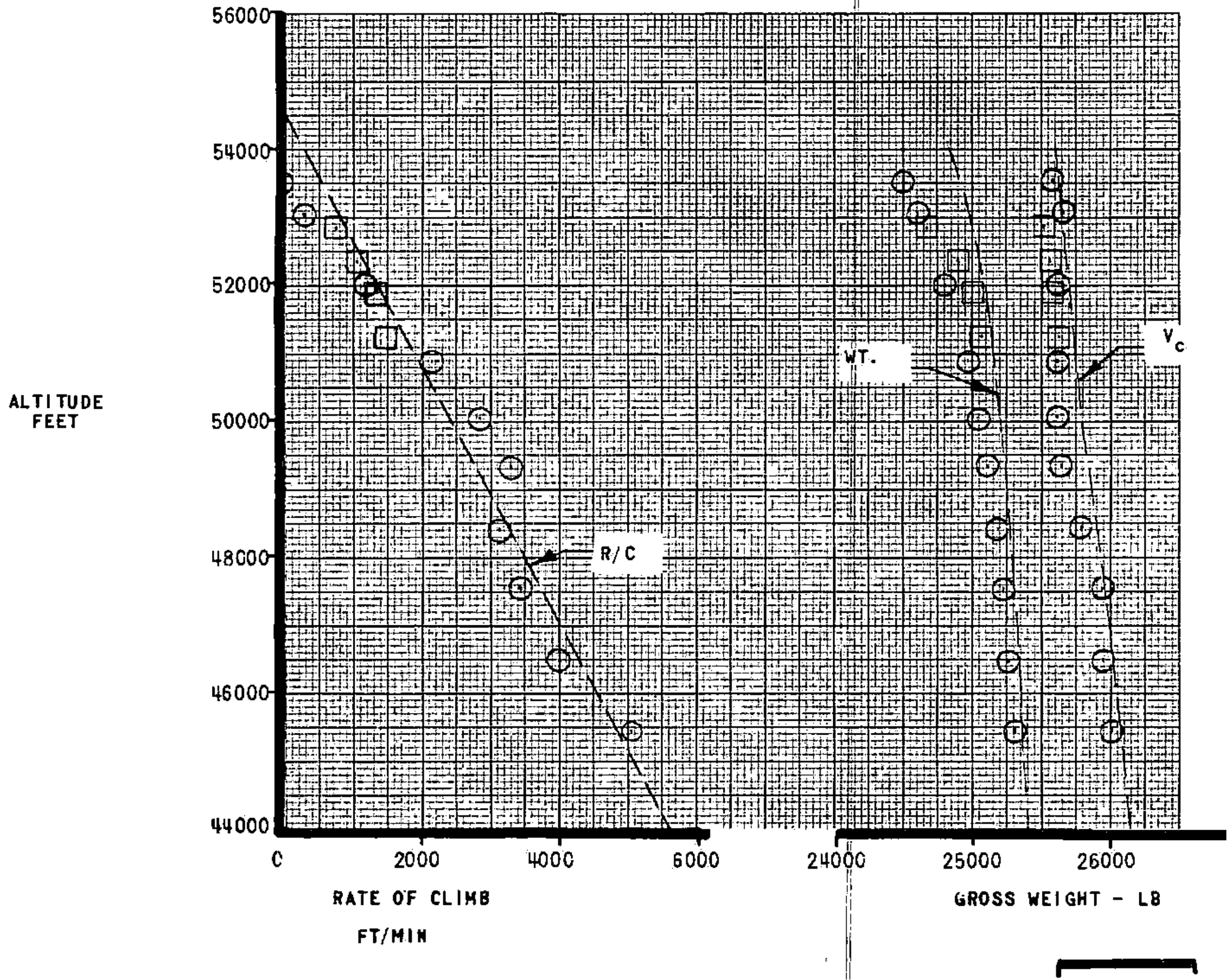
FIGURE 19

CLIMB PERFORMANCE

MAXIMUM POWER

J57-P-41 ENG S/N 604217
A/B S/N 600563
TRIMMED TO SIMULATE A
J57-P-23 ENG.

NOTE: DASHED LINES ARE DATA FROM PHASE II YF-102A, AFFTC-TR-55-31



NOTE: DATA NOT PRESENTED AT LOWER ALT. BECAUSE CLIMBS WERE NOT FLOWN AT CHECK CLIMB SPEED SCHEDULE

FIGURE 20

LEVEL FLIGHT PERFORMANCE

CORRECTED THRUST vs MACH NO.

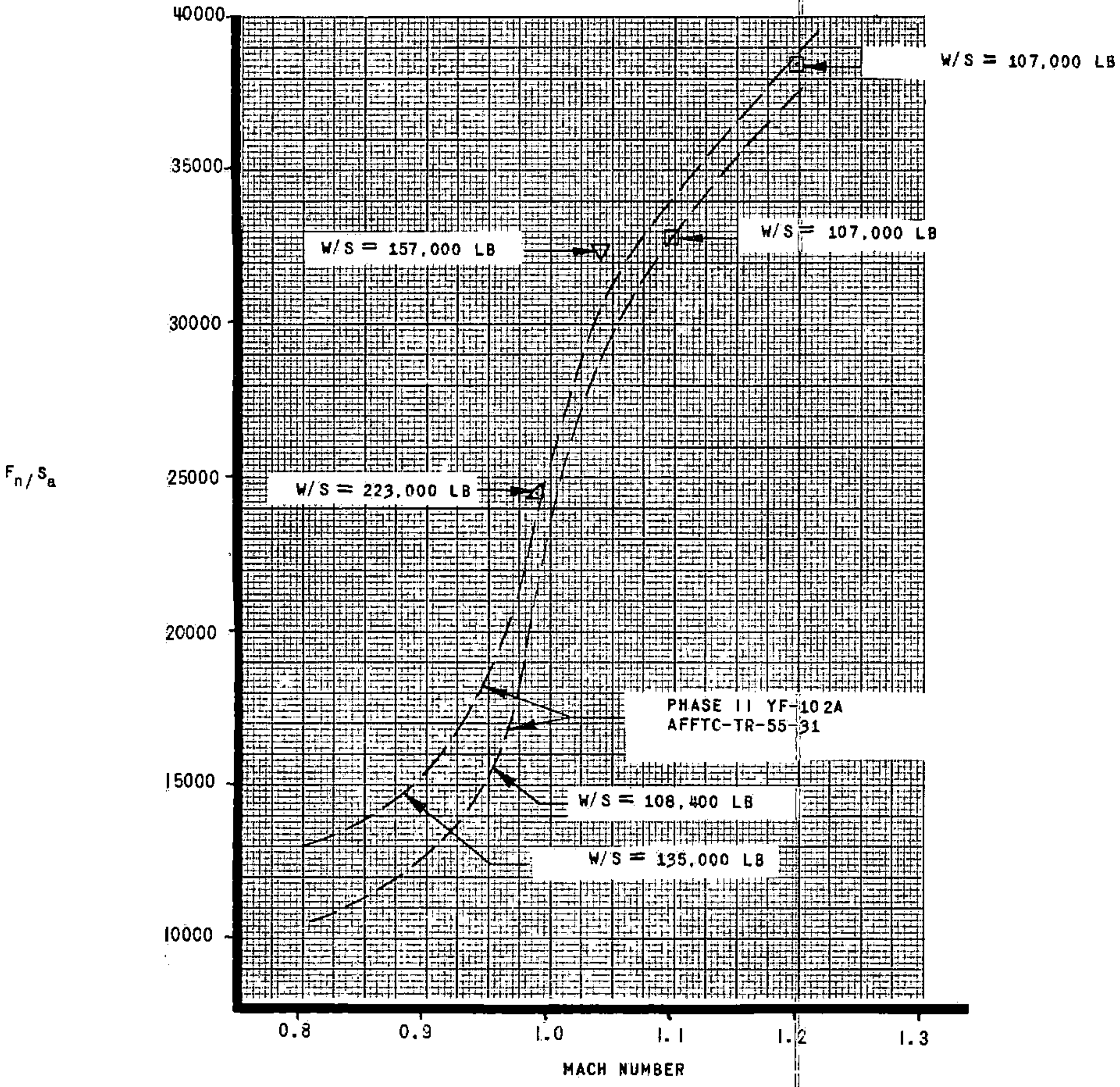


FIGURE 21 DRAG RISE

SYM	ALTITUDE	W/S
⊡	37,000 FT.	107,000 LB
▽	44,000 FT.	157,000 LB
◿	50,500 FT.	223,000 LB

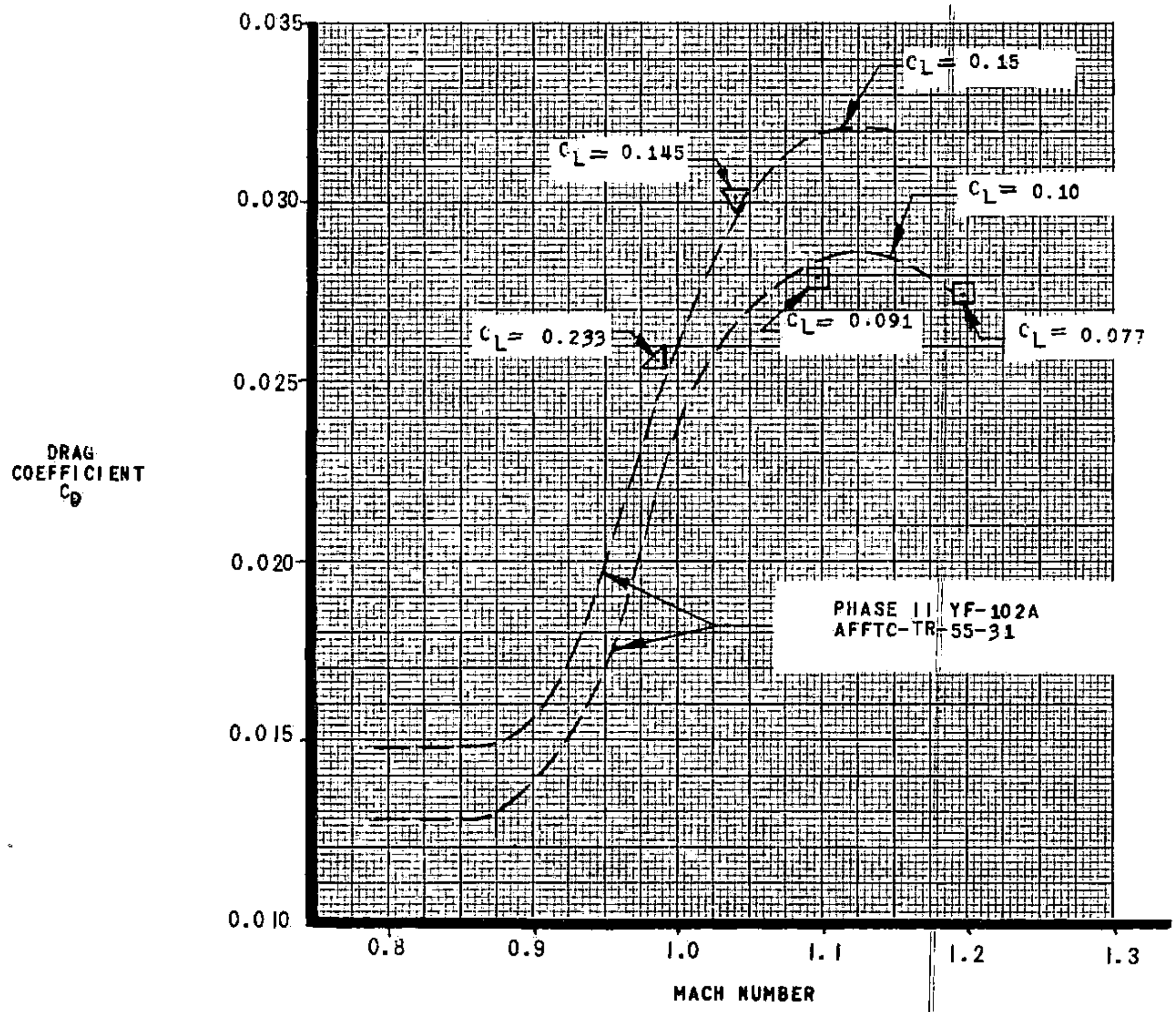


FIGURE 22 STATIC THRUST CALIBRATION

SYM DATE
○ 17 JAN.
▣ 17 JAN.

J57-P-41 S/N 604217
AFTERBURNER S/N 600563
IRIS TYPE NOZZLE
DOWNSTREAM PROBES

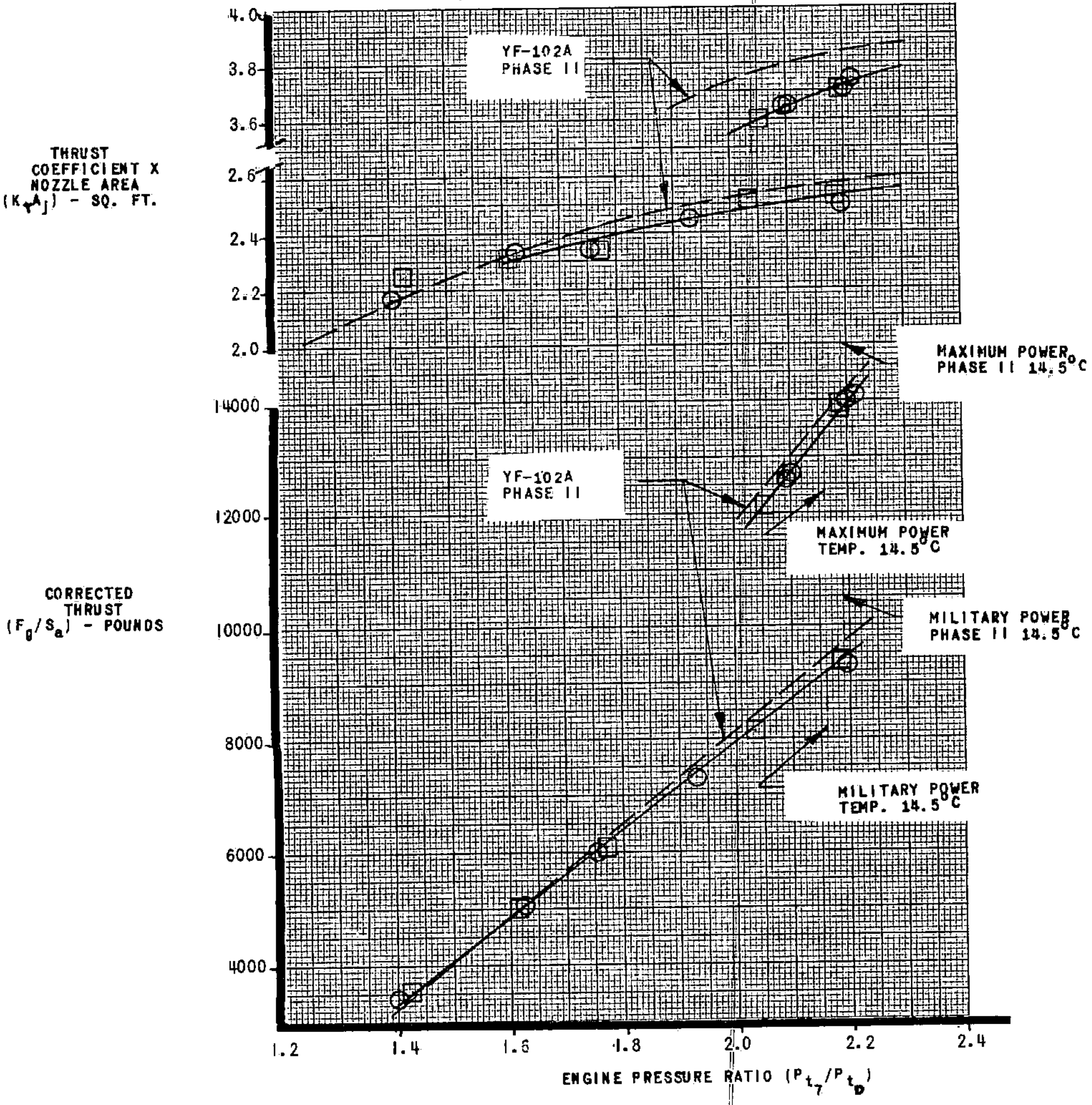


TABLE I

ROLLS FROM 60 DEGREE BANK TO 60 DEGREE BANK

12,000 feet, 180 knots IAS. (Power approach configuration, gear down.)

Total Aileron (deg)	Roll Rate (deg/sec)	Maximum Sideslip Angle (deg)	Maximum Normal Acc. (g)	Yaw Damper and Turn Coordinator
3.8 L	40 L	1.5 L	1.4	On
3.3 R	20 R	1.5 R	1.0	On
7.5 L	50 L	2.5 L	1.2	On
7.0 R	60 R	3.5 R	1.2	On
7.5 L	60 L	2.0 L	1.4	On
8.0 R	70 R	3.0 R	1.3	On
9.0 L	60 L	3.0 L	1.5	On
12.3 R	80 R	4.0 R	1.4	On
11.2 L	70 L	4.0 L	1.6	On
14.3 R	85 R	4.0 R	1.4	On
14.0 L	70 L	5.5 L	1.8	On
14.3 R	90 R	4.0 R	1.4	On

TABLE II

ROLLS FROM WINGS LEVEL THROUGH 360 OR MORE DEGREES

Cruise Configuration, 15,000 Feet

	Total Aileron (deg)	Roll Rate (deg/sec)	Maximum Sideslip Angle (deg)	Maximum Normal Acc. (g)	Number of Cont. Rolls	Yaw Damper and Turn Coordinator
Trim: 0.53 Mach number 260 knots IAS 15,000 feet	5 L	55 L	2 R	1.5	1	On*
	12 L	140 L	5.3 L	2.3	1	On
	13.5 L	140 L	5.75 L	2.5	2	On
	12.8 L	140 L	6.25 L	2.3	2	On
	12.8 L	130 L	6.25 L	3.0	2	On
Trim: 0.62 Mach number 310 knots IAS 15,000 feet	3 L	40 L	1.75 R	1.6	1	On
	4.8 L	65 L	1.5 L	1.9	1	On
	13 L	160 L	4.25 L	2.3	1	On
	13 L	160 L	6.5 L	3.3	2	On
	12.5 L	160 L	6.0 L	2.8	2	On
	10.3 R	140 R	3.0 R	2.1	2	On
	13.1 L	175 L	5.0 L	2.2	2	On
Trim: 0.81 Mach number 410 knots IAS 15,000 feet	4.8 R	100 R	2.0 L	1.6	1	On
	2.8 L	60 L	1.5 R	1.3	1	On
	4.2 L	90 L	1.5 R	1.6	1	On
	8 L	140 L	1.25 L	2.0	1	On
	11.2 L	190 L	1.0 L	2.2	1	On
	11.2 L	215 L	2.0 L	2.6	1	On
	9.5 L	175 L	1.75 L	2.3	2	On
	10 L	180 L	2.0 L	2.4	2	On
	10.1 L	180 L	5.0 L	2.8	2	On
	11.5 L	190 L	2.25 L	2.5	2	On
Trim: 0.95 Mach number 480 knots IAS 15,000 feet	8.5 L	140 L	1.5 L	1.9	60°–60°**	On
	8.75R	160 R	1.5 R	1.65	60°–60°**	On
	9.0 L	150 L	2.5 L	2.0	60°–60°**	On
	8.25R	135 R	1.5 R	2.4	60°–60°**	On
	7.25L	120 L	1.5 L	2.0	60°–60°**	On
	8.75R	130 R	2.25 R	2.0	60°–60°**	On
	7.5 L	135 L	2.0 L	1.8	1	On
	8.75R	140 R	2.0 R	1.3	1	On
	10.5 L	170 L	4.75 L	2.9	1	Off*

** These rolls were made with only 60 degree to 60 degree change in bank angle.

* The yaw damper and turn coordinator were both tied in to one switch in the cockpit and both were either on or off.

TABLE III

ROLLS FROM WINGS LEVEL THROUGH 360 OR MORE DEGREES

Cruise Configuration, 35,000 Feet

	Total Aileron (deg)	Roll Rate (deg/sec)	Maximum Sideslip Angle (deg)	Maximum Normal Acc. (g)	Number of Cont. Rolls	Yaw Damper and Turn Coordinator
Trim: 0.77 Mach number 255 knots IAS 35,000 feet	7.6 L	130 L	1.0 L	1.9	1	On
	7.5 R	120 R	.7 R	2.2	1	On
	10 L	175 L	1.5 L	2.1	1	On
	9.9 R	150 R	1.4 L	2.5	1	On
	13 L	185 L	2.5 L	2.4	1	On
	10 R	175 R	1.5 R	2.1	1	On
	11.5 L	180 L	5.0 L	2.2	1	On
	10 R	180 R	3.5 R	2.0	1	On
	9.2 L	125 L	5.7 L	3.1	2	On
	8.7 L	135 L	5.8 L	3.2	2	On
	9.5 L	150 L	3.7 L	2.5	3	On
	10.7 L	170 L	4.0 L	3.0	4	On
	10 L	150 L	3.0 L	3.1	3	On
	10.2 L	130 L	3.6 L	3.2	3	On
	11.2 L	150 L	3.0 L	2.2	1	On
	11.3 L	140 L	2.2 L	2.2	1	Off
	10 L	135 L	3.0 L	2.1	1	On
	10.5 L	135 L	3.5 L	2.2	1	Off
Trim: 0.92 Mach number 314 knots IAS 35,000 feet	7.5 L	150 L	.6 L	1.7	1	On
	8.5 L	150 L	1.5 L	2.0	1	On
	7.0 R	130 R	1.7 L	1.6	1	On
	9.2 L	180 L	1.5 L	1.8	1	On
	8.8 R	160 R	2.0 L	1.8	1	On
	10.7 L	180 L	2.8 L	2.0	1	On
	13.7 R	215 R	2.0 L	2.3	1	On
	12.2 L	210 L	3.0 L	2.0	1	On
	13 R	210 R	2.0 L	2.1	1	On
	12.6 L	215 L	3.4 L	2.3	1	On
	10.2 L	175 L	2.5 L	2.6	1	On
	9 L	160 L	3.5 L	3.3	1	On
	9 L	150 L	3.5 L	4.5	1	On
	9 R	175 R	.7 R	2.8	1	On
	9.2 L	185 L	1.2 L	3.0	3	On
	11 L	175 L	2.0 L	3.2	3	On
	8.8 L	160 L	2.0 R	2.0	1	On
	7 L	170 L	3.0 R	2.7	1	Off
Trim: 1.12 Mach number 400 knots IAS 35,000 feet	5.3 L	80 L	.8 R	1.1	1	On
	4.7 R	75 R	1.5 L	1.5	1	On
	7.5 L	110 L	1.5 R	1.5	1	On
	7.5 R	110 R	1.5 L	1.2	1	On
	11.5 L	175 L	2.0 R	2.2	1	Off
	10.7 R	160 R	2.0 L	1.2	1	On
	12.2 L	185 L	2.3 R	2.4	1	On
	10.7 L	160 L	1.8 R	2.2	2	On
	12.2 L	200 L	2.5 R	2.4	2	Off
	10.5 L	180 L	2.3 R	2.3	2	Off
	12.0 L	200 L	2.5 R	2.8	2	On
	12.7 L	200 L	2.0 R	2.9	1	On

appendix II flight log

FLIGHT	DATE	TIME	PILOT
1	11 Jan 56	1:20	Stephens, R. L., Major, USAF
2	13 Jan 56	:55	Childs, S. R., Major, USAF
3	13 Jan 56	:50	Childs, S. R., Major, USAF
4	18 Jan 56	:35	Childs, S. R., Major, USAF
5	18 Jan 56	:35	Childs, S. R., Major, USAF
6	18 Jan 56	:40	Childs, S. R., Major, USAF
7	19 Jan 56	:55	Hanes, H. A., Col., USAF
8	20 Jan 56	:35	Baker, R. M., Col., USAF
9	20 Jan 56	:55	Holtoner, J. S., Brig. Gen., USAF
10	21 Jan 56	:35	Everest, F. K., Lt. Col., USAF
11	21 Jan 56	:45	White, R. M., Capt., USAF
12	21 Jan 56	:55	Stephens, R. L., Major, USAF
13	22 Feb 56	1:05	Stephens, R. L., Major, USAF
14	23 Feb 56	:35	Childs, S. R., Major, USAF
15	24 Feb 56	:50	Stephens, R. L., Major, USAF
	Total	**12:05**	

TEST DATA CORRECTED FOR INSTRUMENT ERROR

F-102A USAF NO. 53-1813

EVALUATION OF ENLARGED VERTICAL STABILIZER

TEST: Check Climb (Max. Power) Weight at Engine Start 27,640 lb.

Alt. Ft.	IAS Knots	Time Min:Sec	P_{t_7}/P_{t_0}	t_a °C	Gross Wt. Lb.	N_I % RPM
40,000	----	----	2.60	----	25,500	----
41,740	257	27:15	----	-11	----	6260
43,710	249	27:36	----	-12	----	----
44,780	247.5	27:48.2	----	-12	----	6160
45,000	----	----	2.62	----	25,300	----
45,830	240.5	28:02	----	-12	----	6160
46,810	243	28:20	----	-13	----	6150
47,790	224	28:43.8	----	-14	----	6150
48,810	210	28:57.5	----	-15	----	6150
49,780	210	29:17.3	----	-16	----	6105
50,000	----	----	2.67	----	25,000	----
50,320	210	29:35	----	-18	----	6105
50,800	209	29:51	----	-20	----	----
51,330	208.5	30:13	----	-20	----	6120
51,800	210	30:48	----	-21	----	----
52,000	----	----	2.72	----	24,700	----
52,325	210	31:29.7	----	-23	----	6130
52,585	205.5	31:59.2	----	-23	----	----
52,835	204.5	32:21	----	-24	----	6130
53,000	----	----	2.74	----	24,500	----
53,085	201	33:14.5	----	-25	----	----
39,690	284.5	57:01	----	-11	----	
40,000	----	----	2.57	----	25,600	
41,680	269	57:29	----	-11	----	
43,690	256.5	58:02	----	-10	----	
44,760	251	58:26	----	-10	----	
45,000	----	----	2.58	----	25,400	
45,730	243.5	58:42	----	-10	----	
46,780	234	59:02	----	-12	----	
47,810	229	59:21.5	----	-13	----	
48,820	221.5	59:39.5	----	-14	----	
49,830	211	59:55	----	-15	----	
50,000	----	----	2.61	----	25,100	
50,240	210	00:05	----	-15	----	
50,690	211	00:25	----	-16	----	
51,000	----	----	2.62	----	25,050	
51,330	206	00:50.5	----	-16	----	
51,800	203	01:14.5	----	-18	----	
52,310	200.5	02:45	----	-20	----	
52,525	199	03:23	----	-22	----	
52,800	----	----	2.63	----	24,700	

TEST DATA CORRECTED FOR INSTRUMENT ERROR

F-102A USAF NO. 53-1813

EVALUATION OF ENLARGED VERTICAL STABILIZER

TEST: Level Flight Data — Weight at Engine Start 27,640 lb.

Alt. Ft.	IAS Knots	t_a °C	P_{t_7}/P_{t_0}	Gross Wt. Lb	N_1 RPM	A/B
50,630	232.5	-23	2.67	23,800	6170	Off
37,210	374.5	+1	2.41	22,500	5970	On(Min.)
43,910	300.5	-2	2.47	24,050	6120	On
37,020	412.0	+12	2.47	23,000	6180	On

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A USAF NO. 53-1813
EVALUATION OF ENLARGED VERTICAL STABILIZER

TEST: Static Thrust Calibration Weight at Engine Start 27,640 lb.

Run No.	Fg Lb	P_{t_7}/P_a	N_2 % RPM	EGT °C	A/B	
First Run	P_a = 27.65	t_a = 9°C				
1	5540	1.755	89.5	415	Off	
2	13,030	2.215	97.15	590	On	
3	4695	1.625	88.85	400	Off	
4	11,755	2.103	95.35	540	On	
5	3155	1.405	85.15	355	Off	
6	6790	1.93	93.0	490	Off	
7	11,660	2.095	95.15	540	On	
8	435	----	55.90	220	Off	
9	8620	2.195	97.05	585	Off	
10	12,897	2.20	97.15	585	On	
Second Run	P_a = 27.59	t_a = 12°C				
1	5635	1.77	90.2	425	Off	
2	8670	2.185	97.9	600	Off	
3	12,735	2.192	98.0	600	On	
4	4640	1.62	89.7	405	Off	
5	7525	2.035	95.7	540	Off	
6	11,160	2.055	95.8	590	On	
7	3260	1.425	86.2	380	Off	
8	8660	2.185	98.0	600	Off	
9	12,795	2.198	98.05	600	On	

1733

1733

GDC-AYC69-008

Final Report

VOLUME I

FLIGHT TEST RESULTS F-102A/J-57 COMPRESSOR STALL FIX DEVELOPMENT AND EVALUATION

April 1970

Prepared for
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Request No. VE06
San Antonio Air Materiel Area
Kelly Air Force Base, Texas

GENERAL DYNAMICS
Convair Division

Report: GDC-AYC-69-008
Date: December 1969

VOLUME I

FLIGHT TEST RESULTS
F-102A/J-57 COMPRESSOR STALL
FIX DEVELOPMENT AND EVALUATION

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for C. S. Brandt Program Manager

Prepared for: Service Engineering Contract AF 41608-69-C-2184, Request No. VE-06
issued by San Antonio Air Materiel Area, Kelly Air Force Base, Texas

GENERAL DYNAMICS
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San Diego, California

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ABBREVIATIONS

A	Area
A/B	Afterburner
A/C	Aircraft
Accel.	Acceleration
AFAPL	Air Force Aero Propulsion Laboratory
AFB	Air Force Base
AFFTC	Air Force Flight Test Center (Edwards AFB)

C	Temperature - Degrees Centigrade
CAS	Calibrated Air Speed

DCAS	Defense Contracts Administration Services
Decel	Deceleration
Deg.	Degrees

EAFB	Edwards Air Force Base
ECN	Engineering Change Notice
e.g.	For example
EGT	Exhaust Gas Temperature
ENC	Exhaust Nozzle Control
etc.	Et Cetera

F	Temperature - Degrees Fahrenheit
FAA	Federal Aviation Administration
F/C	Fuel Control
FCF	Functional Check Flight
Flt	Flight
F.S.	Fuselage Station
ft	Feet
FTEO	Flight Test Engineering Order
Fwd	Forward

g/m^3	Grams per cubic meter
g	Gravity constant 32.2 feet per second per second
GDC	General Dynamics Convair Division
GFE	Government Furnished Equipment
GFP	Government Furnished Property

ABBREVIATIONS (Continued)

H_i $(H_{ic})(H_t)$	Pressure Altitude - indicated (Instrument Corrected) (True)
i.e.	That is
IRIG B	Standard Time Code Format - 100 pulse per second mode
KCAS	Calibrated airspeed (Knots)
KIAS	Indicated Airspeed (Knots)
Kt	Knots
Lb	Pounds
M (Mi)	Mach True (Indicated)
Max	Maximum
Mil	Military
Min	Minimum/Minutes
N_1	Low Pressure Compressor Rotational Speed
N_2	High Pressure Compressor Rotational Speed
No.	Number
N_Z	Normal Load Factor
OCAMA	Oklahoma City Air Materiel Area
Pct	Percent
P.L.A.	Power Lever Angle
P_s	Static Pressure
P_{s2}	Compressor Inlet Static Pressure
P_{s3}	Low Pressure Compressor Discharge Static Pressure
P_{s4}	High Pressure Compressor Discharge Static Pressure
$P_{s4.5}$	Burner Can Static Pressure
PSI (PSIA) (PSID)	Pounds per square inch (absolute) (differential)
P_t	Total pressure
P_{t2}	Total pressure at Low Pressure Compressor Inlet
P_{t3}	Total pressure at Low Pressure Compressor Discharge
P_{t4}	Total pressure at High Pressure Compressor Discharge
P_{t7}	Total pressure at Low Pressure Turbine Outlet
PWA (P&W)	Pratt and Whitney Aircraft

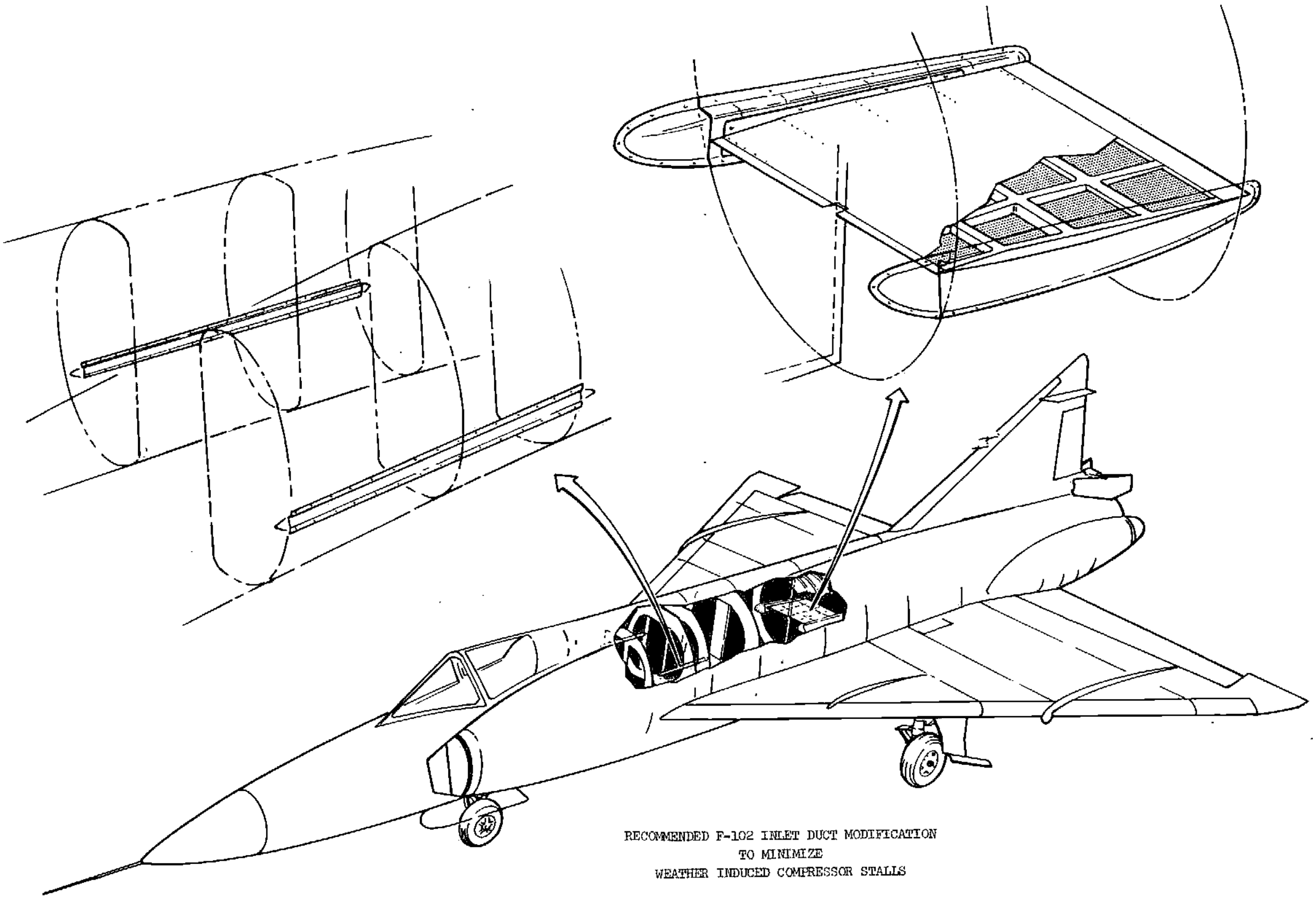
ABBREVIATIONS (Continued)

RAT	Ram Air Temperature - degrees C
Ref	Reference
RNI	Reynolds Number Index
SAAMA	San Antonio Air Materiel Area
SANE	San Antonio Engineering
Sec	Seconds
S.L.	Sea Level
S/N	Serial Number
TCG	Time Code Generator
TCTO	Time Compliance Technical Order
T.O.	Technical Order (Take-off)
T_{t2}	Temperature at Low Pressure Compressor Inlet
T_{t3}	Temperature at Low Pressure Compressor Discharge
T_{t4}	Temperature at High Pressure Compressor Discharge
T_{t5}	Temperature at Low Pressure Turbine Outlet
USAF	United States Air Force
V_i (V_{ic}) (V_t)	Velocity Indicated (instrument Corrected) (True) - (Knots)
WPAFB	Wright-Patterson Air Force Base
W_f(W_{fAB})	Engine Fuel Flow (Afterburner Fuel Flow) - (Pounds/hour)

$$ZC = 1-\left[(P_{t4}/P_{t3})_{stall}-(P_{t4}/P_{t3})_{operating}\right]\Big/\left[(P_{t4}/P_{t3})_{stall}-(P_{t4}/P_{t3})_{low\ map\ value}\right]$$

SYMBOLS

α	Angle of Attack
β	Angle of Sideslip
Δ	Incremental (Prefix)
%	Per Cent
θ	Temperature Ratio
δ	Pressure Ratio



RECOMMENDED F-102 INLET DUCT MODIFICATION
TO MINIMIZE
WEATHER INDUCED COMPRESSOR STALLS

FOREWORD

This report presents results of the F-102A Compressor Stall Fix Development flight test program. Test plans for this program, Convair Report GD/C-AYC-69-003, were prepared in accordance with findings of the compressor stall investigation program reported in Convair Report GD/C-BHF-69-001, July 1969. Seven engine inlet duct configurations were evaluated using aircraft F-102A Serial Number 56-1151. Only one engine modification was evaluated on a by-product basis, because engine responsibility was transferred to OCAMA after program start and authorization from OCAMA to continue the engine fix evaluation is still pending. Due to the large quantity of information obtained during the compressor stall fix development program, this report was prepared in two volumes; Volume I containing the analysis and summary information, with the bulk of substantiating data presented in Volume II.

Under authority of SAAMA Contract Number AF-41608-69-C-2184, Work Statement Request VE-06, flight tests were conducted from 21 July through 20 October 1969.

General Dynamics/Convair, San Diego, California, was assigned prime responsibility in areas of aircraft maintenance, theoretical analysis, test program, coordination, instrumentation, data collection, data analysis and reporting. The Aero Propulsion Lab at Wright-Patterson AFB provided an assist by generating specific engine stall data for sea level conditions. During the test program, maintenance and support were also provided by the following bases:

Tyndall AFB, Florida
Nellis AFB, Nevada
Luke AFB, Arizona
Kirtland AFB, New Mexico
Webb AFB, Texas
Ellington AFB, Texas
MacDill AFB, Florida
Edwards AFB, California
Barksdale AFB, Louisiana
Ontario AFB, California.

This work was administered under direction of J. L. Tubbs, Project Engineer for Service Engineering Division, Directorate of Materiel Management, San Antonio Air Materiel Area, Kelly Air Force Base, Texas.

ABSTRACT

This report presents results of the flight test program to evaluate several engine inlet duct modifications and to evaluate on a by-product basis an engine modification for the purpose of minimizing the occurrence of F-102/J-57 compressor stalls.

The engine modification consisted of increasing the first stage turbine nozzle area (A5) by 3%. The modification increased the compressor stall margin substantially at the higher Reynolds Number Index, and this increase improves the engine tolerance to ice crystal ingestion. It has little or no effect at the low Reynolds Number Index where additional margin is needed.

A total of seven engine inlet duct configurations, in addition to the clean baseline duct, were evaluated. The most effective modification to reduce weather induced compressor stalls consisted of an aft horizontal vane installed at 6 degrees nose-up relative to water line, plus an outboard strake in both left and right ducts, placed ahead of the horizontal vane. The proposed engine stall modifications still require evaluation, however, it is recommended that: (1) engine A5 area be increased as tested, or perhaps as later improved, and (2) the above specified duct modification be incorporated in all F-102 aircraft to minimize the occurrence of weather induced compressor stalls.

1.0 DISCUSSION

The F-102A/J57-P23A compressor stall investigation program, Reference A, was initiated following increasing frequency of F-102A compressor stall reports. In certain cases, these stalls contributed to major accidents. Of the three fighter aircraft with J-57 engines, the F-102 experienced more compressor stalls. Revision of operating procedures, i.e., avoidance of severe storms reduced incidents, but the problem was still of such magnitude that it was investigated by the Air Force. From this investigation, it was deemed necessary to instrument an F-102/J-57 and conduct tests to determine causes of engine stalls and to recommend corrective action. This resulted in authorization of a compressor stall ground and flight test program under authority of SAAMA Contract Number AF-41608-68-C-0132, Work Statement Request 0012, Amendment AC titled "F-102 Compressor Stalls". Work on this program was started 7 April 1968. This program was completed on 9 February 1969 and was reported in Reference A. Results of this investigative ground and flight test program described above resulted in initiation of the test program covered by this report.

The purpose of this report is to present results, conclusions and recommendations of the test program conducted to evaluate several engine inlet duct modifications and one engine modification with the purpose of minimizing F-102 compressor stalls.

During the course of the program, seven engine inlet duct modifications, in addition to the clean inlet duct, were tested. Convair engineering drawings FT 11-68024, FT 11-68025, sheets 1 and 2, and FT 11-68027 were the basic drawings used to modify the aircraft for the seven duct modifications. Figures 7, 8, 9, 12, and 14 illustrate the components used in the duct modifications.

Test flights were conducted 21 July through 20 October 1969, on F-102A airplane S/N 56-1151. Other highlights are shown in Program Milestones, Table I-1. The flight test program consisted of 35 flights for a total of 49.8 flight hours. Weather penetration flights were conducted on 16 flights for a total of 10 separate compressor stalls. Flight maneuvers included snap-ups, roller coasters, and others to induce compressor stalls or to evaluate the effect of duct modification on inlet pressure distortion. Compressor stall was experienced during snap-up maneuvers on three flights. Flight test operations were conducted from 9 Air Force Bases, in addition to Lindbergh Field, San Diego.

1.0 DISCUSSION (Continued)

1.1 MODIFICATIONS TO THE PROGRAM PLAN

The program was modified to adapt to the accumulated test results. The first such change was elimination of the vertical splitter extension (Configuration 3 in the program plan). This was originally intended for a back up configuration should problems develop with Configurations 1 and 2, aft and forward horizontal vanes respectively. Since both configurations 1 and 2 performed to expectations, the need for back-up no longer existed and with the concurrence of SAAMA the back-up configuration was dropped from the test schedule. The second modification to the program was due to test results of Flight 61, which was ostensibly a clean duct flight test. Test results showed that the support strakes for configurations 1 and 2 exerted considerable influence on the flow field with the appearance of forward vane effects. Testing a combined configuration (aft vane plus forward support strakes) followed by testing with the forward support strakes alone was initiated to help isolate the forward support strakes effect. This program change led to a further program change since it was confirmed that the forward vane area near the outboard wall was quite sensitive. Two different strake configurations were tested to determine the influence of counter rotational devices (as discussed in the program plan) in this sensitive duct area.

1.2 TECHNICAL DISCUSSION OF TEST RESULTS

A procedure was specified in the program plan to generate suitable means for comparing test data of the several configurations. This procedure was followed as a general guide with such deviations as required to accommodate input changes.

The Aero Propulsion Lab (AFAPL) at WPAFB provided an assist by generating the sea level N_2 stall line as a function of temperature at Station 3. This is shown in Figure 1. These data were developed by adding an intercooler between the low pressure rotor and the high pressure rotor, and should be similar to the process in storms where ice is melted and vaporized (thus cooling the surrounding air) prior to entering the high pressure rotor. The ordinate in Figure 1 is a pressure function representing N_2 stall margin. It is equal to the difference between pressure ratio at stall, (P_{t4}/P_{t3}) stall, and the operating pressure ratio, $(P_{t4}/P_{t3})_{op}$, divided by the difference between pressure ratio at stall and the bottom line pressure ratio of the compressor map, $(P_{t4}/P_{t3})_{low}$.

1.0 DISCUSSION (Continued)

1.2 TECHNICAL DISCUSSION OF TEST RESULTS (Continued)

The temperature depression at Station 3 can be related to ice crystal concentration by a simple heat balance and the curve made to represent the storm penetration condition where ice is ingested with the air. This is limited to the sea level condition due to limitations of the AFAPL program. Altitude stall margin is outside the scope of data provided to AFAPL by P&WA. Also, it is noted that test data for engine "C" with increased A5 modification could not be made available in time to support the AFAPL program. As in the program plan, data from Figure 1 were converted to the more convenient form of rpm ratio to Reynolds Number Index (RNI). This was done for convenience since compressor pressure ratio is not measured in flight. The rpm ratio chosen was $N_2/\sqrt{\theta_{t3}}/(N_2/\sqrt{\theta_{t3}})_{op.}$ which is the corrected N_2 velocity in the storm conditions divided by the corrected N_2 velocity in dry air for the same condition (operating line versus RNI).

RNI, which is equal to $\delta_{t2}/\theta_{t2}^{1.24}$ is chosen for the abscissa to permit comparison at different flight conditions. The sea level data then represents an RNI of 1.0. It was found that ambient pressure effects on compressor stall margin is a second order effect while ambient temperature is a first order effect. The first cut indicated by this is to vary the stall line at constant physical N_2 velocity according to temperature variation only. This was refined by comparison to test data to give the bottom line of Figure 2. In order to generate data for the increased A5 modification it was assumed that stall pressure ratio versus corrected rpm would remain the same as for the unmodified engine. From this and the operating line for the modified engine the top line of Figure 2 was developed in the same manner as for the unmodified engine.

Effects of distortion were developed empirically using as index of distortion $\left[(\Delta T_{t3})\ \text{max} - (\Delta T_{t3})\ \text{min}\right]/\Delta T_{t3}\ \text{ave.}$ T_{t3} is measured by four rakes at approximately 69, 153, 234, and 298 degrees (looking forward) on the circumference. Each rake holds five thermocouple probes located for equal area sampling. The average of the five probes should be a valid representation of the average temperature at that location. The limitation of four rakes adversely affects a true average Station 3 temperature in the presence of high thermal distortion. Some scatter is therefore expected at high distortion. Reduction of thermal distortion levels, however, will permit a more accurate indication of the average Station 3 temperature.

1.2 TECHNICAL DISCUSSION OF TEST RESULTS (Continued)

ΔT_{t3} in the above formula is the difference between a given rake average in clean air and the same rake's average temperature in an ice crystal laden environment. Since there is virtually no circumferential mixing in the compressor, this temperature depression reflects the change in enthalpy due to two phase (ice plus air) flow at the corresponding circumferential location on the compressor face. This change in enthalpy of the air divided by the change in enthalpy required to convert ice at ambient temperature to vapor at station three air temperature gives a mass percentage of ice to air at the compressor face. Low pressure rotor efficiency changes due to water cooling are assumed to be small and are neglected. Also, the use of total temperature at station three introduces a small error due to the difference between air and water vapor values for velocity of sound and ratio of specific heat. This error at 200 feet per second amounts to 1.5 degree F which is within instrumentation accuracy. Results of this technique show excellent agreement with the National Severe Storms Project, Reference C.

The culmination of these efforts is shown by Figure 3 for the unmodified engine "C" and by Figure 4 for the increased A5 modification to engine "C". Figure 4 will be used in the comparison which follows as the primary evaluation tool. In a secondary role, will be the temperature pattern at Station 3. The temperature excursions will, of course, be reflected in the distortion index but the index does not show which section of the compressor face ingests the high ice concentration. Total ice concentration, as determined by compressor heat balance will also be reported but it is cautioned that these are not directly comparable except through the stall curve.

In discussing ice concentration, the preferred units for heat balance calculations are weight units, which in the present case are chosen to be grains of water per pound of air. Meteorological practice, however, employs weight per unit volume, i.e., grams of water per cubic meter (g/m^3) which is not directly comparable due to the variation of air density with altitude. For the following discussion, both units will be reported. Note that water is used here in the generic sense since in the thunderstorm the water occurs in solid phase (ice crystals) and is converted to the vapor phase prior to station 3.

The following discussion of the configurations tested are arranged by configural characteristics rather than chronologically.

1.0 DISCUSSION (Continued)

1.2.1 Clean Duct Configuration - Baseline

Smoothly patched holes from prior duct modifications were flown in cloud penetrations on Flight 80. This configuration showed a high level of thermal distortion with the characteristic low points in water content at rakes 1 and 3. This pattern is shown on Figure 5. Similarity can be noted between the temperature pattern of Flight 80 and that of Flight 41 behind the water spraying tanker during the Compressor Stall Investigation Program. Ice concentrations as high as 190.5 grains per pound of air were penetrated at relatively low altitude (27,000 to 30,000 feet). The increased compressor stall margin provided by this altitude range permitted the high ice concentrations. A compressor stall was experienced at an ice concentration of 183.3 grains per pound of air (12.85 g/m^3). The distortion index at stall was .5 and the $N_2 / \sqrt{\theta_{t_3}}$ ratio was 1.0928 at .485 RMI.

1.2.2 Roughly Patched Holes (Brazier head screws and raised skin patches) Flown in Cloud Penetrations on Flight 68

This configuration approached the thermal distortion of Flight 80 although the rough patching in the sensitive forward vane area apparently mitigated the low ice concentration at rake #3. Temperature depression patterns are shown on Figure 6. No compressor stalls occurred although ice concentrations as high as 127 grains per pound of air (8.88 g/m^3) were encountered. As in the case of Flight 80, the storm was relatively low in altitude and productive penetrations were limited to 25,600 to 28,230 foot altitude. The distortion index ranged from .799 to 1.095 at the higher ice concentrations and the highest $N_2 / \sqrt{\theta_{t_3}}$ attained was 1.9854 at .524 RMI.

1.2.3 Summary of Clean Configurations

Results of these tests show the improvement in compressor stall margin attributable to the A5 increase in engine "C". The characteristic pattern of thermal distortion due to the inlet is present, but the engine as modified is capable of accommodating substantial quantities of ice and distortion at the test altitudes due to increased stall margin.

1.3 CONFIGURATION #1: AFT HORIZONTAL VANE, FIGURES 7, 8, AND 9

1.3.1 Vane set at 2 degrees relative to aircraft water line (0 degrees angle of incidence to flow) flown in cloud penetration on Flights 51 and 52.

1.3 CONFIGURATION #1: AFT HORIZONTAL VANE, FIGURES 7, 8, AND 9 (Continued)

1.3.1 (Continued)

Temperature depression patterns, shown in Figure 10, are similar to Flight 68 patterns. A compressor stall was experienced during Flight 52 at an ice concentration of 130 grains per pound of air (5.78 g/m^3). The distortion index was .68 and the $N_2/\sqrt{\theta_{t_3}}$ ratio 1.0738 at .337 RNI.

1.3.2 Vane set at 4 degrees relative to water line (2 degrees angle of incidence to the flow) flown in cloud penetrations on Flight 53.

Effect of the increased angle of attack is apparent in the increased ice concentration at rakes 1 and 3 as shown in Figure 11. It is noted that throughout this program the configurations tested were less effective at low ice concentration and for that reason the lower line of this figure is shown without comment. The vane is exerting positive control as evidenced by the reduced distortion and by consistency. Compressor stall was not encountered at ice concentrations as high as 152 grains per pound of air (6.42 g/m^3). The distortion index at higher ice concentrations ranged from .145 to .28 and the maximum $N_2/\sqrt{\theta_{t_3}}$ ratio was 1.085 at .283 RNI.

1.3.3 Vane set at 6 degrees relative to water line (4 degrees angle of incidence to flow) flown in cloud penetrations on Flight 54.

This angle appears to be optimum for this configuration as both flow control and temperature pattern are superior to Flight 53 at little change in pressure patterns. Figure 12 shows the temperature patterns experienced. It is noted that rakes #1, 3, and 4 are held to reasonable levels (relatively consistent ice concentration) due to vane action while rake #2 still reflects a higher ice concentration. It is felt that this is due to the proximity of the vane to the compressor face with insufficient mixing distance downstream of the vane. Compressor stall was not experienced with this configuration. Maximum ice concentration was 153.5 grains per pound of air (6.62 g/m^3). The distortion index at higher ice concentrations ranged from .149 to .270 and the maximum $N_2/\sqrt{\theta_{t_3}}$ was 1.094 at .339 RNI.

1.0 DISCUSSION (Continued)

1.3 CONFIGURATION #1: AFT HORIZONTAL VANE, FIGURES 7, 8, AND 9 (Continued)

1.3.4 Summary of Configuration #1

At positive angles of incidence this vane induced a substantial amount of the ice present to the lower portion of the compressor face thus decreasing the distortion. The flow phenomena which cause the enthalpic distortion originates well forward in the duct system, however. The conditions which cause high ice concentrations are therefore well established upstream of the #1 vane and there is insufficient mixing length downstream of the vane to homogenize the mixture. Radial pressure distortion at the 90 degrees and 270 degrees location was increased by the wash from this vane, especially at the higher vane angles, but is within limits. Overall pressure distortion was not affected and at the 6 degrees position pressure distortion was reduced in high angle flight. Any change in pressure recovery was within the instrumentation accuracy and not detected.

1.4 CONFIGURATION #2: FORWARD HORIZONTAL VANES, FIGURES 13 AND 14

1.4.1 Vanes set at 0 degrees angle of incidence, flown in cloud penetrations on Flight 56.

As shown in Figure 15, the temperature patterns produced by this configuration are quite different than Configuration #1. The forward location is sensitive to altering the flow patterns and rotation even without the aid of angle of incidence. The control of ice concentration is evident by the consistency of pattern for the several passes. The forward location also permits any vane wash to be vitiated well ahead of the compressor face thus minimizing any pressure distortion effects. Ice concentrations as high as 169 grains per pounds of air (8.63 g/m^3) were encountered. A compressor stall occurred at an ice concentration of 151.7 grains per pound of air (8.09 g/m^3). The distortion index at the higher ice concentrations ranged from .140 to .235. The $N_2/\sqrt{\theta_{t_3}}$ ratio was 1.058 at .412 RNI for the maximum concentration and 1.0968 at .375 RNI just prior to the compressor stall.

1.4 CONFIGURATION #2: FORWARD HORIZONTAL VANES, FIGURES 13 AND 14 (Continued)

1.4.2 Vanes set at 2 degrees angle of incidence, flown in cloud penetrations on Flight 57.

Effect of increased angle on the temperature patterns is shown on Figure 16. This configuration exhibits all the advantages of Flight 56, plus more effective control over the ice concentration. Compressor stalls were experienced at ice concentrations of 159.7 and 153.8 grains per pound of air (6.28 and 6.59 g/m^3). $N_2/\sqrt{\theta_{t_3}}$ ratios at the threshold of stall were 1.0989 at .320 RNI and 1.0955 at .353 RNI, respectively. Distortion index at the higher ice concentrations ranged from .122 to .199.

1.4.3 Summary of Configuration #2

Due primarily to the sensitive location chosen for Configuration #2, its influence on flow characteristics is superior to Configuration #1. Side effects of pressure distortion that arise from proximity to the compressor face with Configuration #1 are not present in Configuration #2.

1.5 SUPPORT STRAKES AT FORWARD DUCT LOCATION

1.5.1 Support strakes only, flown in cloud penetrations on Flight 61.

This configuration was intended to be "clean" on the assumption that support strakes would make little difference. Support strakes, in fact, have an effect on the flow that resembles Configuration #2 as illustrated by the temperature patterns on Figure 17. Because of this resemblence it is assumed that the improvement is due to the forward support strakes. As noted in the program changes, results of this test redirected the program to investigate in greater depth the area of the forward vane. The shape of the forward support strakes is the source of the term "strakes" for this and subsequent configurations. A compressor stall was sustained at an ice concentration of 168.2 grains per pound of air (8.78 g/m^3). The distortion index ranged from .178 to .196 at the higher ice concentration and the $N_2/\sqrt{\theta_{t_3}}$ ratio at the threshold of stall was 1.11 at .396 RNI.

1.5 SUPPORT STRAKES AT FORWARD DUCT LOCATION (Continued)

1.5.2 Configuration #4, forward Support Strakes only, flown in cloud penetrations on Flight 66.

A single support strake is shown in Figure 14. Temperature depression patterns are shown on Figure 18. These exhibit the same characteristics as Flight 61 or Flight 56, thereby confirming the premise that these support strakes were the cause of the Flight 61 improvement. Conversely the aft support strakes exert little or no effect on the flow geometry. A compressor stall was experienced at an ice concentration of 181 grains per pound of air (9.23 g/m^3). distortion index at the higher ice concentrations ranged from .159 to .351 and the $N_2/\sqrt{\theta_{t3}}$ ratio at the threshold of stall was 1.1039 at .394 RNI.

1.5.3 Configuration #5, forward 3-inch strakes flown in cloud penetrations on Flights 69 and 72 (intervening flights were ferry flights).

These strakes utilized the same cross-sectional shape as Configuration #2 but the span was reduced to provide 3-inch penetration into the duct passage. One of these strakes is shown in Figures 19 and 20. The purpose had been to improve on Configuration #4. Such improvement did not materialize as can be seen by the temperature depression patterns on Figures 21 and 22. Ice concentrations as high as 183 grains per pound of air (13.74 g/m^3) were penetrated and a compressor stall was sustained at a concentration of 170 grains of water per pound of air (10.16 g/m^3). Distortion index ranged from .141 to .371 at the higher ice concentrations and the $N_2/\sqrt{\theta_{t3}}$ ratio at the threshold of stall was 1.11002 at .409 RNI.

1.5.4 Configuration #6, forward 1.5 inch strakes flown in cloud penetrations on Flight 75.

This was a further attempt to capitalize on the sensitive duct area discovered on Flight 61. As shown by the temperature depression profile on Figure 23, this configuration is inferior to that of Flight 61. Compressor stall was not suffered on this flight but ice concentrations up to 163 grains per pound of air (10.5 g/m^3) were penetrated. Distortion index ranged from .228 to .654 at the higher ice concentrations and the $N_2/\sqrt{\theta_{t3}}$ ratio was 1.09863 at .507 RNI.

1.5 SUPPORT STRAKES AT FORWARD DUCT LOCATION (Continued)

1.5.5 Summary of Strakes

Although the promise was found in Flight 61, it is apparent from the subsequent testing that considerable development will be required to realize that promise. As stated in the program plan, devices to oppose the flow rotation may offer a low cost solution but the development costs will quite likely be high. Alone, the strakes did not offer the same measure of control of the ice-air mixture as the configurations of Flights 57 or 64.

1.6 COMBINED CONFIGURATION

Configuration flown in cloud penetrations on Flight 63 and Flight 64.

This consisted of the configuration No. 4 forward support strakes, plus configuration No. 1 vane at 6 degrees relative to water line. The two configurations, individually good, complemented each other for a synergistic effect. The greatest tolerance to ice ingestion, lowest distortion index, and most positive control of the ice-air mixture resulted. Temperature depression patterns for the two flights are shown on Figures 24 and 25. Flight 64, due to the higher ice concentrations is of primary interest in this discussion. Compressor stalls were experienced in one cloud at 192.9 grains of water per pound of air (11.18 g/m^3) and on a subsequent pass at an ice concentration of 198.8 grains per pound of air (13.23 g/m^3). The distortion index ranged from .050 to .215 and the $N_2/\sqrt{\theta_{t_3}}$ ratios at the threshold of stall were 1.12377 at .412 RNI and 1.12376 at .529 RNI, respectively.

The compressor stalls were multipulse indicating a saturation condition, i.e., steady state combination of distortion and ice concentration that lies solidly inside the stall region. Note that most stalls in this program crossed into the stall region in a transient manner lacking the combination which would cause stalls to persist. With the low distortion level of this configuration, it seems certain that the compressor would have stalled with any duct system.

Based upon high tolerance to ice, low distortion level and excellent control of externally induced distortion, this is the recommended configuration. Due to excellent results obtained with this configuration, further testing was accomplished on Flight 65 to evaluate any side-effects in cruise, side-slip, and high speed flight. No adverse effects were observed.

1.0 DISCUSSION (Continued)

1.7 SUMMARY OF CONFIGURATIONS

Figure 26 plots test points for each configuration on the previously generated stall margin curve. The plot verifies the primary choice of configuration to be the combined configuration (Vane #1 at 6 degrees angle plus forward support strakes) and shows relative effectiveness of lesser configurations. This form of presentation, however, is recognized as esoteric and a more useful form for flight planning has been generated. Figures 27 through 30 paradigmatically represent these data. Effect of ambient temperature is small and may be neglected. The top line on these curves (ice induced engine stall-basic) represents an RNI of .225. As shown on Figure 26, the severe slope of the stall lines at lower values of RNI would have an adverse influence on accuracy. Figure 31 gives the conversion from engineering units to meteorological units of water concentration, i.e., grains of water per pound of air versus grams of water per cubic meter.

1.8 OTHER SOURCES OF THERMAL DISTORTION - NON-HOMOGENEITY OF ICE FILLED CLOUDS

Tops of the cumulo-nimbus clouds penetrated are a seething mass with definite shear surfaces causing turbulence. Variability of ice concentration due to this action was anticipated but thermal distortion was not expected due to the small inlet capture area. Test data showed thermal distortion resulting from external stimuli, however. That cloud irregularities are capable of inducing thermal distortion implies sharply defined shear surfaces in this area of the cloud. This action, then, imposes a requirement on the duct fixes to control the thermal distortion induced externally as well as the internally induced (duct) thermal distortion. To illustrate that such anomalies are controllable by the duct fixes, Figures 32, 33, and 34 show time histories of the temperature differentials for Flights 75, 56, and 64, respectively. As before ΔT_{t3} represents average temperature depression for each rake (5 probes) from the before cloud entry condition.

1.9 MANEUVERS

Yaw and roller coaster maneuvers within an ice cloud were tested for the possible effect on thermal distortion. This testing was accomplished on Flight 75. Figure 35 shows that these maneuvers aggravated the thermal distortion present from external and internal sources. This would suggest that sudden or violent maneuvers in an ice cloud should be discouraged due to reduction in stall margin by thermal distortion.

1.10 PRESSURE RECOVERY AND DISTORTION

As stated in the program plan, pressure distortion and to a limited degree pressure recovery were monitored throughout the program by means of the P_{t2} rakes. Of primary concern was the pressure distortion to give assurance that the duct fixes for ice ingestion did not introduce a pressure problem. Compressor face pressure maps were developed for typical points at cruise, high velocity, high angle of attack and snap-up. To better visualize these, ring profiles were also generated. Both appear in Volume II with their respective flight data. It is noted that the aft vane configurations show the effect of wash from the vanes in radial distortion at a point downstream of the vane. Since this increase is not excessive and since the average of the probes at this circumferential location remains the same, it is considered acceptable.

Regarding total pressure recovery, none of the fixes tested showed any deviation from the scatter band inherent in this type instrumentation. Should performance testing be required, a more accurate evaluation of pressure recovery would be required. To accomplish this, the P_{t2} transducers would be referenced to P_{to} using more precise sensors for greater accuracy.

1.11 EFFECT OF A5 INCREASE ON ENGINE STALL MARGIN

At mid program, engine responsibility was transferred from SAAMA to OCAMA, which required additional negotiations for engine tests. Approval to conduct these tests has not yet been received. In accord with the original program intent to overlap tests where possible, some engine stability testing was accomplished as by-product to weather investigation.

As noted in discussion concerning weather, the A5 modification did increase compressor stall margin substantially at the higher Reynolds Number Index and this increase improves the engine's tolerance to ice crystal ingestion. As seen in Figure 39, at 150 grains of water per pound of air the improvement due to the duct modification is very nearly equal to the improvement due to the A5 increase although the duct modification has clearly greater effect at higher altitude.

The A5 increase has little or no effect at the low Reynolds Number Index where additional margin is needed. In order to test this, the snap-up maneuver was performed in three different duct configurations, i.e., the #1 aft horizontal vane at 2 degrees to WL (0 degrees angle of incidence), the #2 forward horizontal vanes at 2 degrees angle of incidence and with the vane support strakes only. That the duct modifications had very little

1.11 EFFECT OF A5 INCREASE ON ENGINE STALL MARGIN (Continued)

impact and that the A5 modification had little effect is shown in Figure 36 which is the upper part (N_2) of the P&WA "Church Window" curve. All three versions stalled in about the same region as the unmodified engine. Expressed in terms of flight parameters, these data are shown again in Figure 37. The increased scatter is due to variations in test day ambient temperatures but the data again evidences no significant improvement in basic stall margin at low RNI.

In order to visualize the effects of ambient temperature and provide flight planning data for compressor stall sensitivity the lower limits of compressor stall boundaries have been superposed on the low velocity profile of the aircraft in Figure 38. This curve was developed from Figure 36 by overplotting operating lines at constant Mach number for each ambient condition. The RNI at the intersection of these operating lines and the probable stall line can then be converted to altitude for that Mach number. Curves in Figure 38 apply to both the unmodified engine and that with the A5 increase. The balance of the engine program is pending OCAMA approval at this time, which includes testing of revised afterburner meters and continuing efforts with P&WA to improve stall margin at low RNI.

As of 10 March 1970, information has been received that the turbine nozzle area measurement may be in error. This has been coordinated with Pratt & Whitney and OCAMA (Mr. George Davis, OCNTTJ) to determine, if any, the impact on this program. The improved technique consists of a more accurate vane profiling to establish vane class and the concurrent elimination of the second area measurement of the assembly. Mr. Davis points out that the more accurate determination of vane class shows random errors in the older technique. However, the second determination of nozzle area with the old system helped to minimize accumulated error. Therefore, while it is felt that the new system is inherently more accurate, this should not alter the results of this program. The history of engine "C" shows at least three separate measurements of turbine nozzle area from initial build up at OCAMA to hot section inspection at SAAMA, followed by premodification inspection at OCAMA. The gas generator characteristics following modification agree with predictions and a new class of vanes was struck for this modification. The possibility of erroneous, turbine nozzle area measurement may explain some past and recent "stall prone" engines but again it is not believed that the results of this program are affected.

Engine calibration and trim data developed during this program are presented in Volume II, Appendix VI, of this report. It should be noted that the trim curve generated is unique to this engine and P_{t_7} system.

1.12 TEST RESULTS AS AFFECTING THE F-102 FLIGHT MANUAL

In the existing Flight Manual, procedures and explanations concerning compressor stall represent an accumulation of information from diverse sources. Findings of this compressor stall program are more authoritative and should, properly, supplant the earlier sources as the basis for the Flight Manual. Accordingly, recommended revisions to the Flight Manual congruous with the compressor stall program are included in this report.

As stated in the final report of the investigative phase (Reference A), most of the putative sources of compressor stall in the F-102 were not corroborated. These include afterburner initiation or termination, fuel control temperature bias, inlet flow disturbances (due to high angle of attack, etc.) or pressure distortion. In the minor failure category, failure of the intercompressor bleed valve caused an easily distinguishable N_1 stall at low engine rpm and a rich afterburner meter precipitated an N_2 stall somewhat earlier than would have occurred due to basic engine margin. No other systems failures caused compressor stalls. Major failures such as partially destroyed (burned) turbine sections are known to cause persistent compressor stalls but were not tested. In the non-failure category only two sources of compressor stall were found: enthalpic distortion in storm caused ice crystals and low Reynolds Number Index. Causes of these are discussed in the final report on the investigative phase (Reference A) and the cure for ice crystal induced compressor stalls is more fully discussed elsewhere in this report.

It is, therefore, clear that only those explanations and procedures applicable to the real causes of compressor stall are pertinent to the Flight Manual. Conversely, passages in the existing Handbook which imply other causes or confute findings of the compressor stall program should be deleted.

Appendix II of this report details recommended changes to the existing Flight Manual.

1.13 PILOT COMMENTS

As a result of the compressor stall fix program, new information was obtained which is of value to pilots and engineers alike. This information, as well as test methods and techniques used to acquire it, is discussed under the major headings of thunderstorm penetrations and snap-up maneuvers.

1.0 DISCUSSION (Continued)

1.13.1 Thunderstorm Penetrations

The purpose of performing thunderstorm penetrations was to evaluate effects of various inlet duct flow control devices on F-102/J-57 compressor stall susceptibility in severe storms at the highest possible ice concentration. Basic configuration of the aircraft was a standard F-102A with 230-gallon external tanks, no IR probe, and a J-57 engine with a 3% increase in A5. Various configurations were investigated as specified in Paragraphs 1.2 through 1.7. Preparation, flight planning, and airspace usage are covered in Appendix I under flight operations. After arrival in the storm area, a particular storm to be penetrated was selected by the pilot on the basis of size (height and breadth), anticipated ice concentration (judged by contrast and composition), and stage of development. Another important consideration was to select a storm for which a predetermined heading would exit the cloud quickly after encountering the stall. The altitude selected to penetrate was based on finding the level with highest ice concentration but above the level at which hail might be encountered. All weather flights were flown with intention of forcing that particular duct-engine configuration into a stall, therefore the worst part of the storm was deliberately penetrated. Several other non-stall penetrations were performed to compare distortion at lower levels of ice concentration.

A summary of duct configurations, flight test conditions, and stall occurrences is included in Table I-2. A technical discussion of test results is included in Paragraphs 1.2 through 1.11. With increased stall margin of the modified engine at high RNI (Paragraph 1.11), no stalls were encountered flying over or around storms or in cirrus clouds. In every case, engine stalls were N_2 rotor stalls and were encountered only in the cumulus portion of the storm. No engine flameouts occurred as a result of a compressor stall nor were any engine shutdowns required to prevent an overtemperature condition. Although a throttle chop to idle was required to break several N_2 stalls (Flights 61, 64, 80) and prevent an overtemperature, stall or overtemperature conditions did not re-occur at stabilized idle while departing the cloud. These data are not conclusive as to further stall occurrence at stabilized idle since storms were usually exited within several minutes after breaking the stall. Effect of remaining in the storm and resetting cruise power was not investigated.

1.0 DISCUSSION (Continued)

1.13.1 Thunderstorm Penetrations (Continued)

Engine stalls during thunderstorm penetrations were of two typed: random bangs (usually one or two - sometimes three or four) and multi-pulse stalls. Random bangs were recognized by intermittent bangs spaced at 1 to 1½ seconds and were caused by high thermal distortion and a transient condition of high ice concentration which immediately cleared itself. For this stall, thrust setting was left constant and ignition was energized as a precaution to prevent flameout until engine operation returned to normal. This type stall never caused an over-temperature condition. Multi-pulse stalls were recognized by rapid bangs (2 to 3 cps) and rising EGT and were caused by a high steady state combination of ice concentration and thermal distortion. It was necessary to chop the throttle to idle to break this type stall and prevent an over-temperature condition. Ignition was also energized to prevent flameout and the nose was lowered to maintain airspeed. Both stall types always occurred in the region of greatest turbulence. Three factors caused this: one, the region of greatest turbulence in a thunderstorm is usually the area of greatest moisture concentration; two, cloud turbulence can induce thermal distortion at the engine (Paragraph 1.8); and three, sudden or violent maneuvers induced by turbulence in an ice cloud reduce stall margin (Paragraph 1.9).

It was not determined during the program if use of ignition did in fact prevent engine flameout during a compressor stall since no stall recoveries were performed without use of the ignition. The automatic ignition is activated by a rate change to the N_2 compressor discharge pressure of not less than 1 psi/sec, and will not necessarily actuate as a result of a compressor stall without flameout. The auto-ignition system never did sense enough burner pressure drop at the stall to actuate the system before the pilot reacted.

Pilot conclusions, based on cockpit look and feel data, are that the combined configuration of No. 4 forward support strakes plus No. 1 vane at 6 degrees relative to water line (Paragraph 1.6) is the best configuration. This configuration enabled the F-102/J-57 to accept the greatest level of ice concentration and turbulence prior to engine stall. The highest ice concentration experienced in this configuration was 198.8 grains per pound of

1.13.1 Thunderstorm Penetrations (Continued)

air at 29,000 feet (Paragraph 1.6). This weight concentration at this altitude is equivalent to meteorological units of 13.23 grams per cubic meter which is approximately twice the annual mean maximum thunderstorm concentration world-wide. Thus, this configuration of inlet duct flow control devices and modified engine (increased A5) should be free from stall in all but the most severe storms.

Pilot recommendations are to update the F-102A Flight Manual procedures and description as outlined in Appendix II.

1.13.2 Snap-Up Maneuvers

Purpose of performing snap-up maneuvers was to determine effect of the engine A5 increase on compressor stall margin at low Reynolds Number Index (RNI) and to check pressure distortion at the compressor face. Basic configuration of the aircraft was a standard F-102A with no IR probe and J-57 engine with a 3% increase in A5. Various inlet duct flow control devices were installed as specified in Paragraph 1.11. Preparation, flight planning, and airspace usage are covered in Appendix I under flight operations. Snap-up procedure was as follows: (1) commence descent from 40,000 feet at maximum power; (2) at 35,000 feet and Mach 1.1, rotate to 45 degree nose-up pitch attitude (less than 2g) and hold; (3) at 170 KIAS, apply forward and aft stick pulse followed by right rudder jab to simulate interceptor pilot trying to center the steering dot; (4) recover as per F-102A Flight Manual procedures. All snap-up flights were performed within glide distance of at least a 12,000 foot runway.

A summary of duct configurations, flight test conditions, and stall occurrences is included in Table I-2 (Flights 50, 58, 60). A technical discussion of test results is included in Paragraph 1.11. On each of the three flights, N_2 rotor stalls were encountered as recognized by random bangs (1 to 3 per second) and rise in EGT to approximately maximum allowable temperature. On two flights (50 and 58), N_2 stall progressed into a total compressor stall recognized by a rapid series of low energy bangs (15 cps) and rumbling, and a rapid rise in EGT requiring

1.0 DISCUSSION (Continued)

1.13.2 Snap-Up Maneuvers (Continued)

engine shutdown to prevent exceeding engine limits. From Figures 36 and 37, it can be seen that stall margin of the engine at low RNI was not changed with increased A5. Figure 38 would help a pilot planning a snap-up flight to avoid stall by proper selection of flight conditions. Stall recovery techniques and results were as follows:

Flight 50 - At stall, throttle snapped out of burner to military thrust, ignition energized, stall progressed into total compressor stall with EGT climbing through 696 degrees C, as the engine was shut down. Airstart was satisfactory at 35 - 38,000 feet and 275 KIAS.

Flight 58 - At stall, throttle snapped out of burner to idle, ignition energized, N_2 stall broken, recovered to wings level 20 degrees nose down, airspeed below 170 KIAS, another N_2 stall followed by total compressor stall with EGT climbing through 688 degrees C, as the engine was shutdown. Airstart was satisfactory at 38,000 feet and 270 KIAS.

Flight 60 - At stall, throttle snapped out of burner to idle, ignition energized, N_2 stall broken, recovered to wings level 40 degrees nose down to recover airspeed faster and get to lower altitude quicker, no engine shutdown required.

On the basis of these results, the best engine stall recovery technique during a snap-up was to snap the throttle from burner to idle, while energizing the ignition, to break the N_2 stall. Then recover the aircraft as quickly as possible by 45 degrees nose low attitude to a lower altitude and higher airspeed, where RNI is higher, in order to prevent another stall. This procedure was modified slightly to require checking for EGT rise to maximum allowable before snapping to idle to standardize compressor stall procedures for snap-up and thunderstorm induced stalls (refer to Appendix II). When the engine was shut down to prevent over-temperature, considerable duct rumble was experienced while gliding to a lower altitude for restart. An airspeed of 250 - 275 KIAS, as recommended by the stall investigation program, was used

1.0 DISCUSSION (Continued)

1.13.2 Snap-Up Maneuvers (Continued)

for restart after each of the two shut downs. Altitude for successful airstart was between 35 - 38,000 feet. It would be helpful for the interceptor pilot to have a second EGT gage jewel presented on the radarscope to allow EGT monitoring in event of compressor stall during snap-up flight.

Pilot conclusions are that the engine A5 increase had no effect on stall margin at low RNI.

Pilot recommendations are:

1. To update the F-102A Flight Manual procedures and description as outlined in Appendix II.
2. To incorporate a second EGT jewel on the radar scope ring.
3. To include a stall limit versus temperature figure (similar to Figure 38) in the F-102A Flight Manual.

2.0 CONCLUSIONS

2.1 The 3% increase in the first stage turbine nozzle area (A5) of the engine was not exhaustively tested due to lack of contract funding by OCAMA. Limited testing was accomplished as by-product to the ice crystal ingestion fix, however. The A5 increase improved compressor stall margin at higher Reynolds Number Index which improves the engine tolerance to ice crystals. The A5 increase did not improve compressor stall margin at low Reynolds Number Index and hence did not improve the F-102 compressor stall tolerance for high altitude low speed flight with sub-standard ambient temperatures. Both level flight and snap-up maneuvers are affected.

2.2 The crystal induced compressor stall fix development was satisfactorily concluded. In order of testing, the following duct configurations were evaluated:

1. Configuration #1, aft horizontal vane, tested at 2, 4, and 6 degrees to water line
2. Configuration #2, forward horizontal vanes, tested at 0 and 2 degrees angle of incidence, plus support strakes for configuration #1
3. Configurations #1 and #2 support strakes only (no vanes installed)
4. Combined configuration: configuration #1 at 6 degrees to water line plus support strakes for configuration #2
5. Configuration #4, support strakes for configuration #2
6. Clean ducts - support strake attachment holes filled with screws
7. Configuration #5, three-inch strakes at 2 degrees
8. Configuration #6, one and one-half inch strakes at 2 degrees
9. Clean ducts - support strake attachment holes filled and smoothed.

In order of superiority, the combined configuration of configuration #1 at 6 degrees angle relative to water line plus the support strakes for configuration #2 is clearly the superior configuration (Frontispiece), followed by configuration #2, followed by the support strakes of configuration #2. The combined configuration sustained ice concentrations of nearly 200 grains per pound of air and reduced thermal distortion from more than 70% to less than 10%. Excellent control of externally induced thermal distortion was demonstrated at a minor increase in pressure distortion downstream of the vane. The combined configuration modification to the inlet duct is a satisfactory solution to the ice crystal induced compressor stall problem and effectively eliminates the inlet as a contributor to this problem.

3.0 RECOMMENDATIONS

3.1 Install duct fix consisting of aft horizontal vane at six degree angle to water line, and support strakes for the forward horizontal vanes into the F-102A fleet as part of the weather stall margin fix.

3.2 In accordance with the test plan for this program, test the engine for evaluation and development of high pressure compressor stall margin associated with recommended modifications to improve the stall margin at low Reynolds Number Index.

3.3 If above recommended engine testing is not accomplished, it is recommended that the weather stall margin fix include the 3% increased A5 area.

3.4 Revise Technical Order 1F-102A-1 Pilots Flight Manual to reflect improved information concerning flight procedures related to compressor stall incidents. (Reference Appendix II.)

3.5 Add a parallel circuit to the exhaust gas temperature gage jewel so that the jewel is presented on the radar scope ring.

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17. F-102A Aircraft Incident Reports of F-102 Compressor Stalls, SAAMA, Kelly Air Force Base, Texas

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18. Test Program Plan for F-102A Compressor Stall Investigation, GD/Convair Report GD/C-AYC-68-002, July 1968

19. Problem Analysis, F-102A Compressor Stall Study, GD/Convair Report No. GD/C-AYC-68-001, May 1968

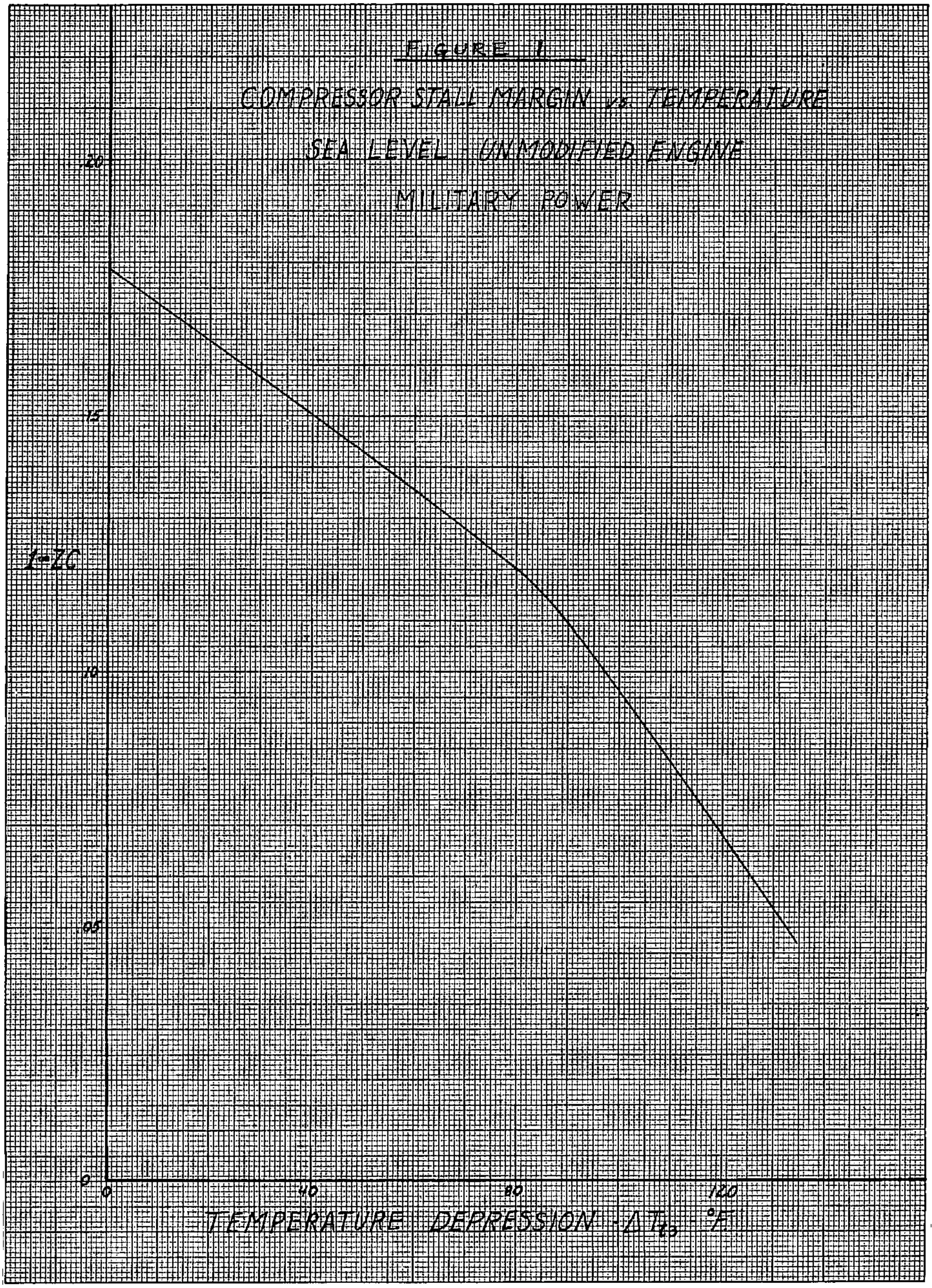


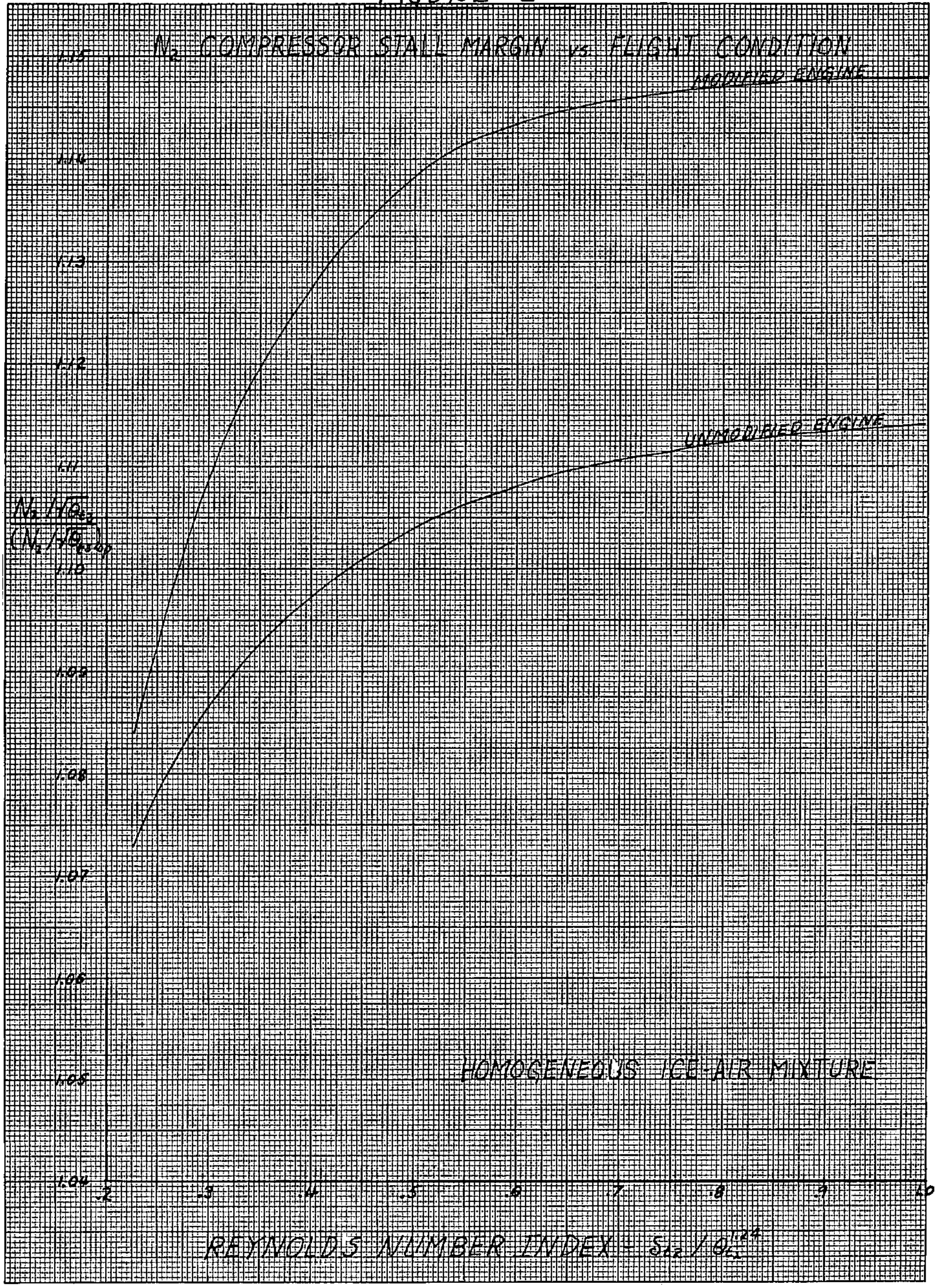
FIGURE 1

COMPRESSOR STALL MARGIN vs. TEMPERATURE

SEA LEVEL UNMODIFIED ENGINE

MILITARY POWER

FIGURE 2


N2 COMPRESSOR STALL MARGIN vs FLIGHT CONDITION
MODIFIED ENGINE
UNMODIFIED ENGINE
HOMOGENEOUS ICE-AIR MIXTURE
N2/√θt2 / (N2/√θt2)SOP
1.15
1.14
1.13
1.12
1.11
1.10
1.09
1.08
1.07
1.06
1.05
1.04
.2
.3
.4
.5
.6
.7
.8
.9
1.0
REYNOLDS NUMBER INDEX - δt2/θt2^1.24

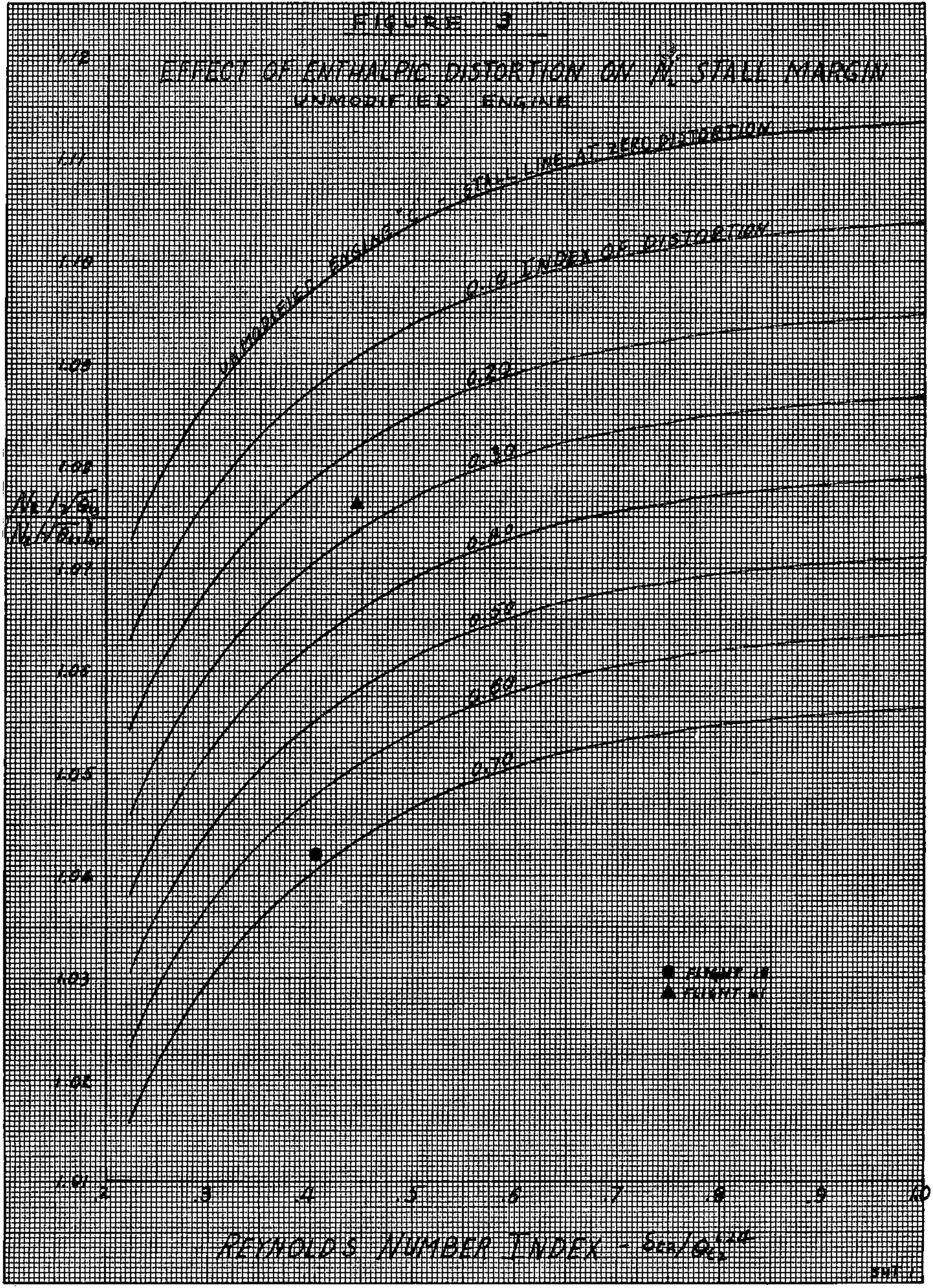


FIGURE 3

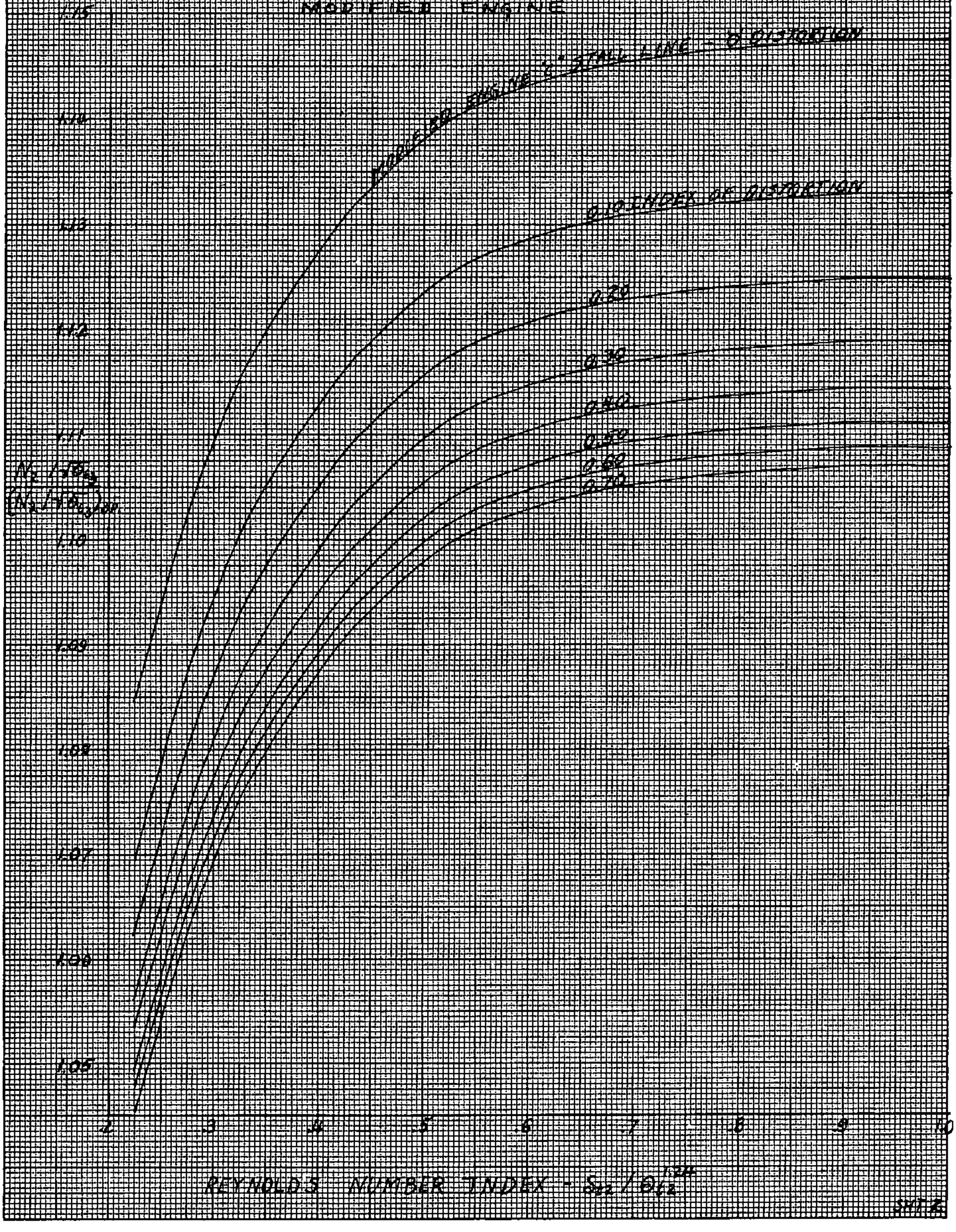
EFFECT OF ENTHALPIC DISTORTION ON N_L STALL MARGIN

UNMODIFIED ENGINE

FIGURE 4

EFFECT OF ENTHALPIC DISTORTION ON N_2 STALL MARGIN

MODIFIED ENGINE



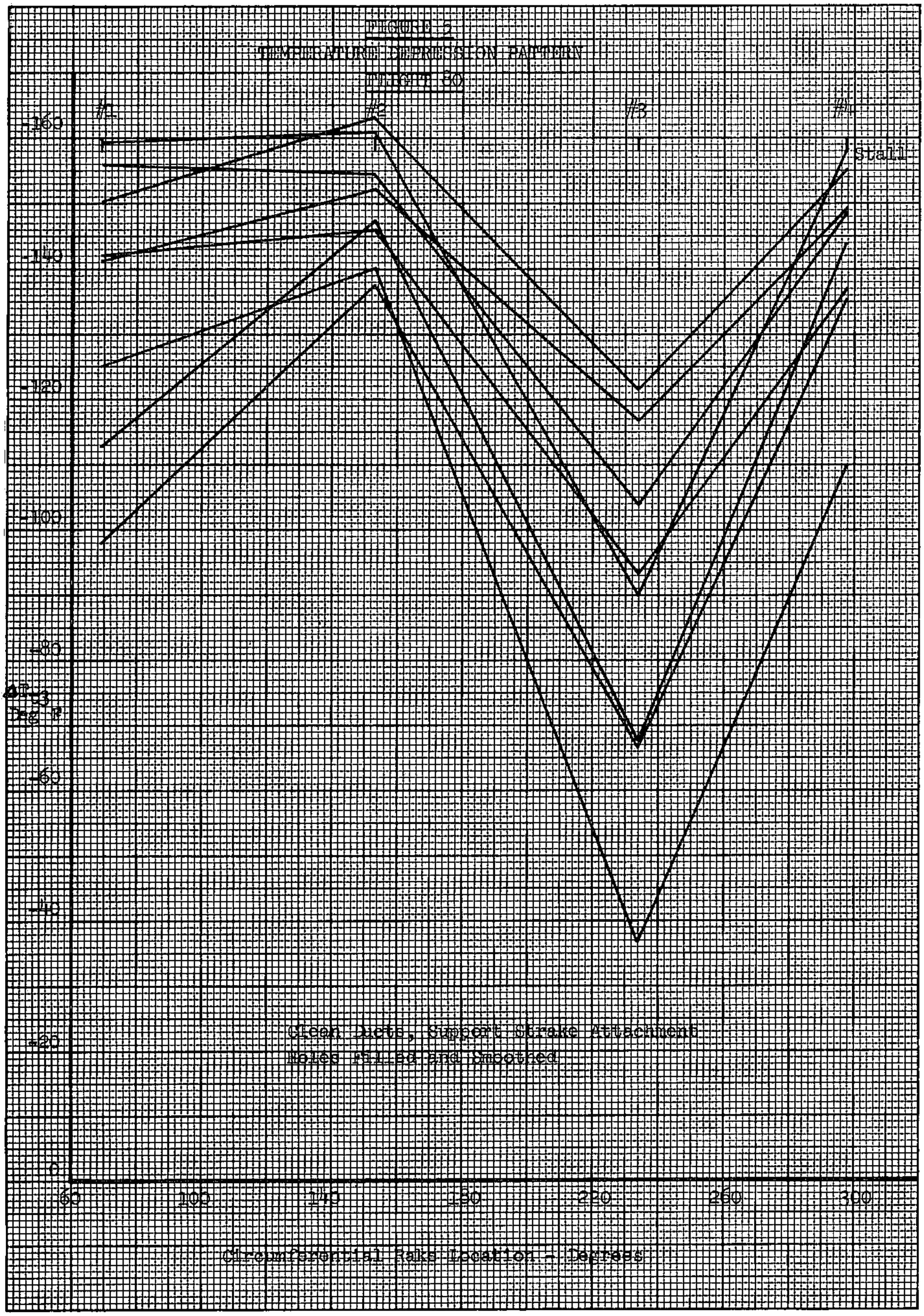


FIGURE 2

TEMPERATURE DEPRESSION PATTERN

FLIGHT 80

Clean Ducts, Support Strake Attachment Holes Filled and Smoothed

FIGURE 6

TEMPERATURE DEPRESSION PATTERN

FLIGHT 68

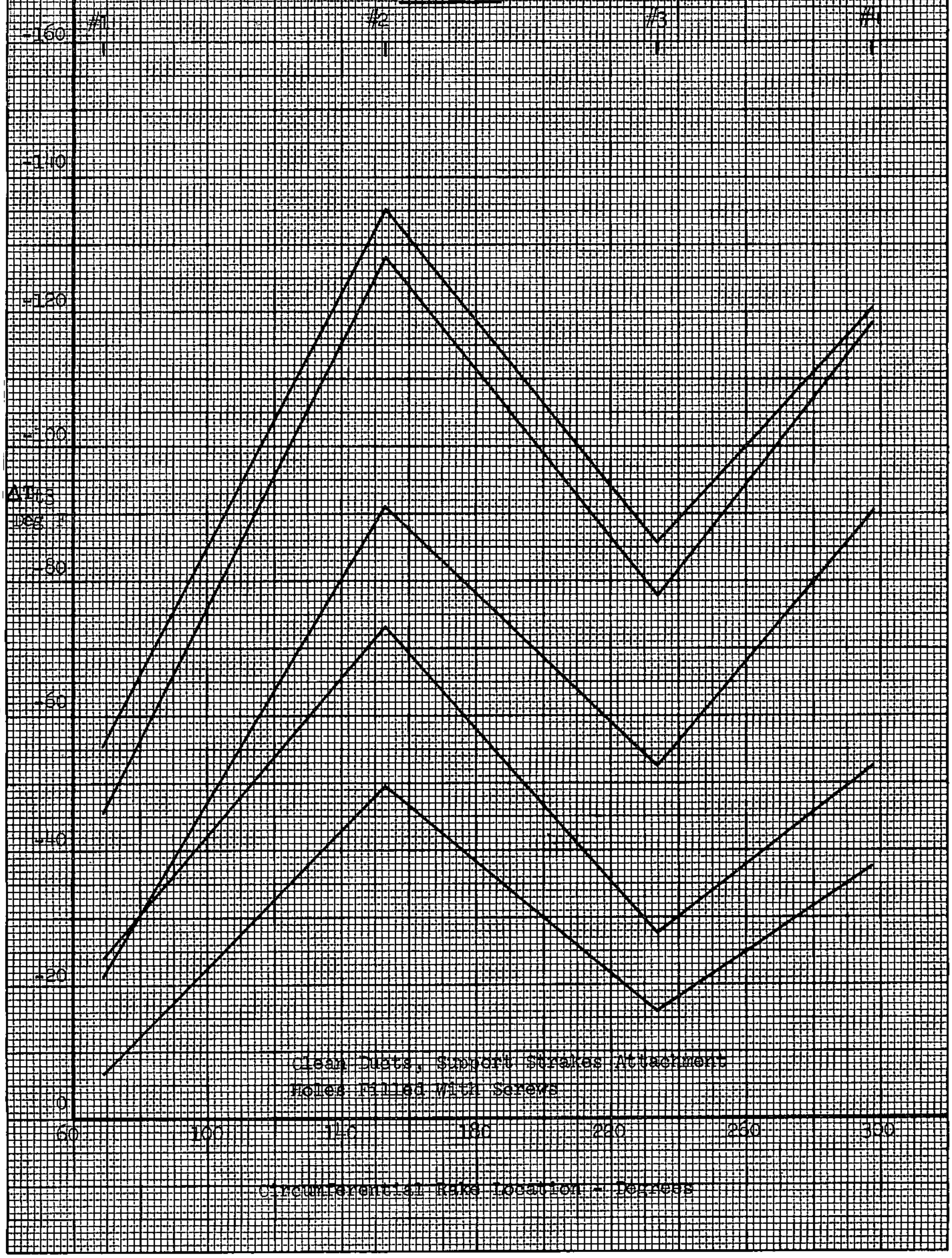


FIGURE 7

CONFIGURATION #1, AFT HORIZONTAL VANE

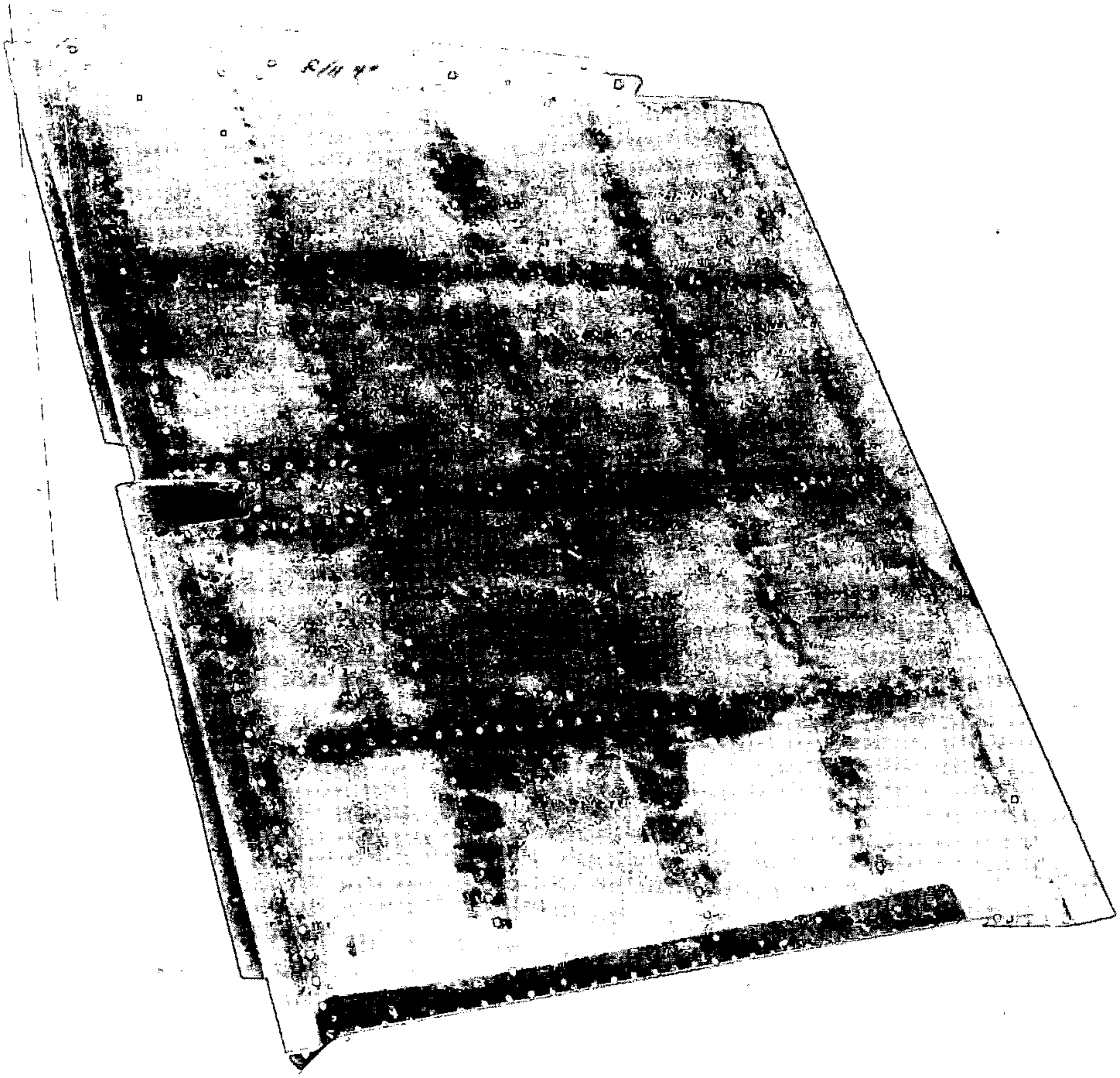


FIGURE 8

CONFIGURATION #1, AFT HORIZONTAL VANE

INSTALLATION AT TWO DEGREES NOSE-UP

VIEW LOOKING FORWARD

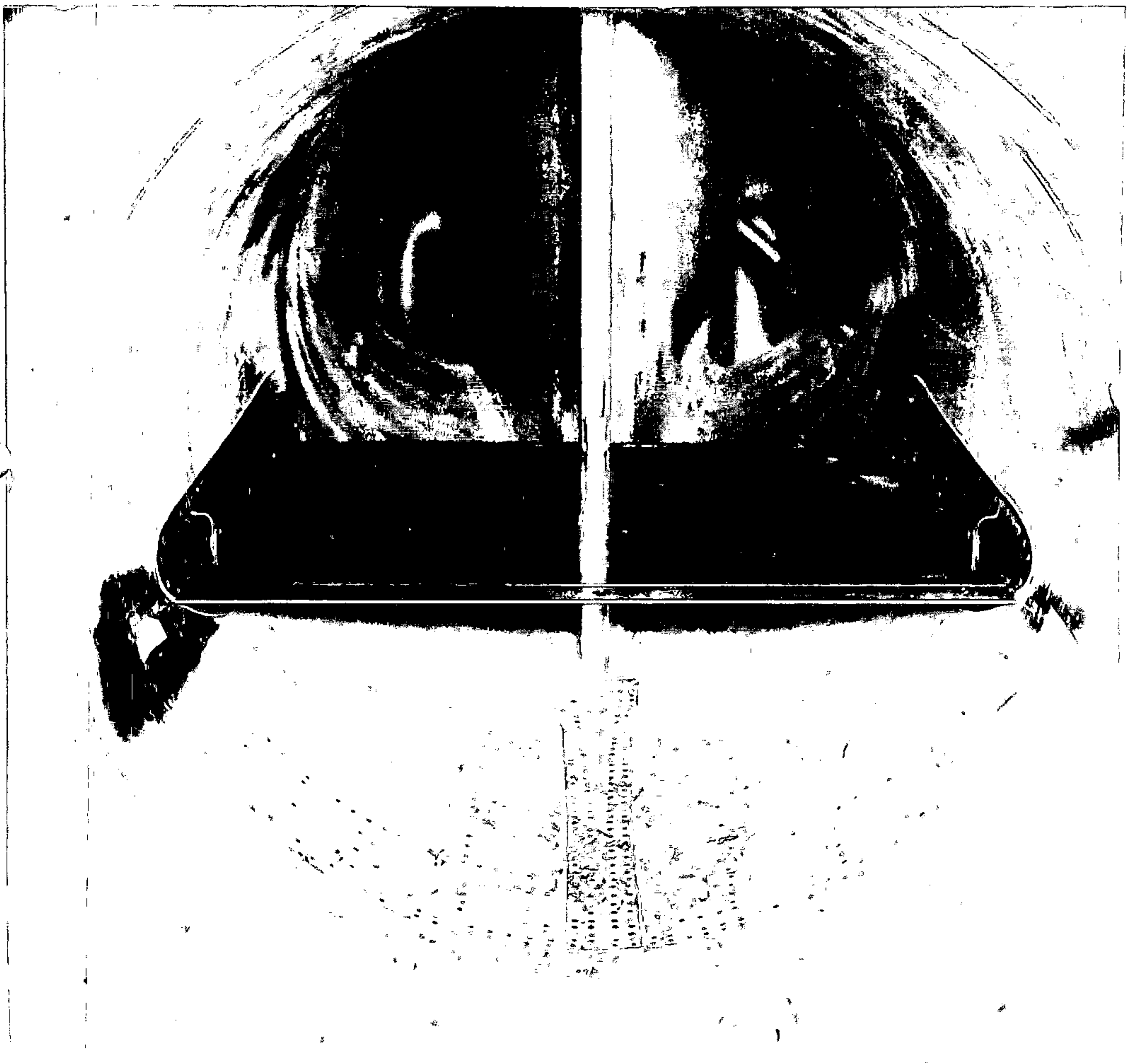


FIGURE 9

CONFIGURATION #1, AFT HORIZONTAL VANE

INSTALLATION AT TWO DEGREES NOSE-UP

VIEW LOOKING AFT

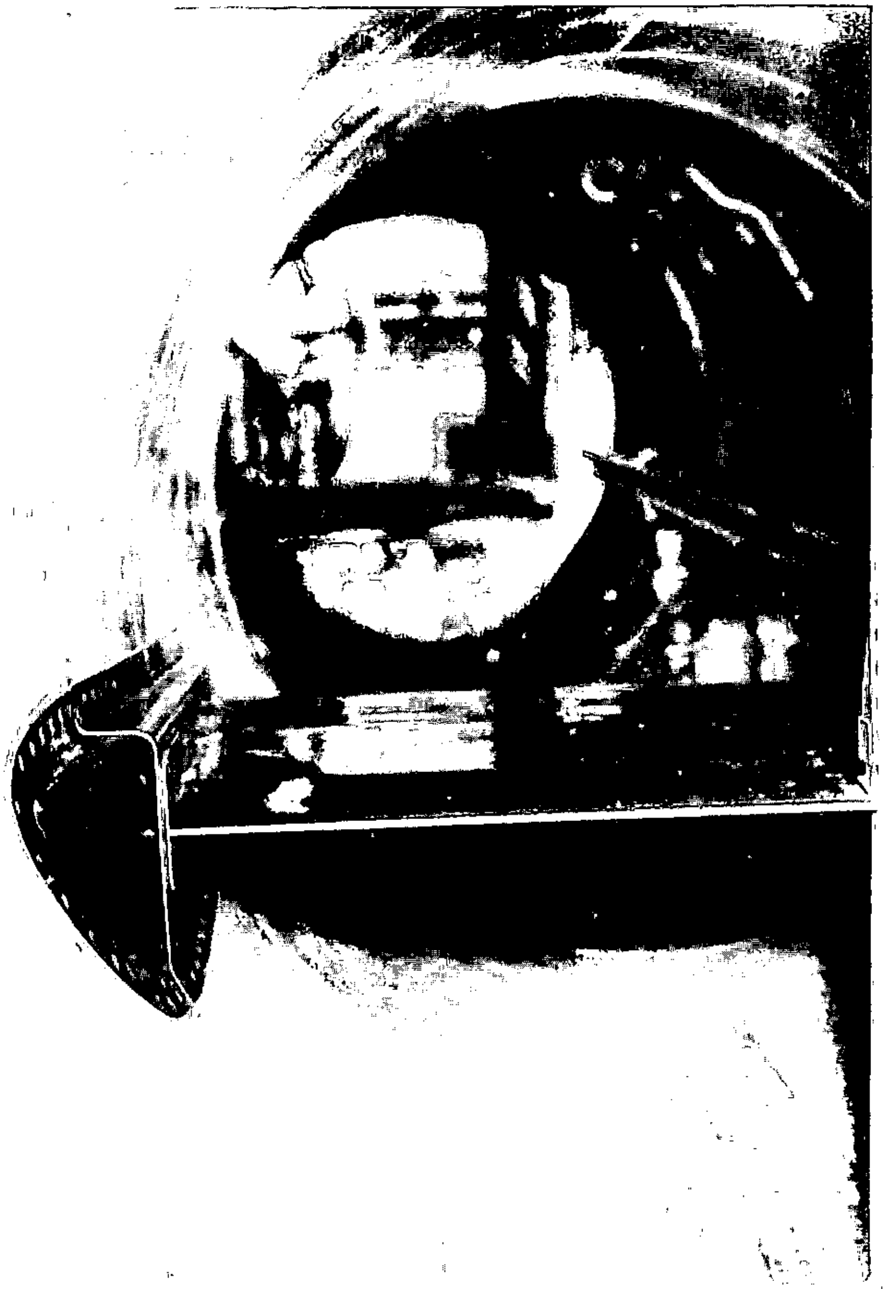
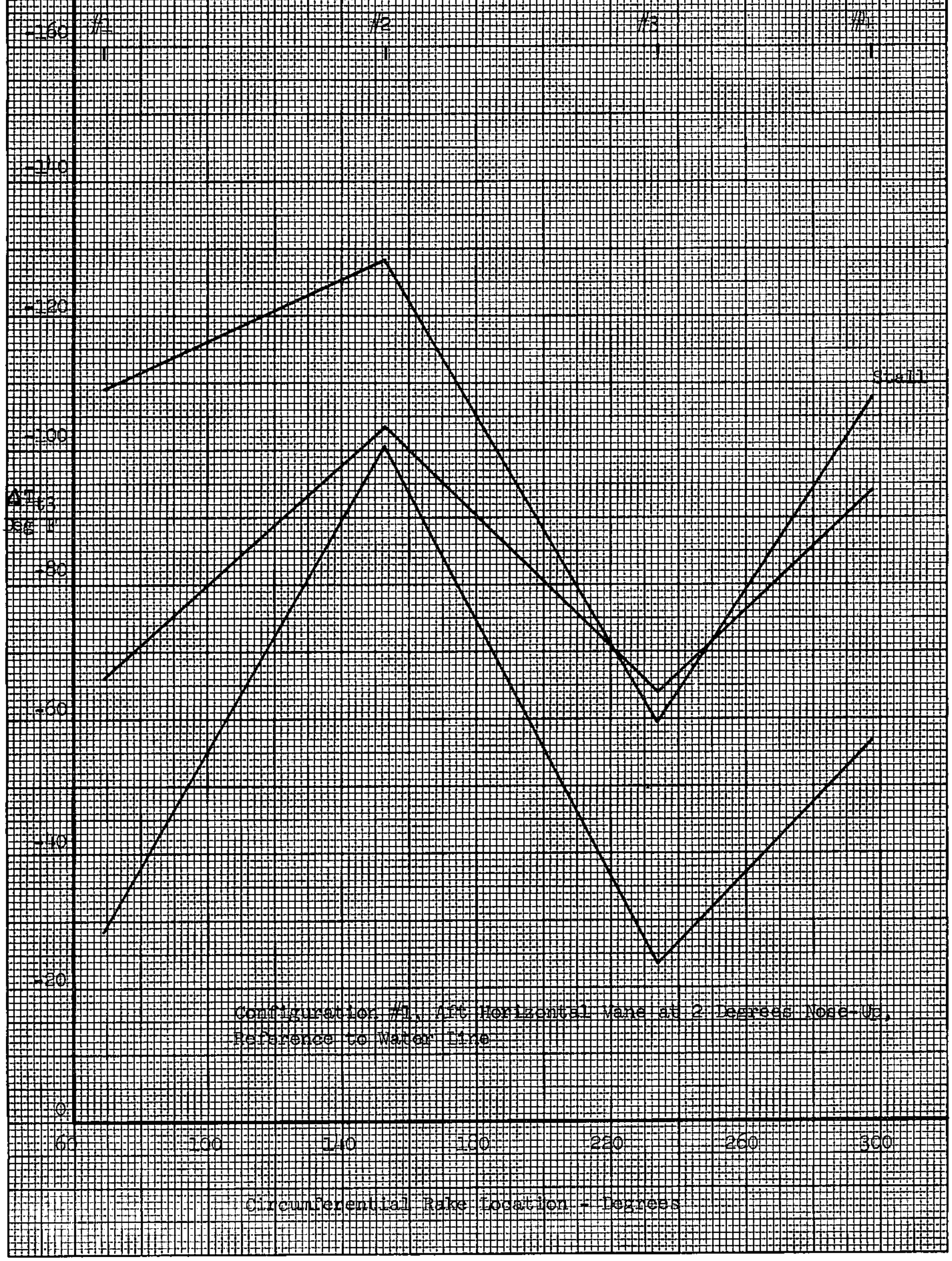


FIGURE 10

TEMPERATURE DEPRESSION PATTERN

FLIGHT 52



Configuration #1, Aft Horizontal Vane at 2 Degrees Nose-Up, Reference to Water Line

FIGURE 11

TEMPERATURE DEPRESSION PATTERN

FLIGHT 53

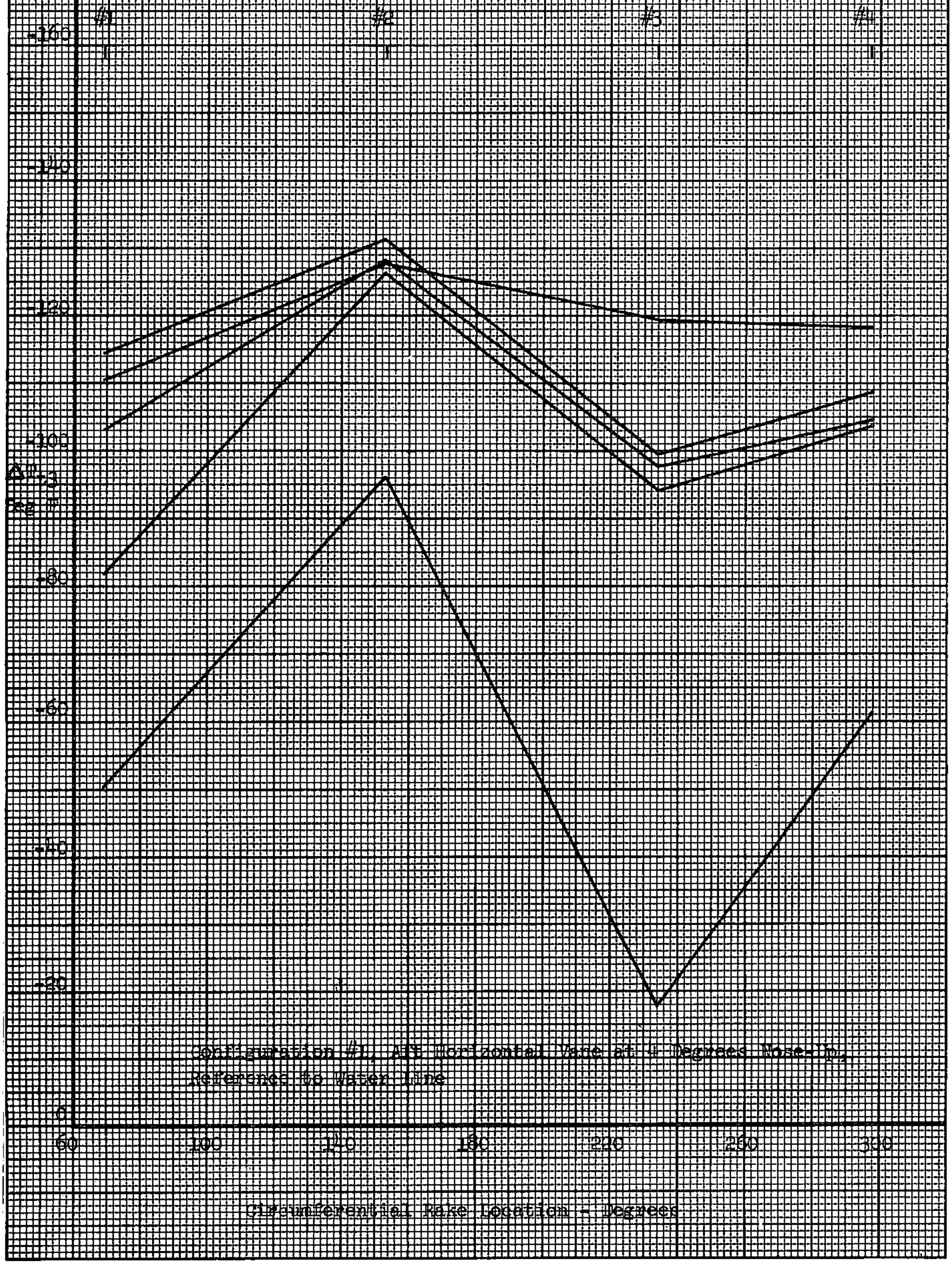
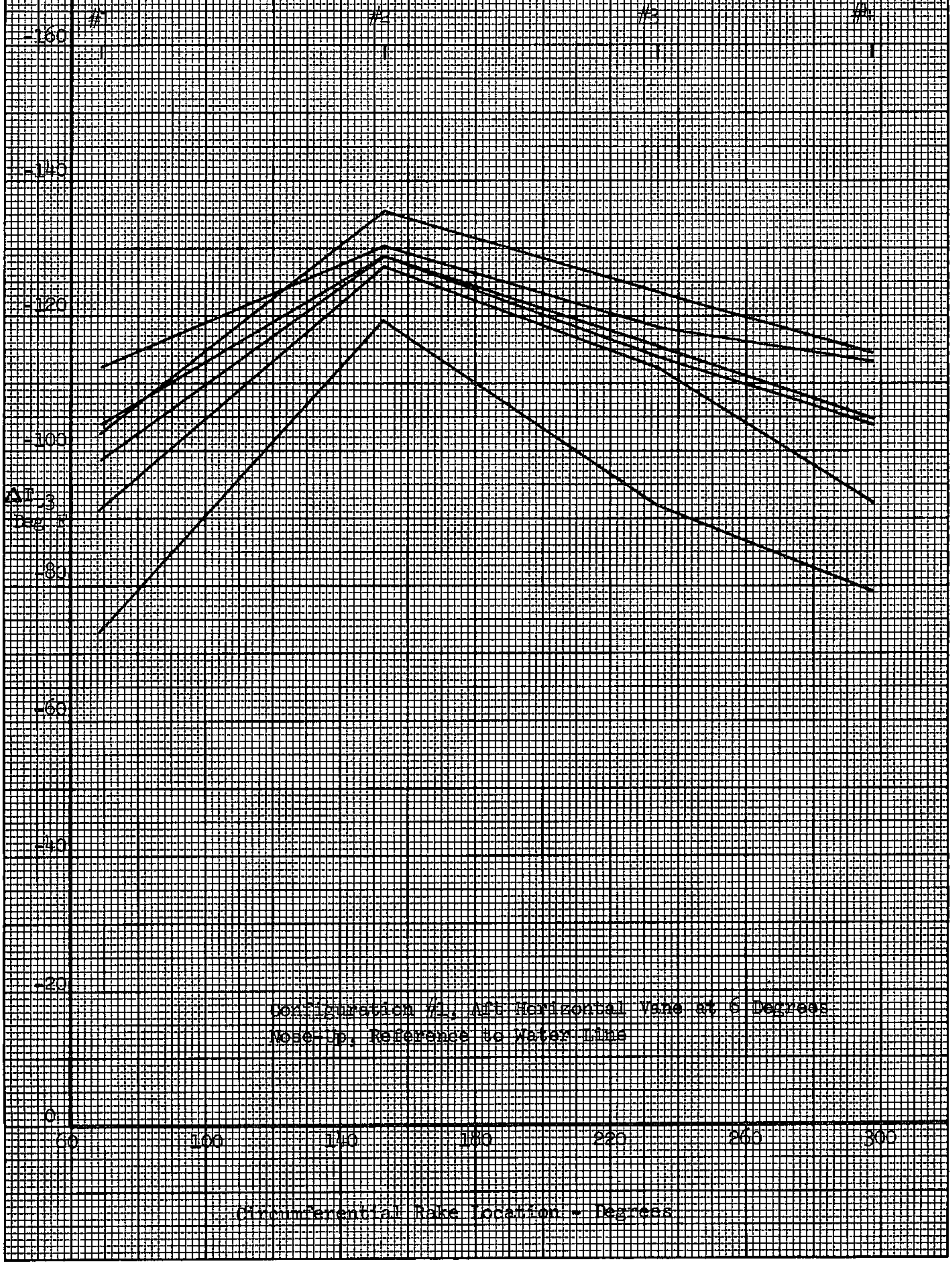


FIGURE 12

TEMPERATURE DEPRESSION PATTERN

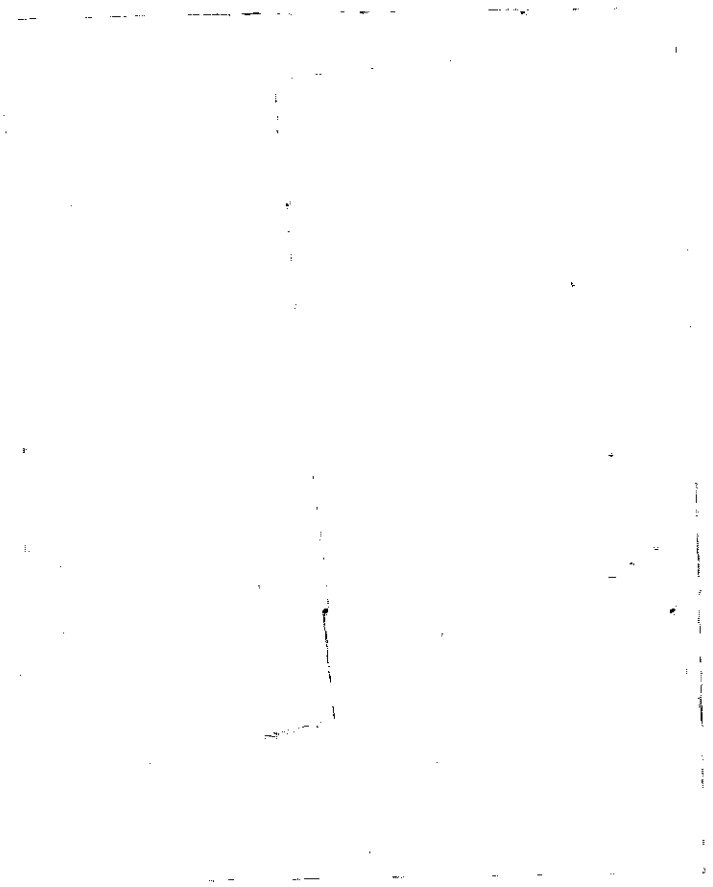
FLIGHT 54



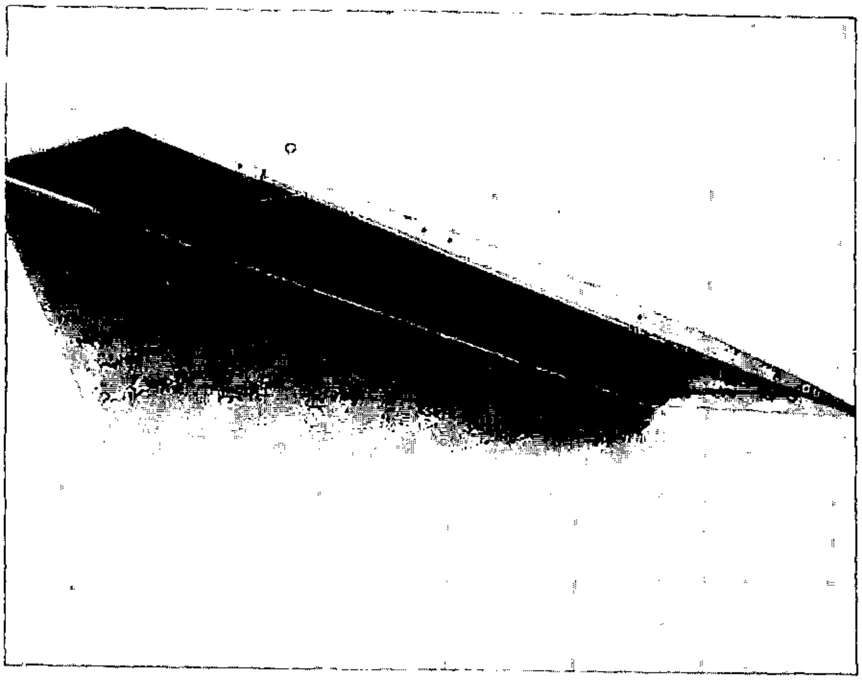
Configuration #1, Aft Horizontal Vane at 6 Degrees Nose-Up, Reference to Water Line

FIGURE 13

CONFIGURATION #2, FORWARD HORIZONTAL VANE



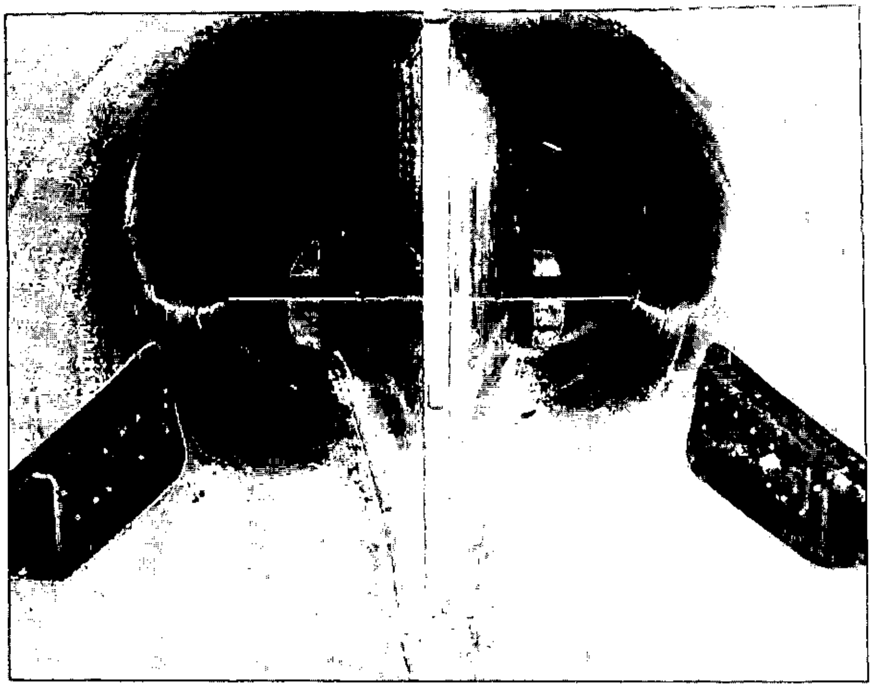
Aft End



Forward End

FIGURE 14

CONFIGURATION #2 INSTALLATION



View Looking Forward

(Also showing support strakes for Configuration #1)

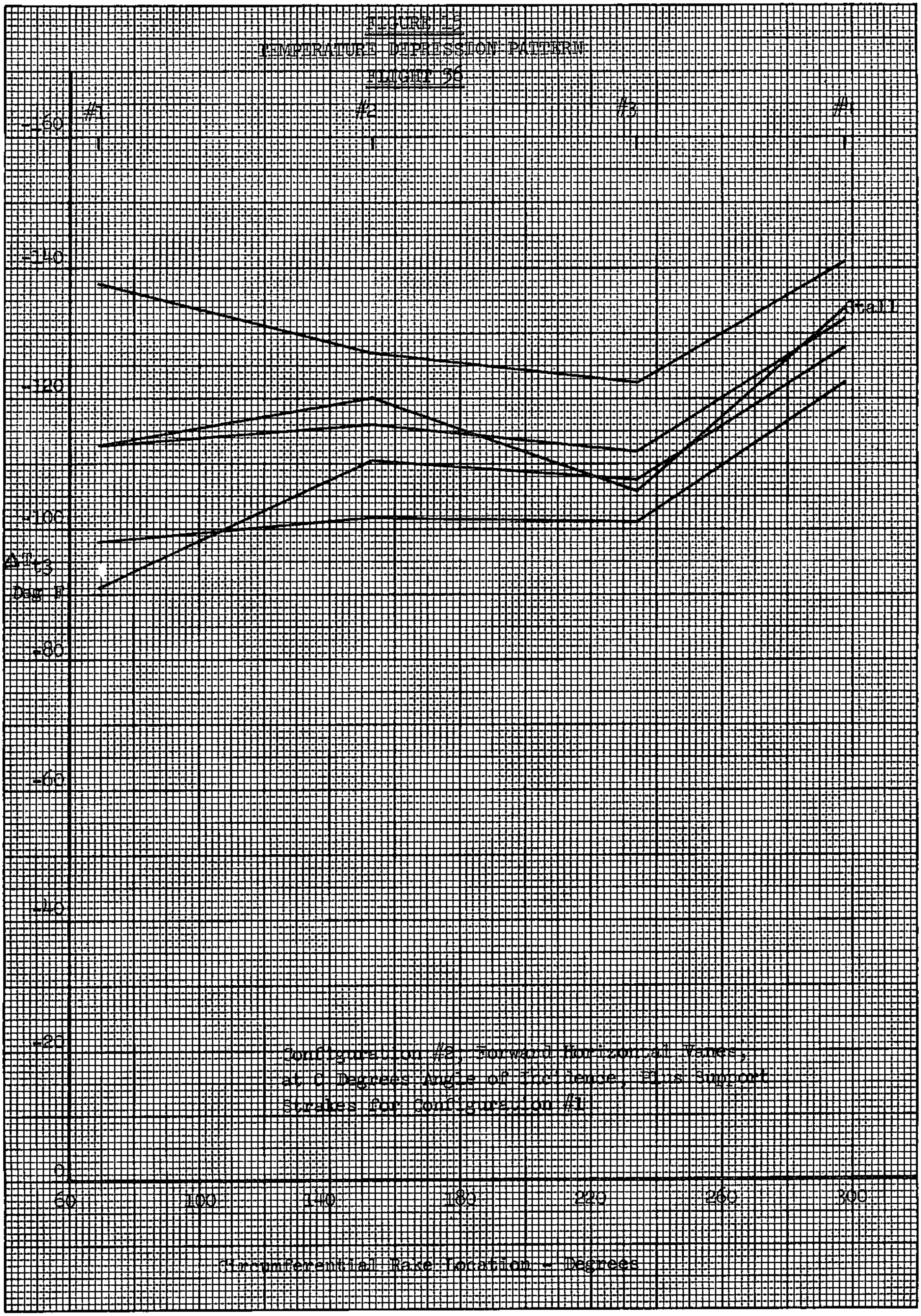
View Looking Forward

Support strake for left forward vane

FIGURE 12

TEMPERATURE DEPRESSION PATTERN

FLIGHT 56

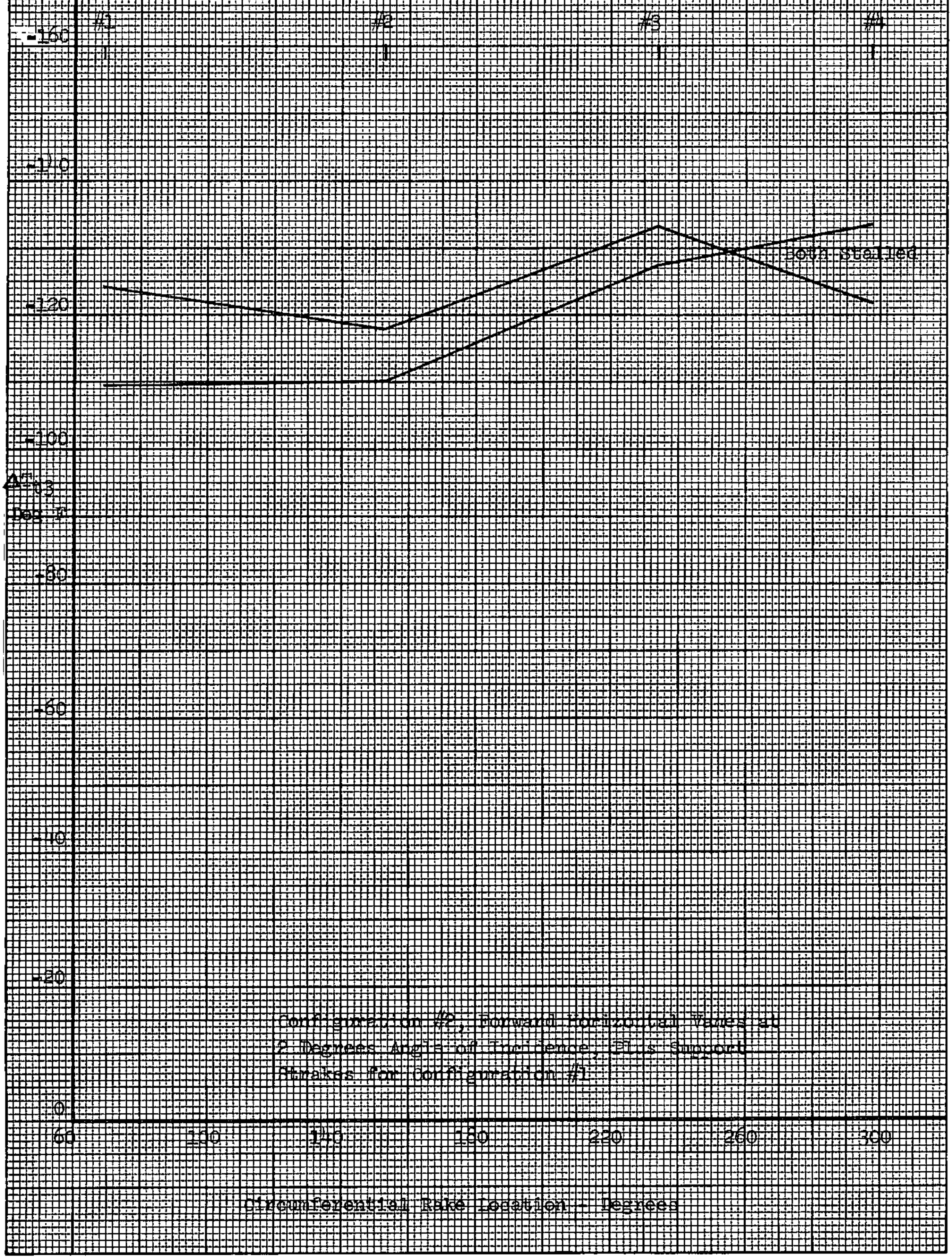


Configuration #2, Forward Horizontal Vanes, at 0 Degrees Angle of Incidence, Plus Support Strakes for Configuration #1

FIGURE 16

TEMPERATURE DEPRESSION PATTERN

FLIGHT 57



Configuration #2, Forward Horizontal Vanes at 2 Degrees Angle of Incidence, Plus Support Strakes for Configuration #1

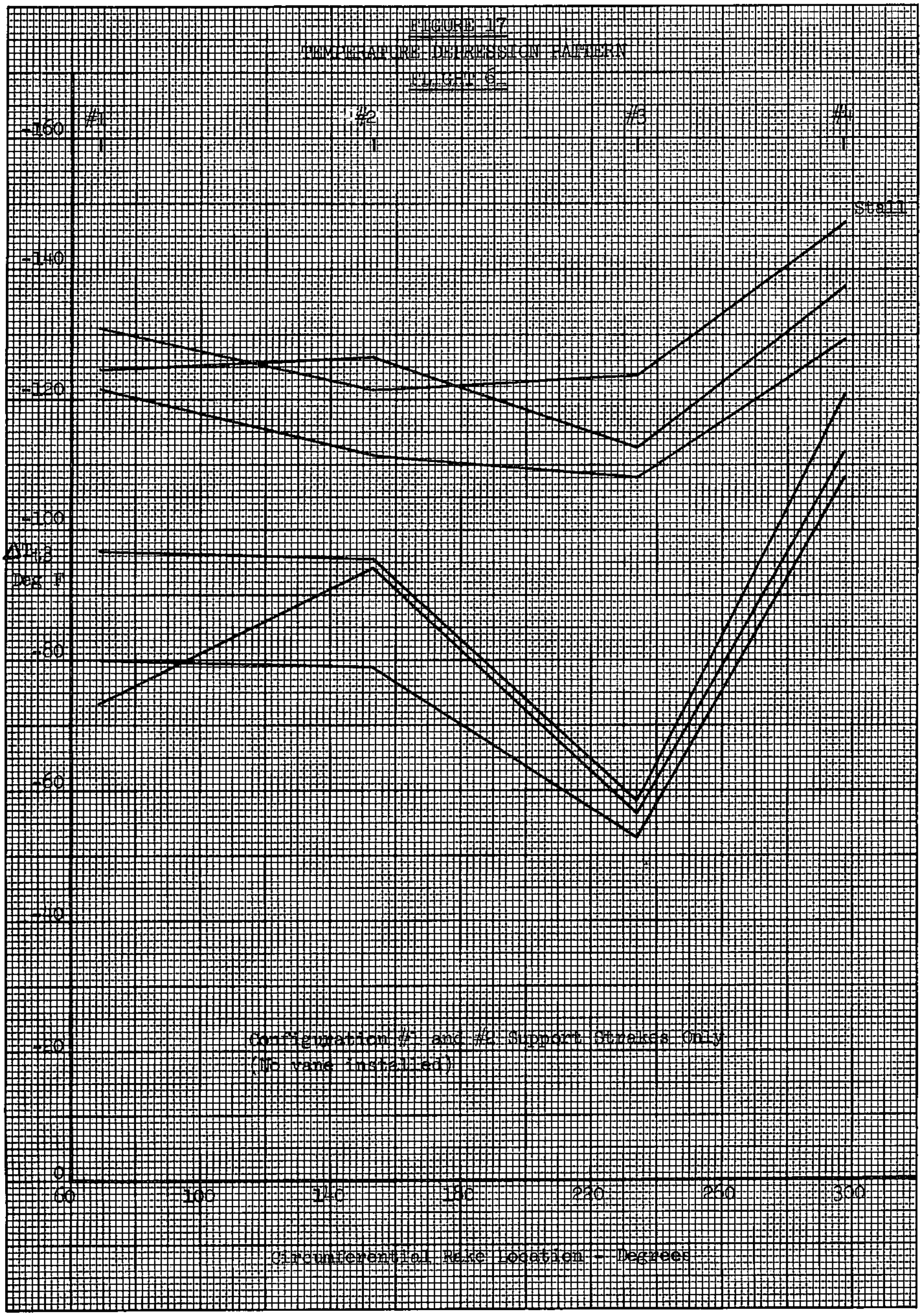
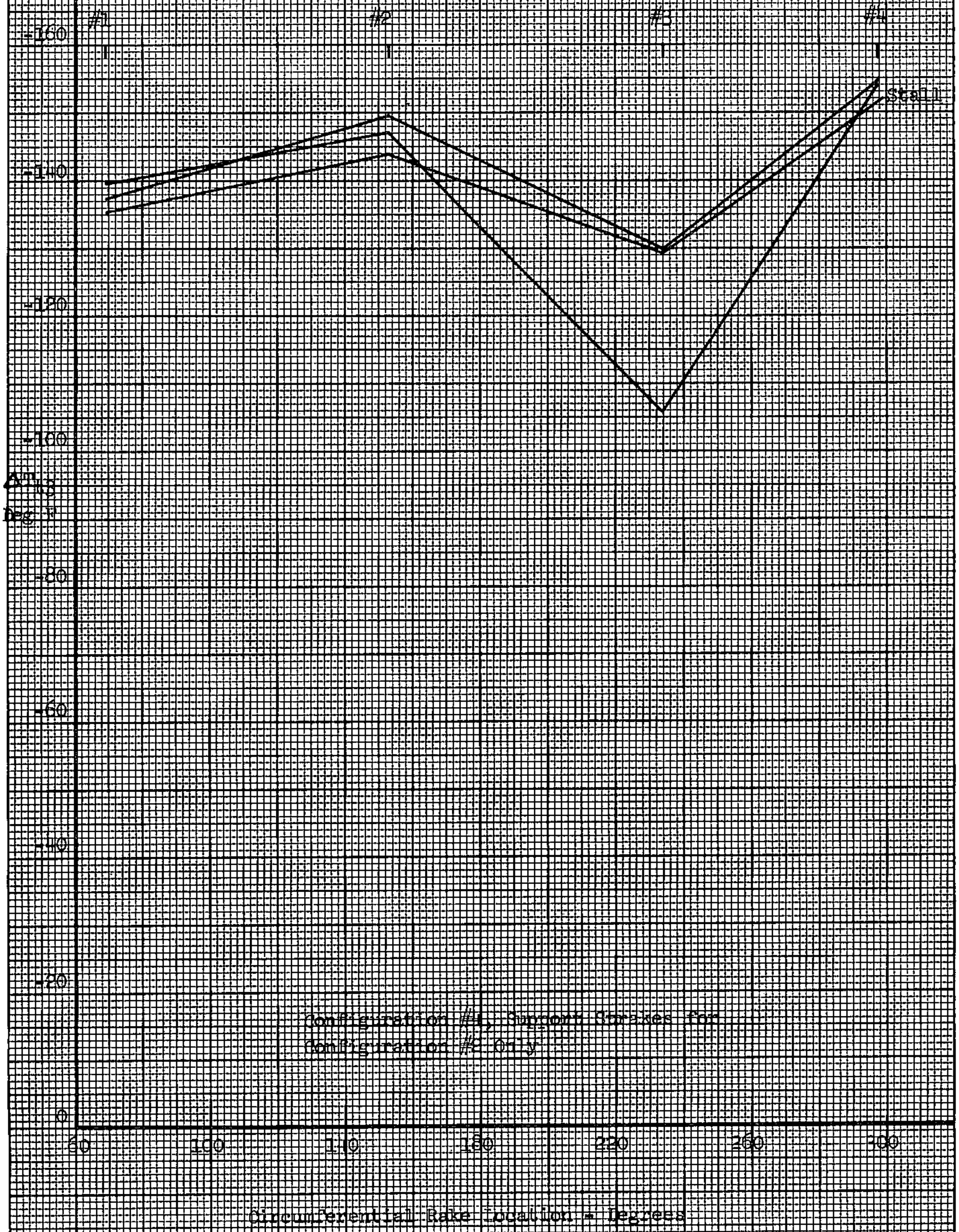


FIGURE 17
TEMPERATURE DEPRESSION PATTERN
FLIGHT 6

Configuration #1 and #2 Support Strakes Only
(No vane installed)

FIGURE 18
TEMPERATURE DEPRESSION PATTERN
FLIGHT 66



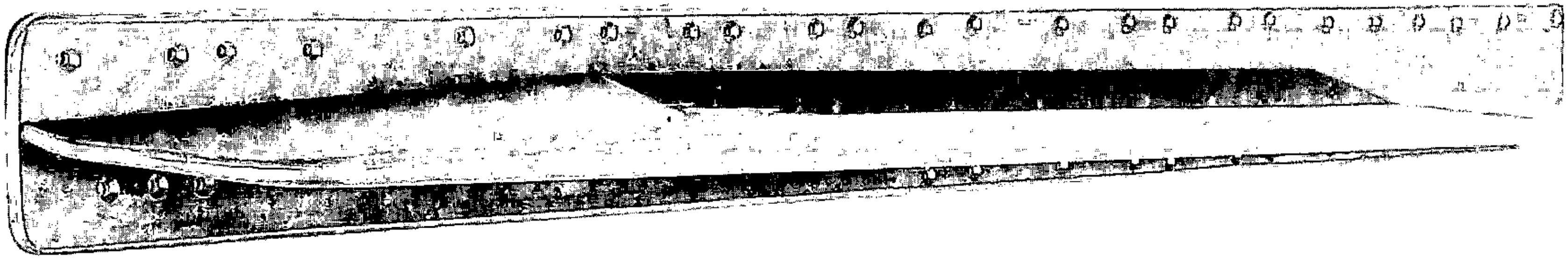


FIGURE 19

CONFIGURATION #5, FORWARD THREE-INCH STRAKE

FIGURE 20

CONFIGURATION #5, THREE-INCH STRAKE INSTALLATION

VIEW LOOKING FORWARD

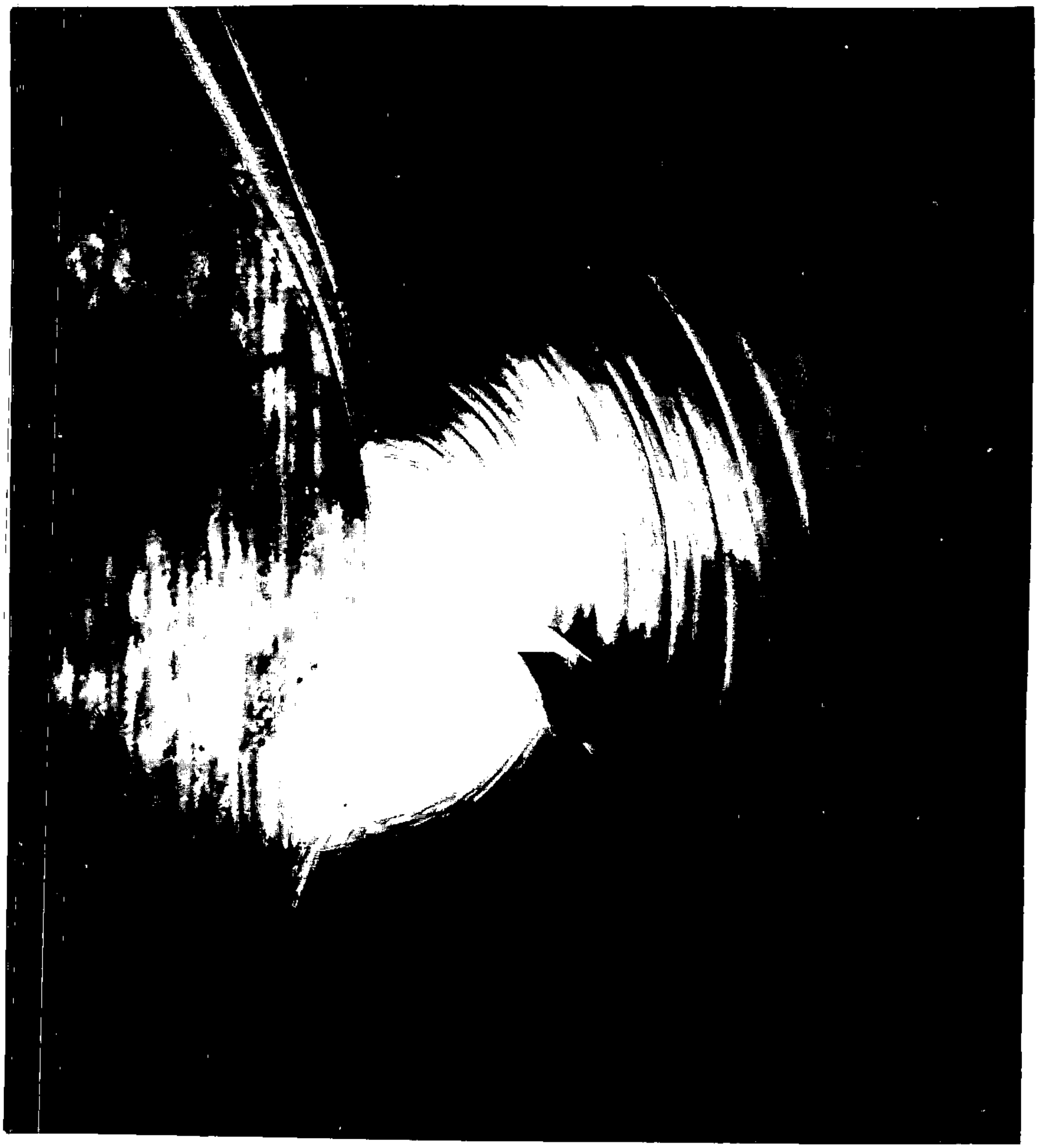


FIGURE 21

TEMPERATURE DEPRESSION PATTERN

FLIGHT 69

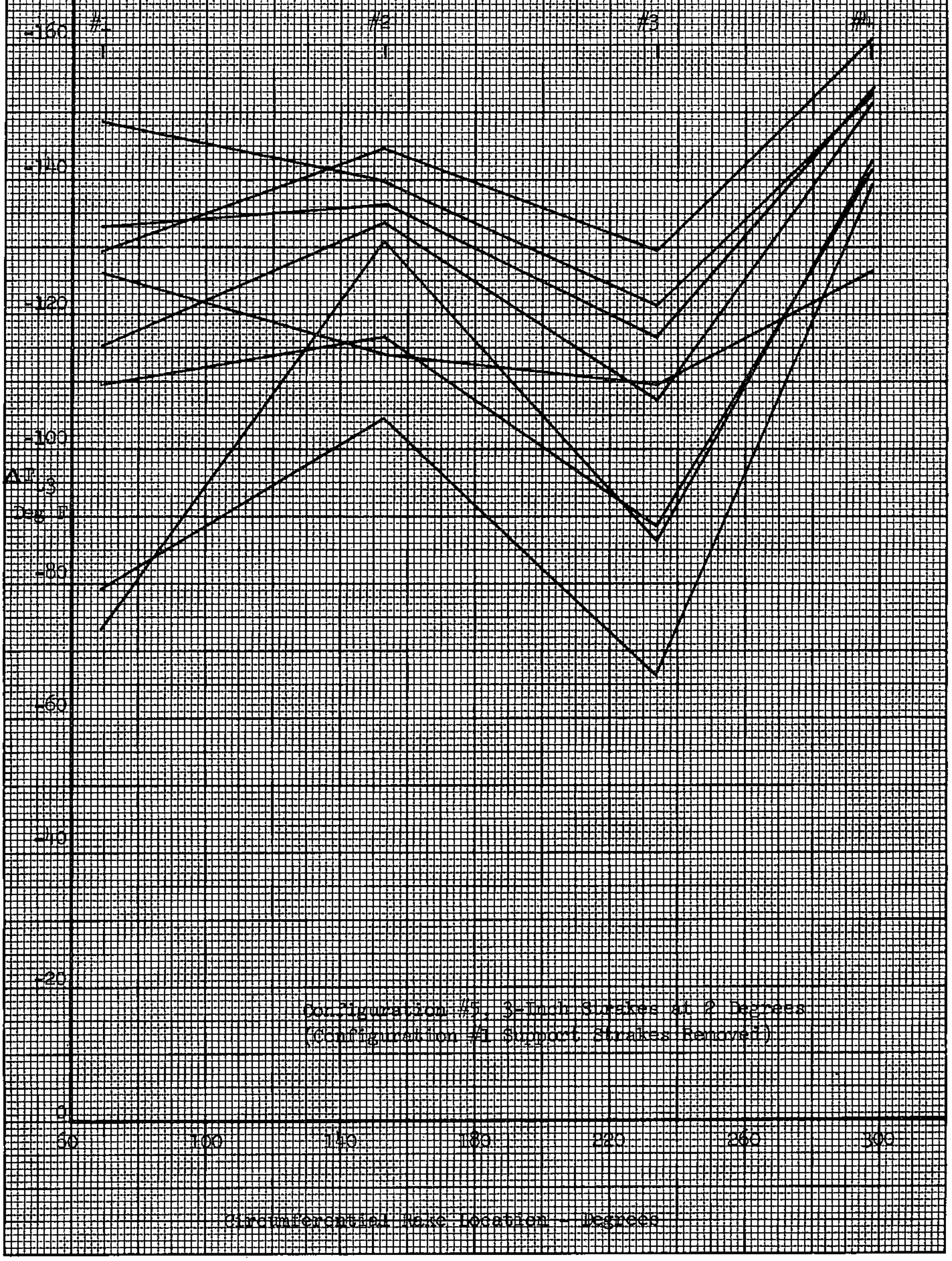
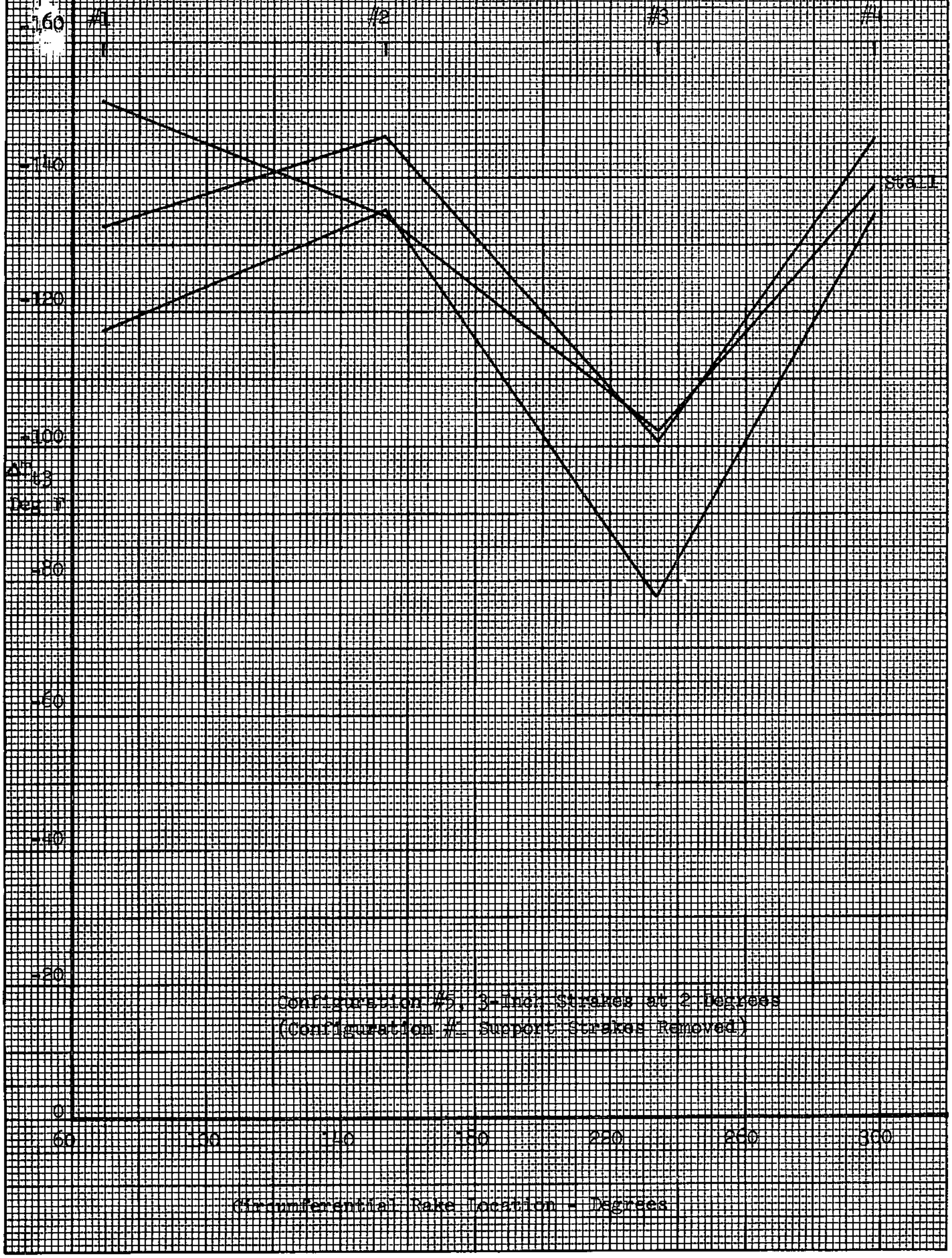


FIGURE 22

TEMPERATURE DEPRESSION PATTERN

FLIGHT 72

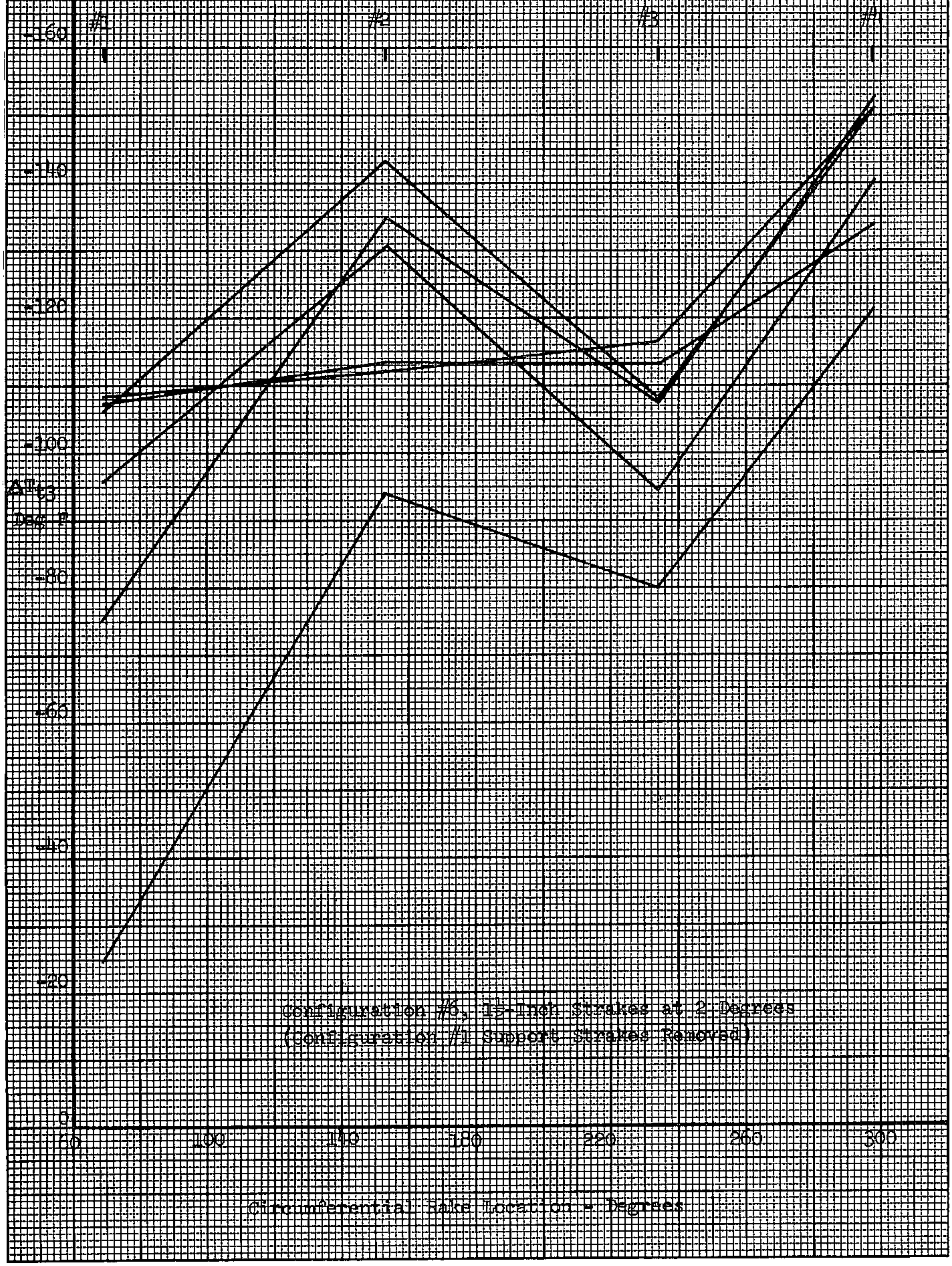


Configuration #5, 3-Inch Strakes at 2 Degrees
(Configuration #1 Support Strakes Removed)

FIGURE 23

TEMPERATURE DEPRESSION PATTERN

FLIGHT 75

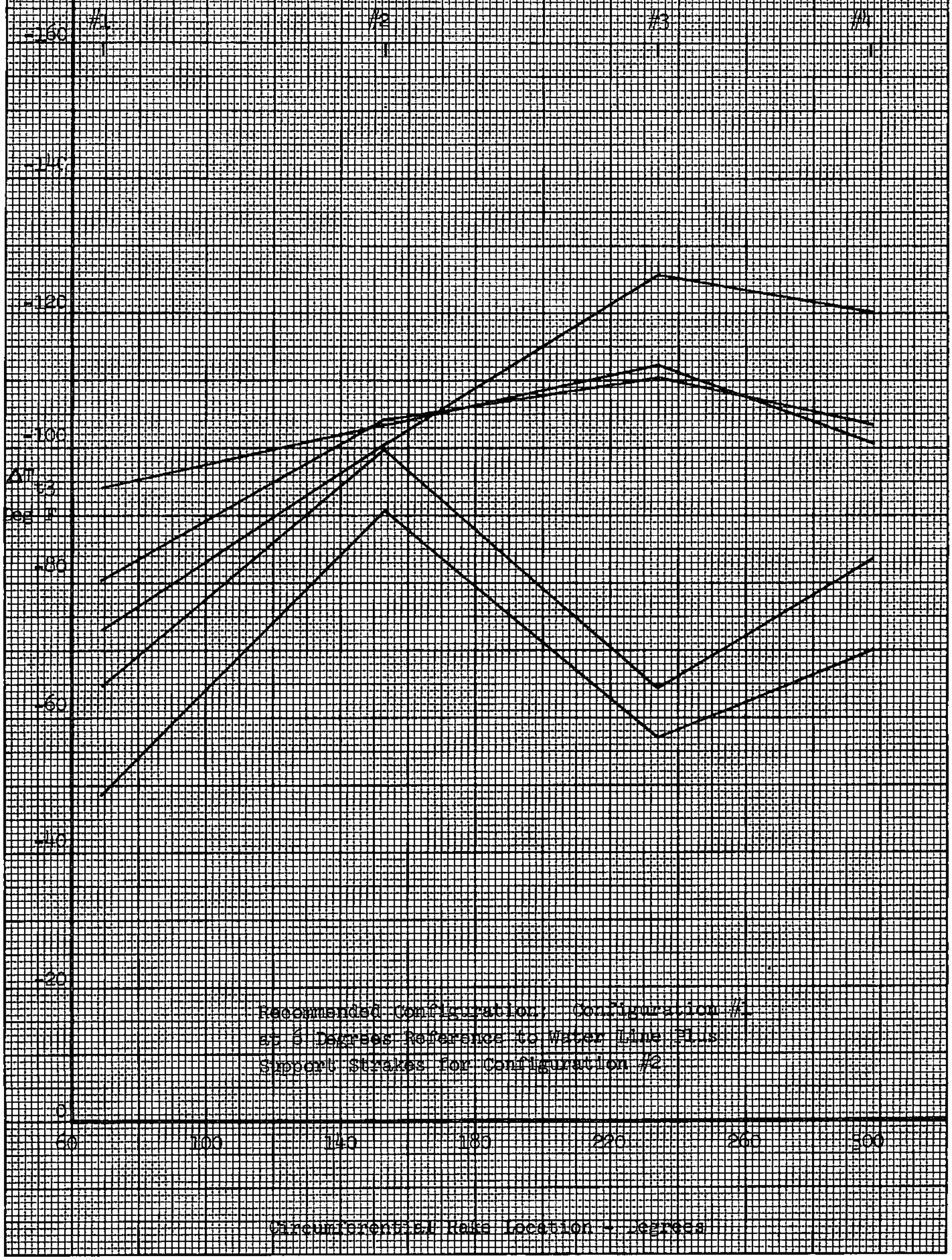


Configuration #6, $1\frac{1}{2}$-Inch Strakes at 2 Degrees
(Configuration #1 Support Strakes Removed)

FIGURE 24

TEMPERATURE DEPRESSION PATTERN

FLIGHT 63

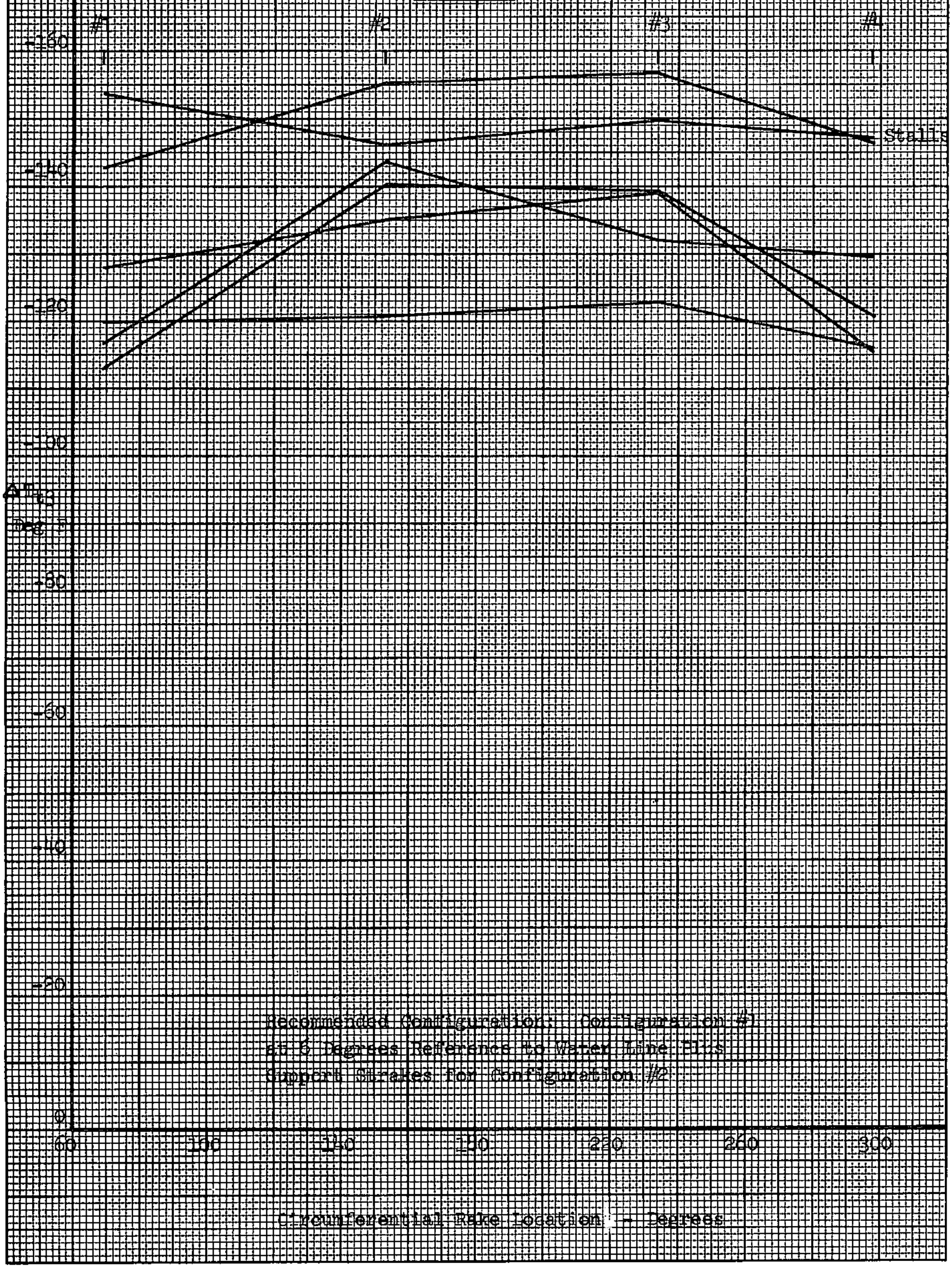


Recommended Configuration; Configuration #1 at 6 Degrees Reference to Water Line Plus Support Strakes for Configuration #2

FIGURE 25

TEMPERATURE DEPRESSION PATTERN

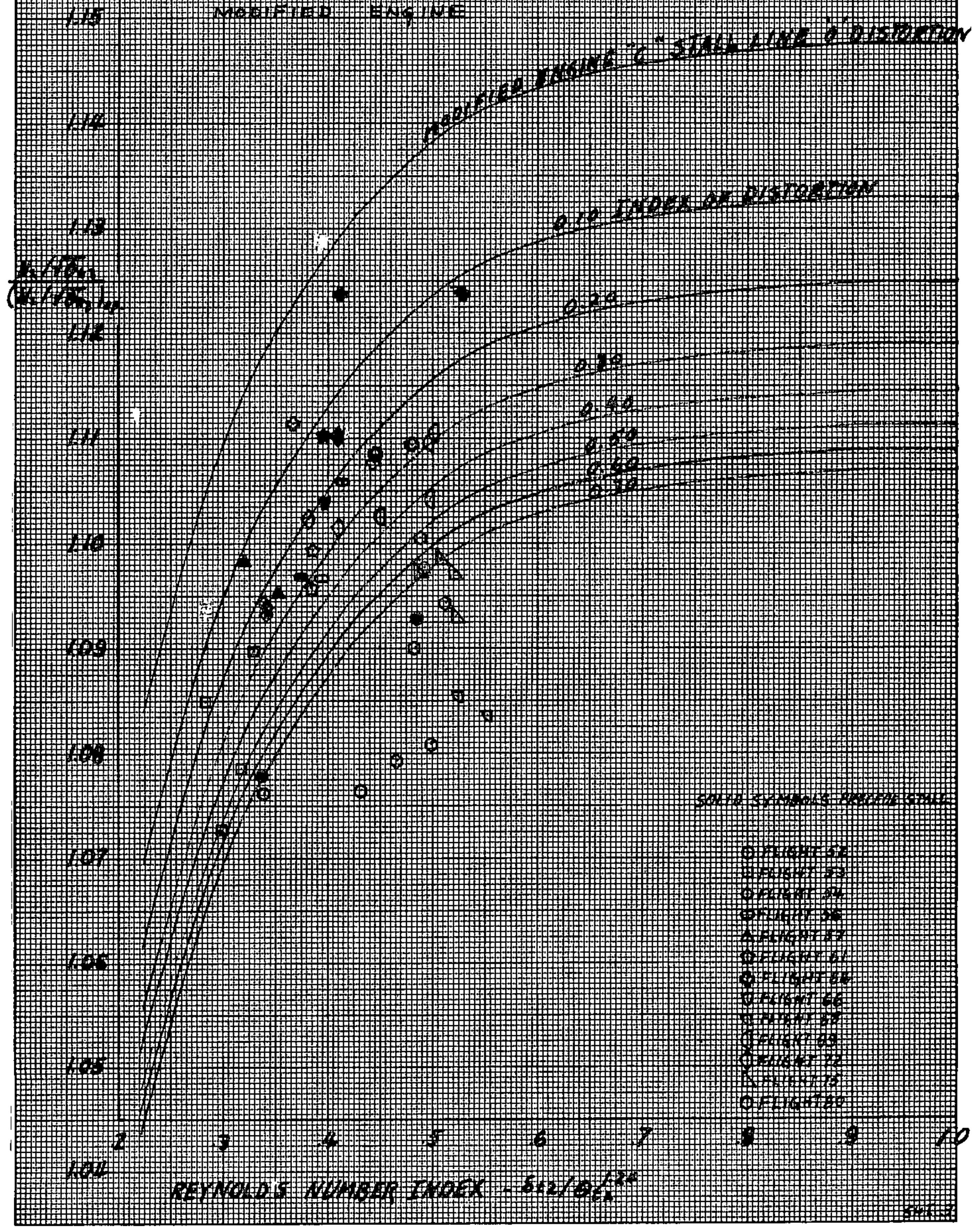
FLIGHT 64



Recommended Configuration: Configuration #1 at 6 Degrees Reference to Water Line Plus Support Strakes for Configuration #2

FIGURE 26

EFFECT OF ENTHALPIC DISTORTION ON N_2 STALL MARGIN



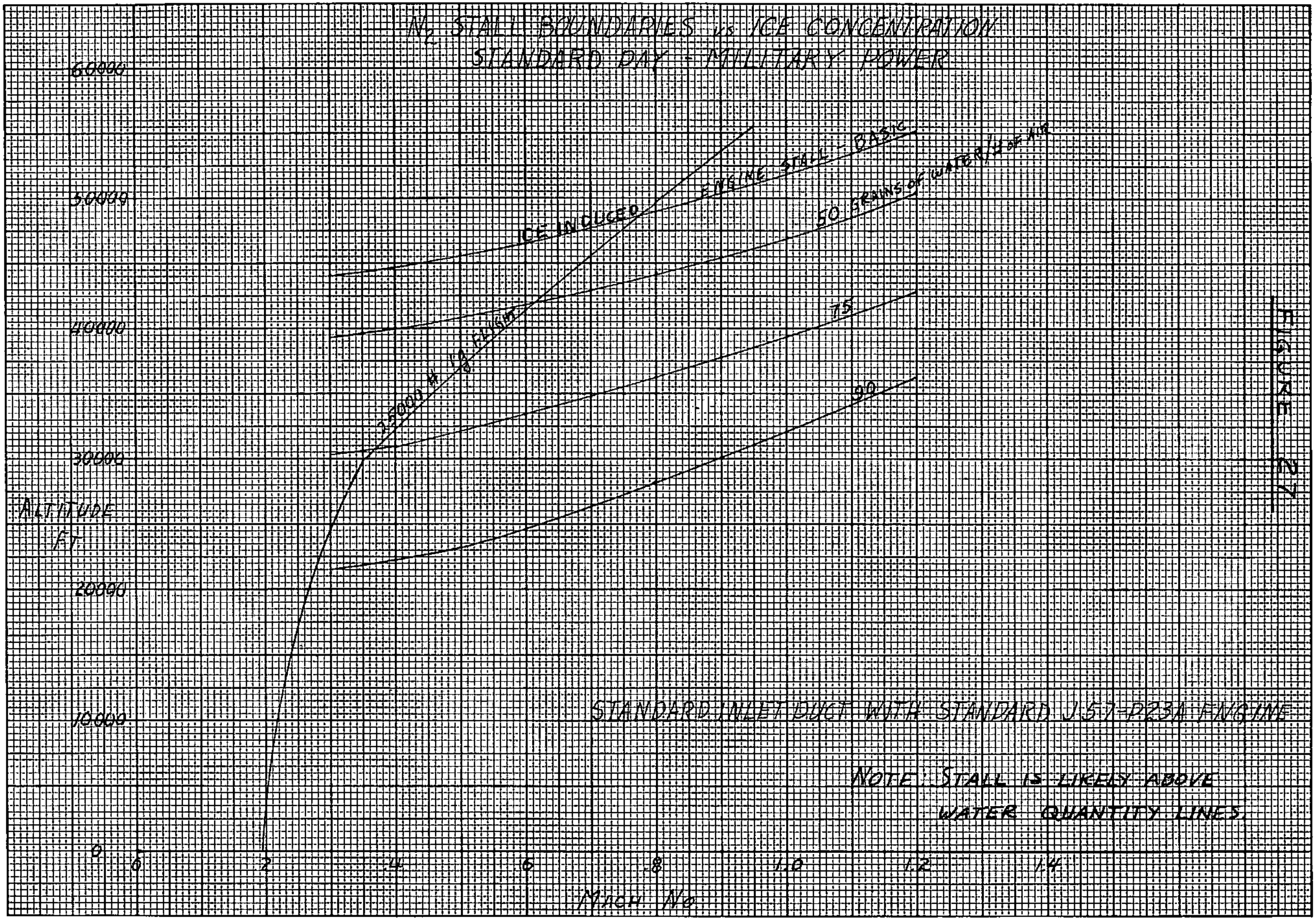


FIGURE 27

N₂ STALL BOUNDARIES vs ICE CONCENTRATION
STANDARD DAY - MILITARY POWER

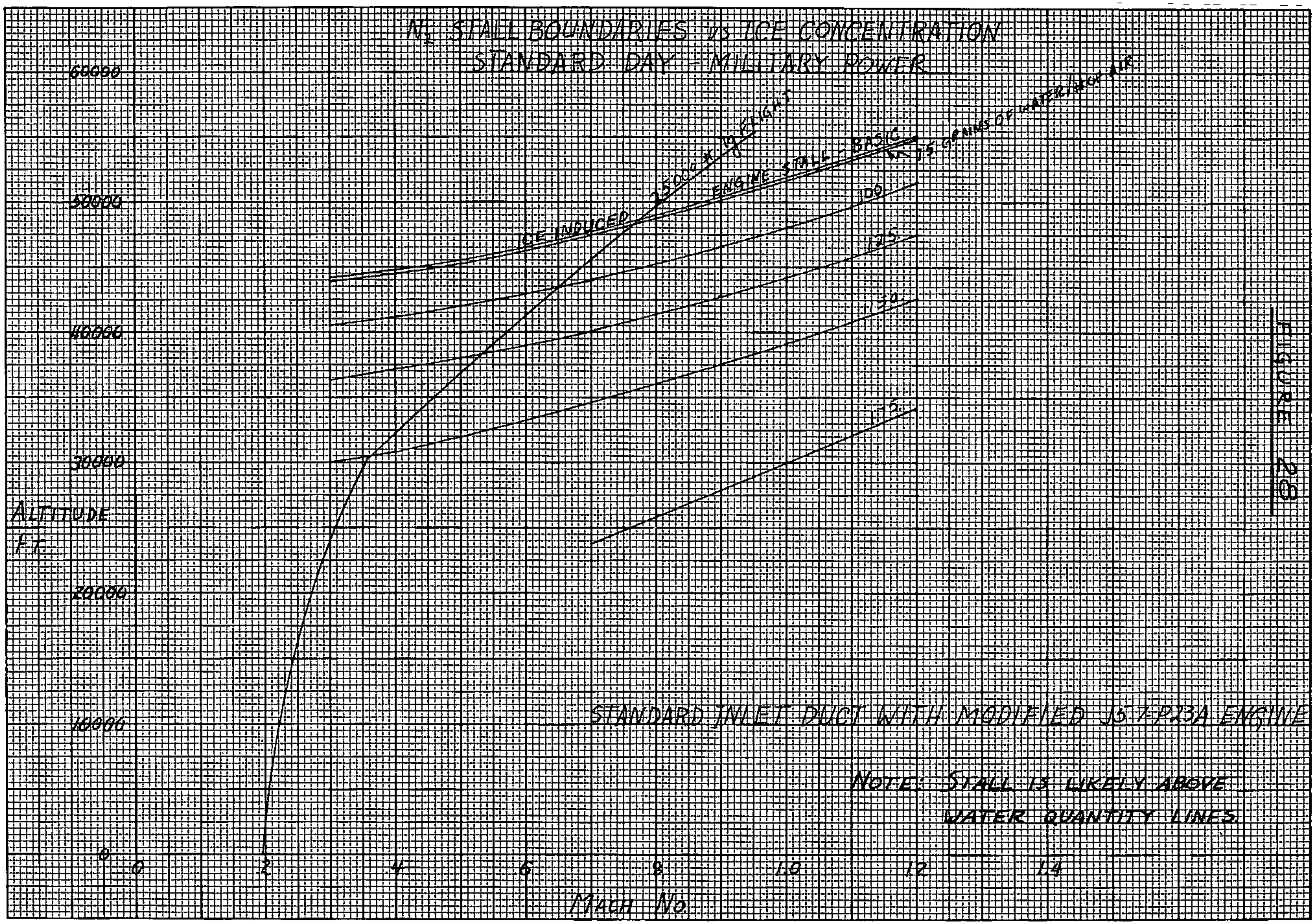


FIGURE 28

N$_2$ STALL BOUNDARIES vs ICE CONCENTRATION
STANDARD DAY - MILITARY POWER

STANDARD INLET DUCT WITH MODIFIED J57-P23A ENGINE

NOTE: STALL IS LIKELY ABOVE WATER QUANTITY LINES.

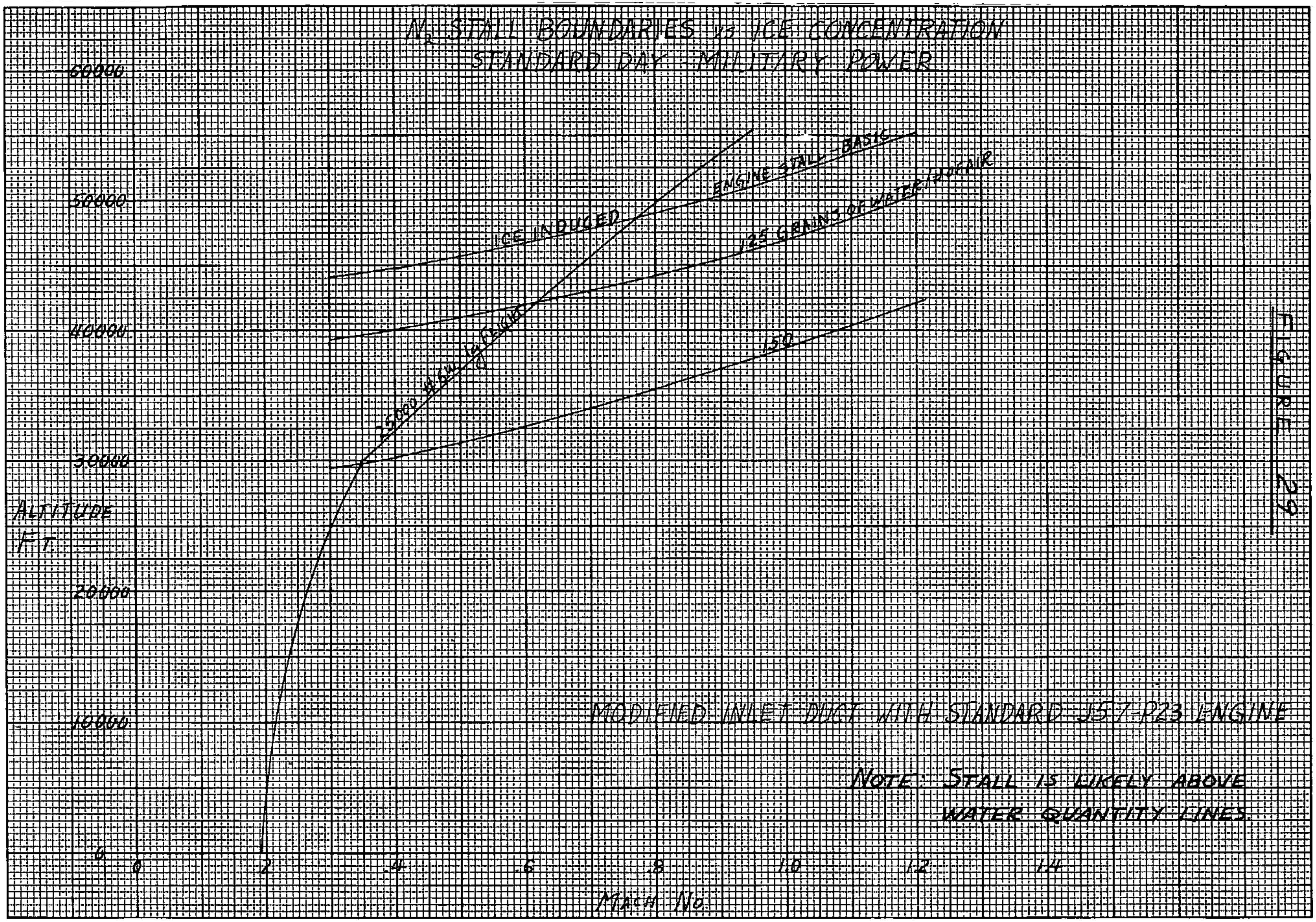


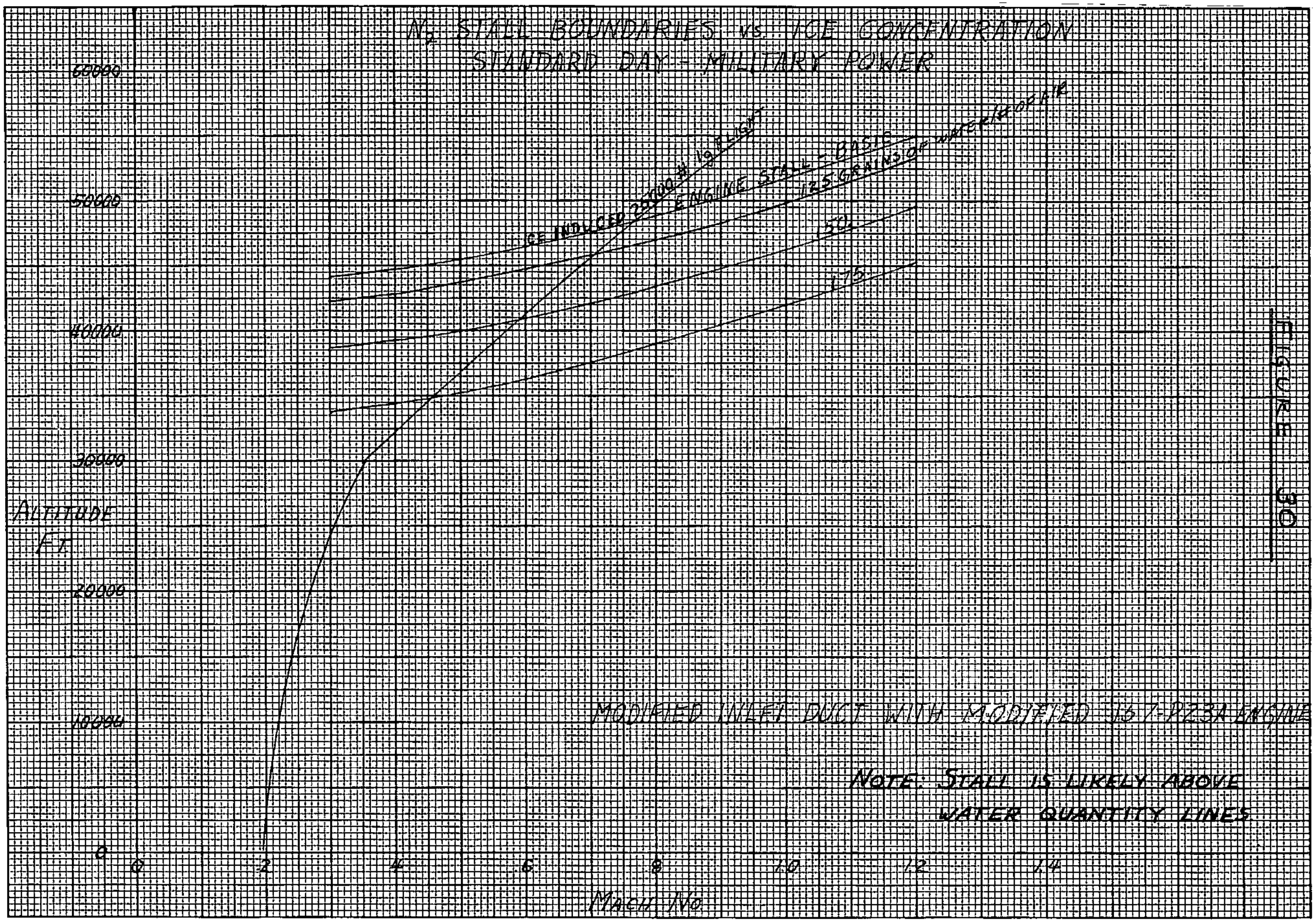
FIGURE 29

N_2 STALL BOUNDARIES vs ICE CONCENTRATION
STANDARD DAY - MILITARY POWER

MODIFIED INLET DUCT WITH STANDARD J57-P23 ENGINE

NOTE: STALL IS LIKELY ABOVE WATER QUANTITY LINES.

FIGURE 30



N₂ STALL BOUNDARIES vs ICE CONCENTRATION
STANDARD DAY - MILITARY POWER

MODIFIED INLET DUCT WITH MODIFIED J57-P23A ENGINE

NOTE: STALL IS LIKELY ABOVE WATER QUANTITY LINES

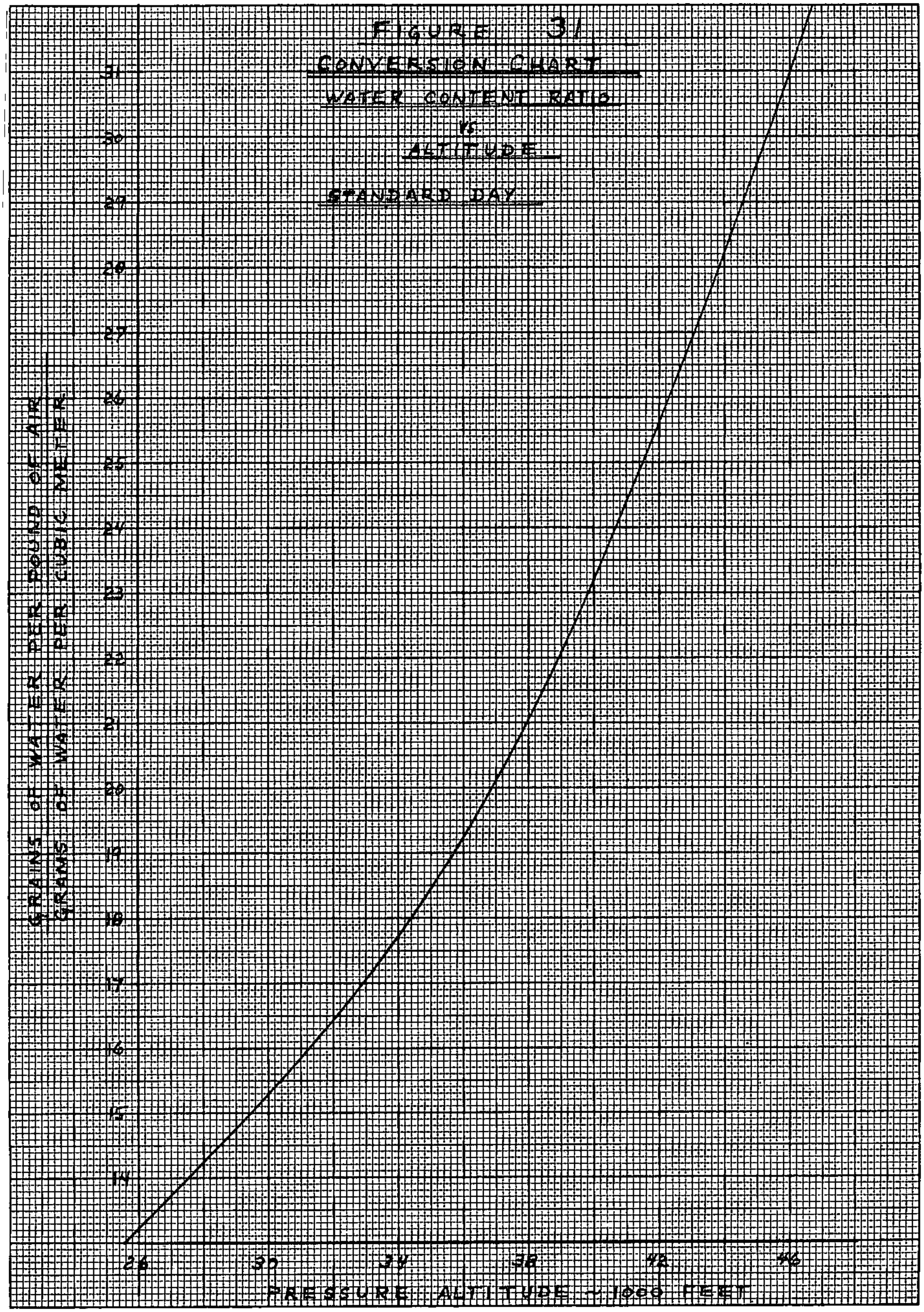


FIGURE 31

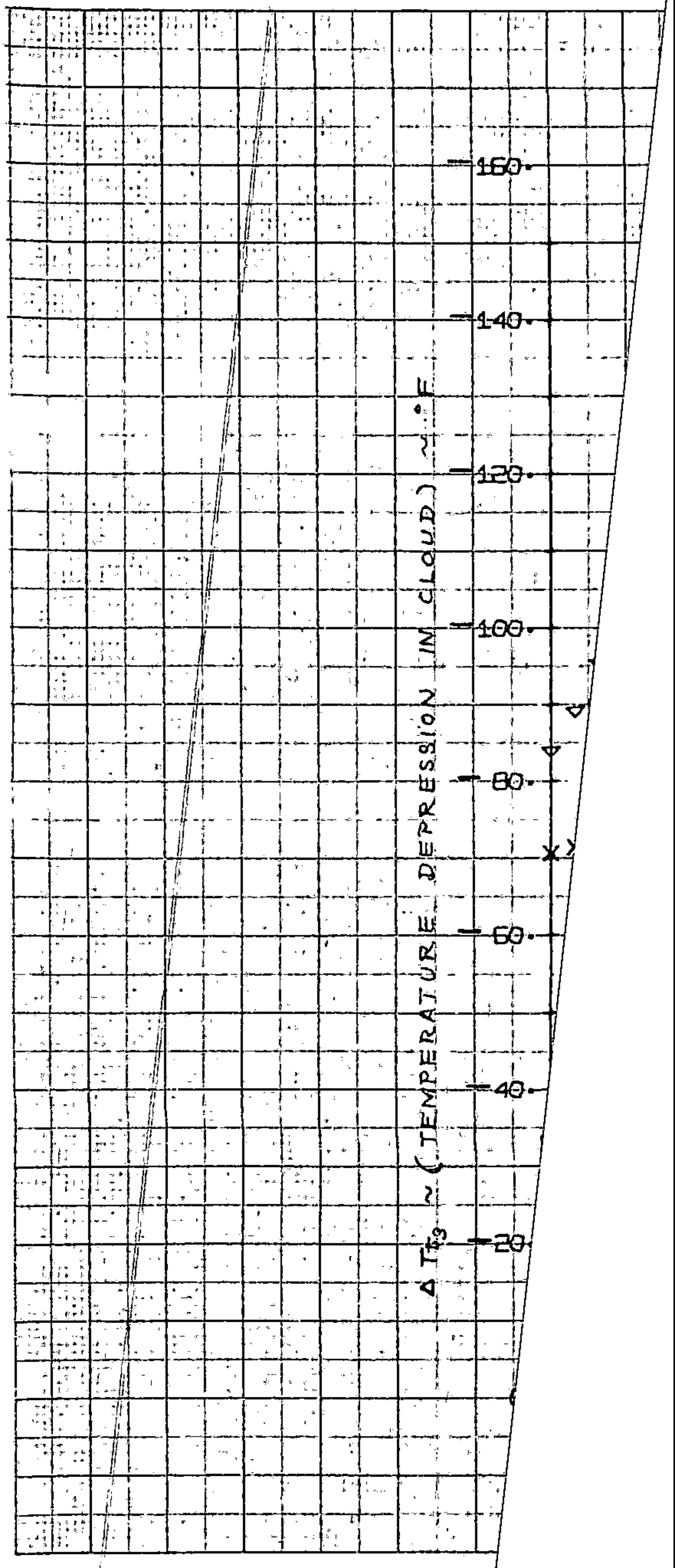
CONVERSION CHART

WATER CONTENT RATIO

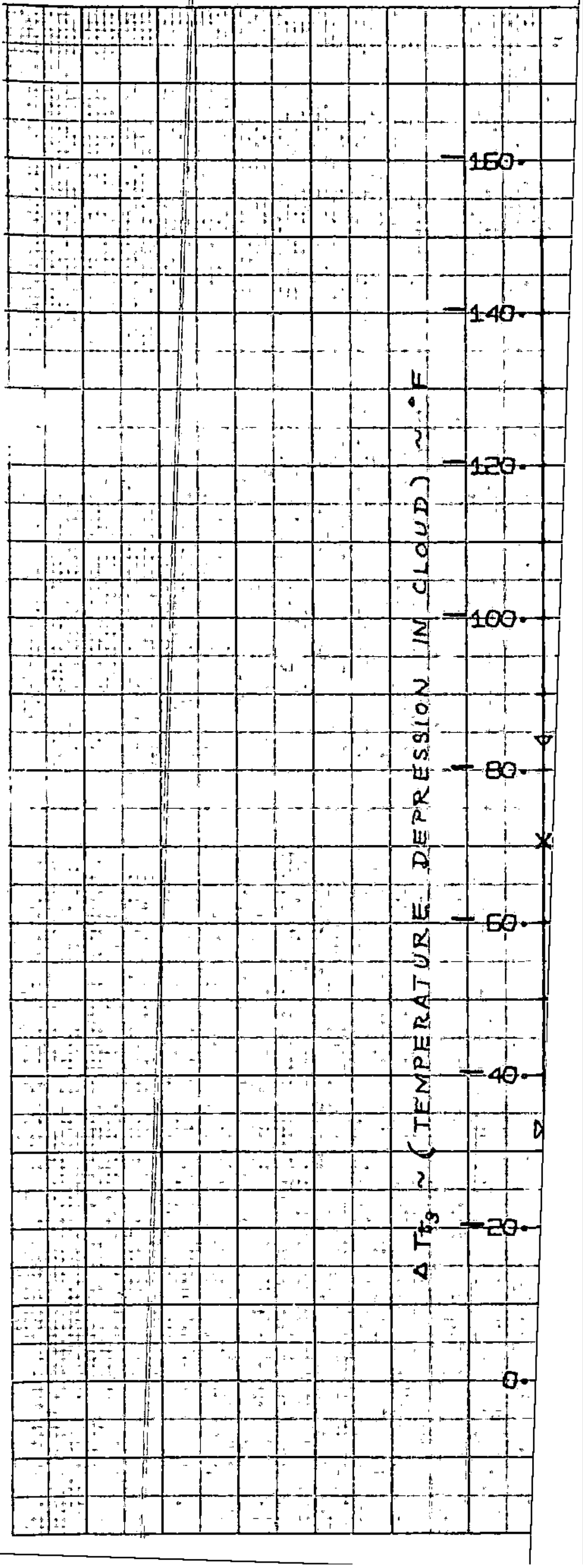
VS

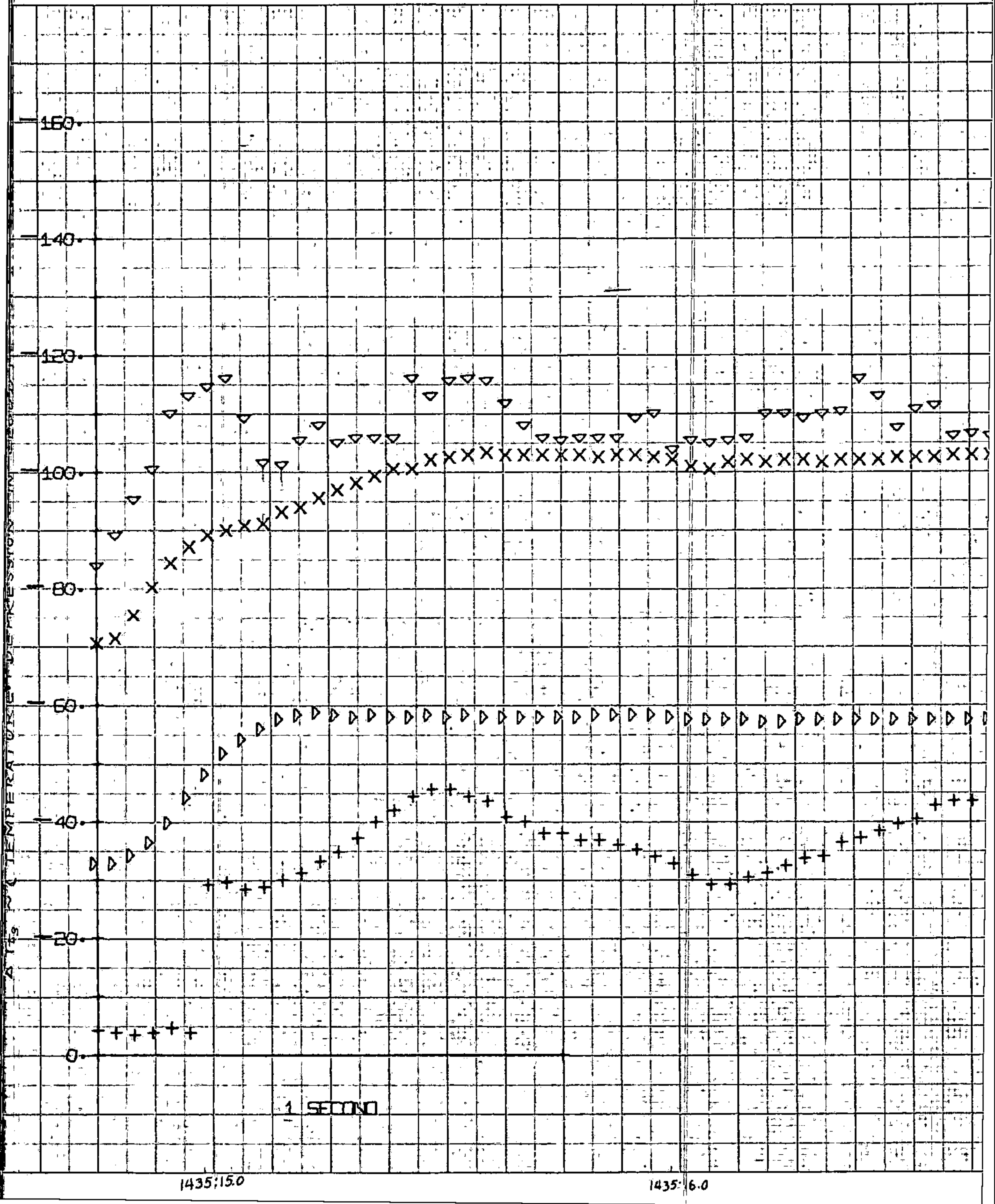
ALTITUDE

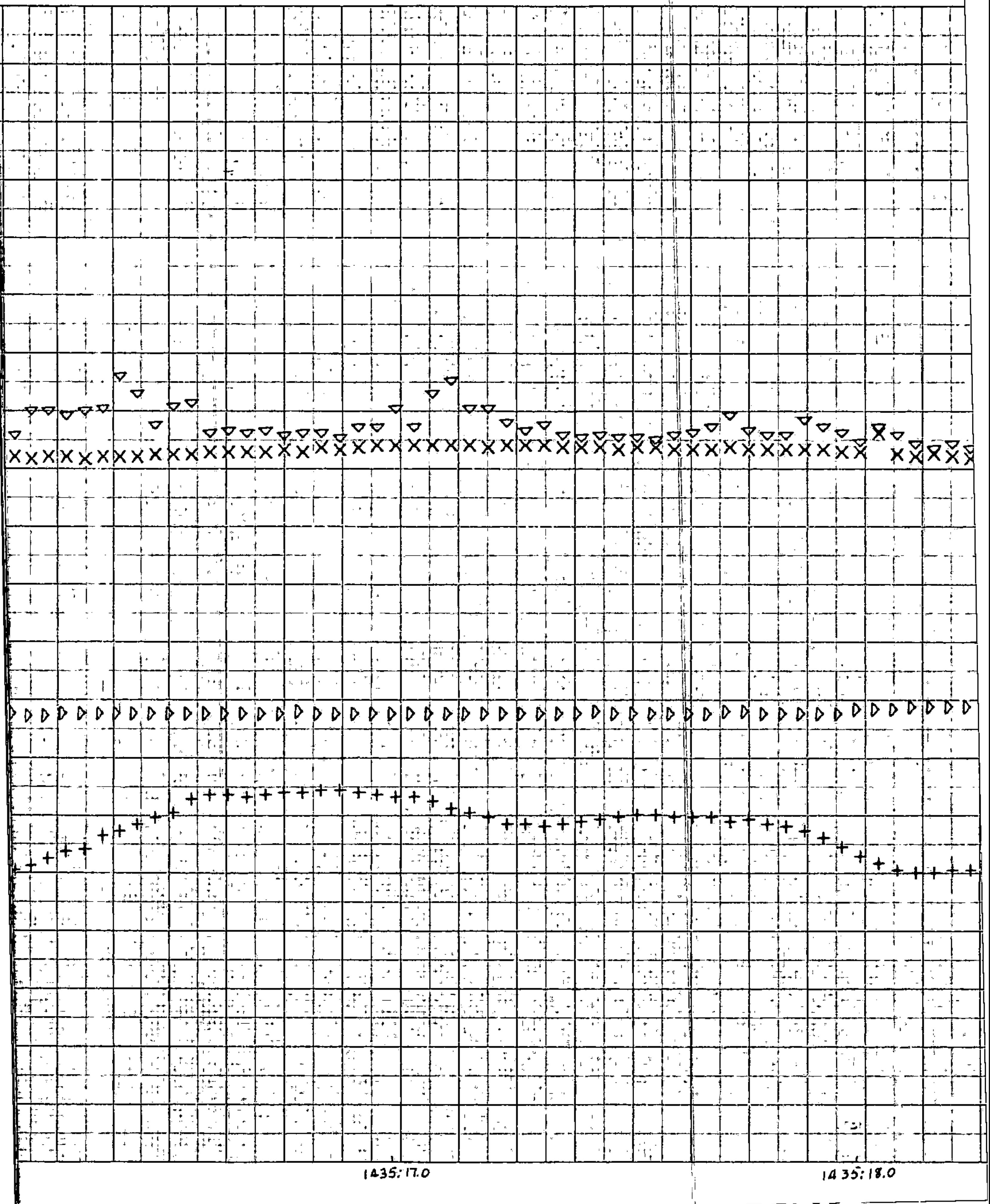
STANDARD DAY



ΔT$_{S3}$ ~ (TEMPERATURE DEPRESSION IN CLOUD) ~ °F
160
140
120
100
80
60
40
20


ΔT$_{Tg}$ ~ (TEMPERATURE DEPRESSION IN CLOUD) ~ °F
160
140
120
100
80
60
40
20
0


ΔT49 N2 TEMPERATURE DEPRESSION IN DEGREES
160
140
120
100
80
60
40
20
0
1 SECOND
1435:15.0
1435:16.0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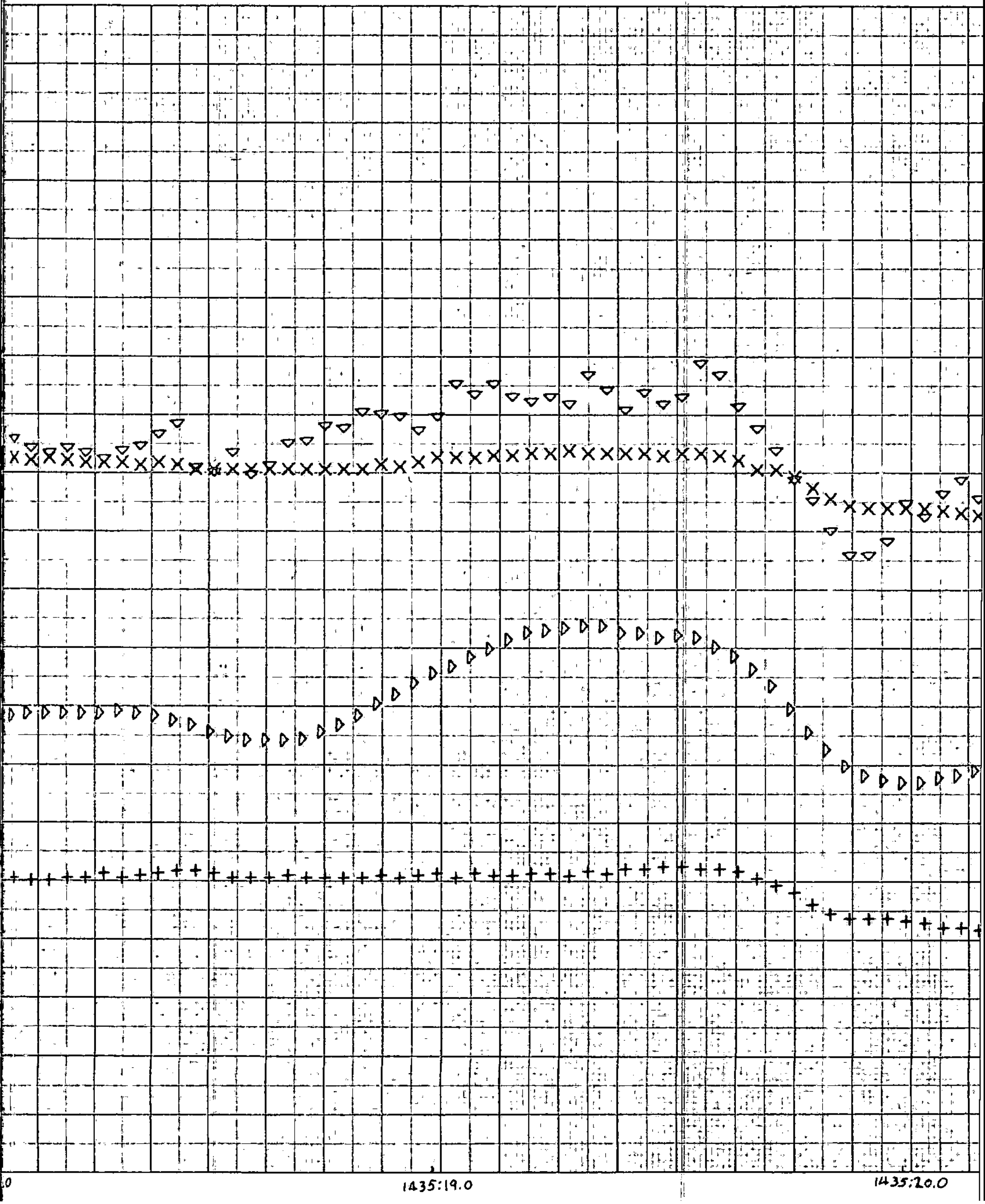
1435:18.0


0
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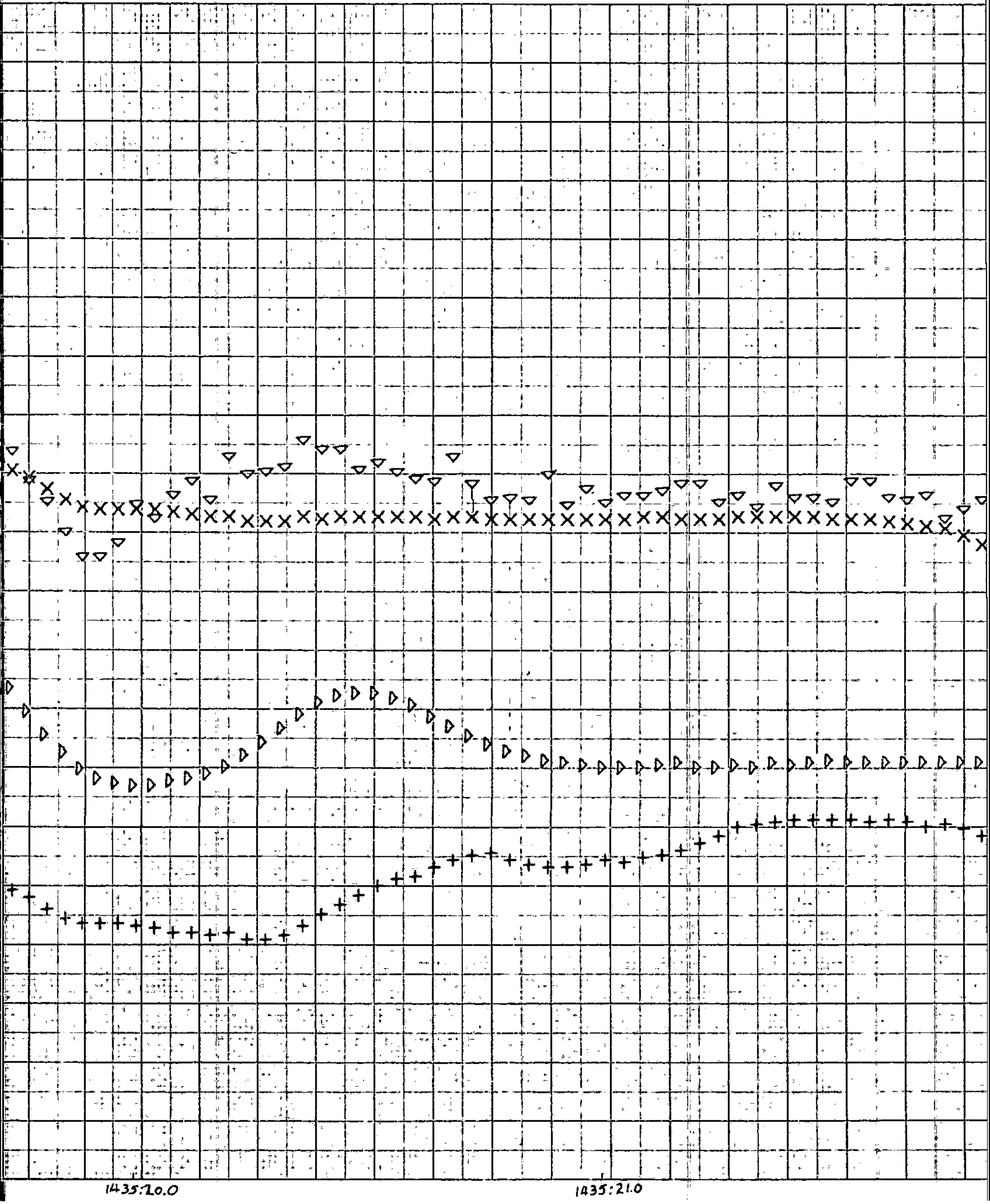

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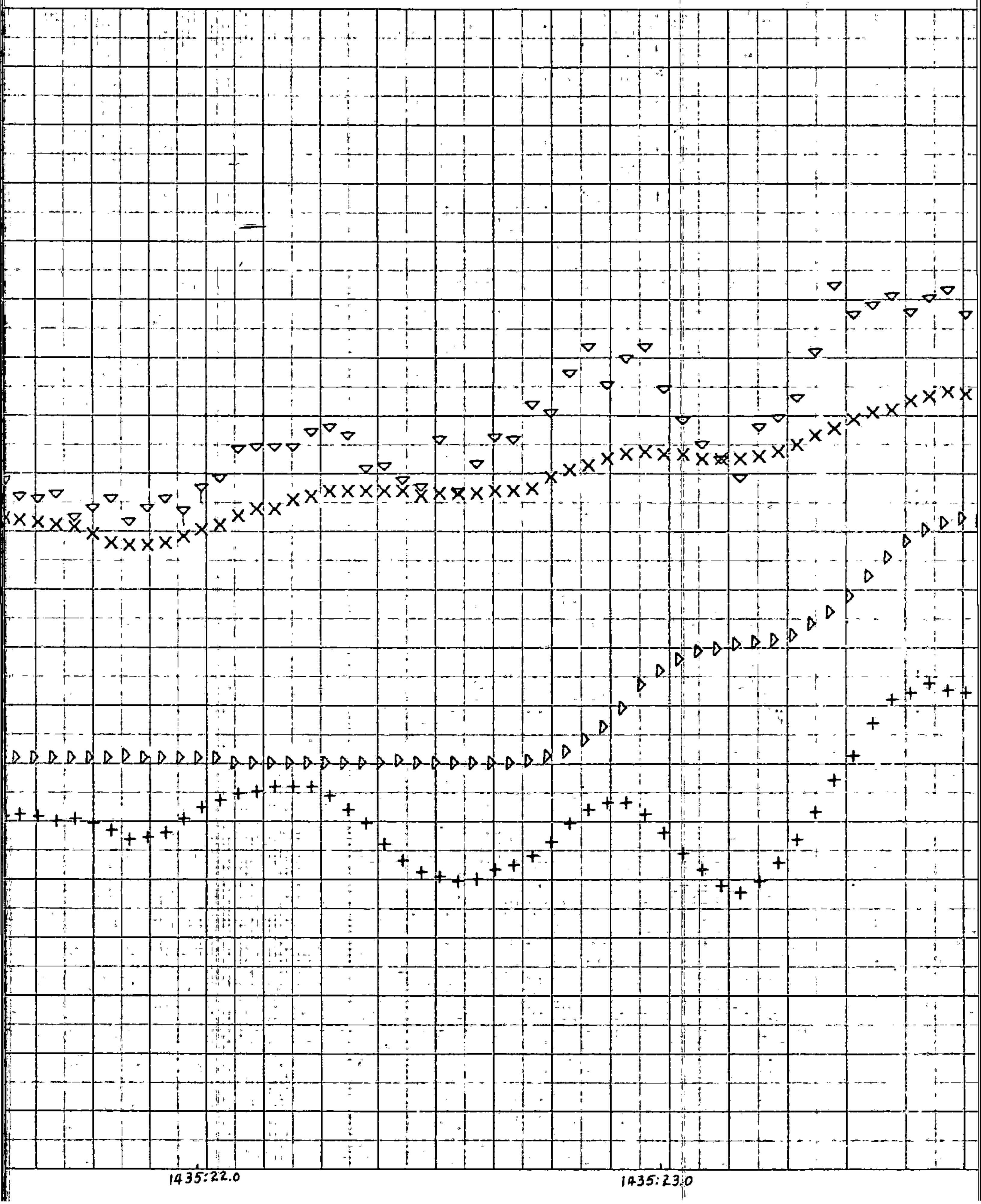
1435:21.0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1435:23.0

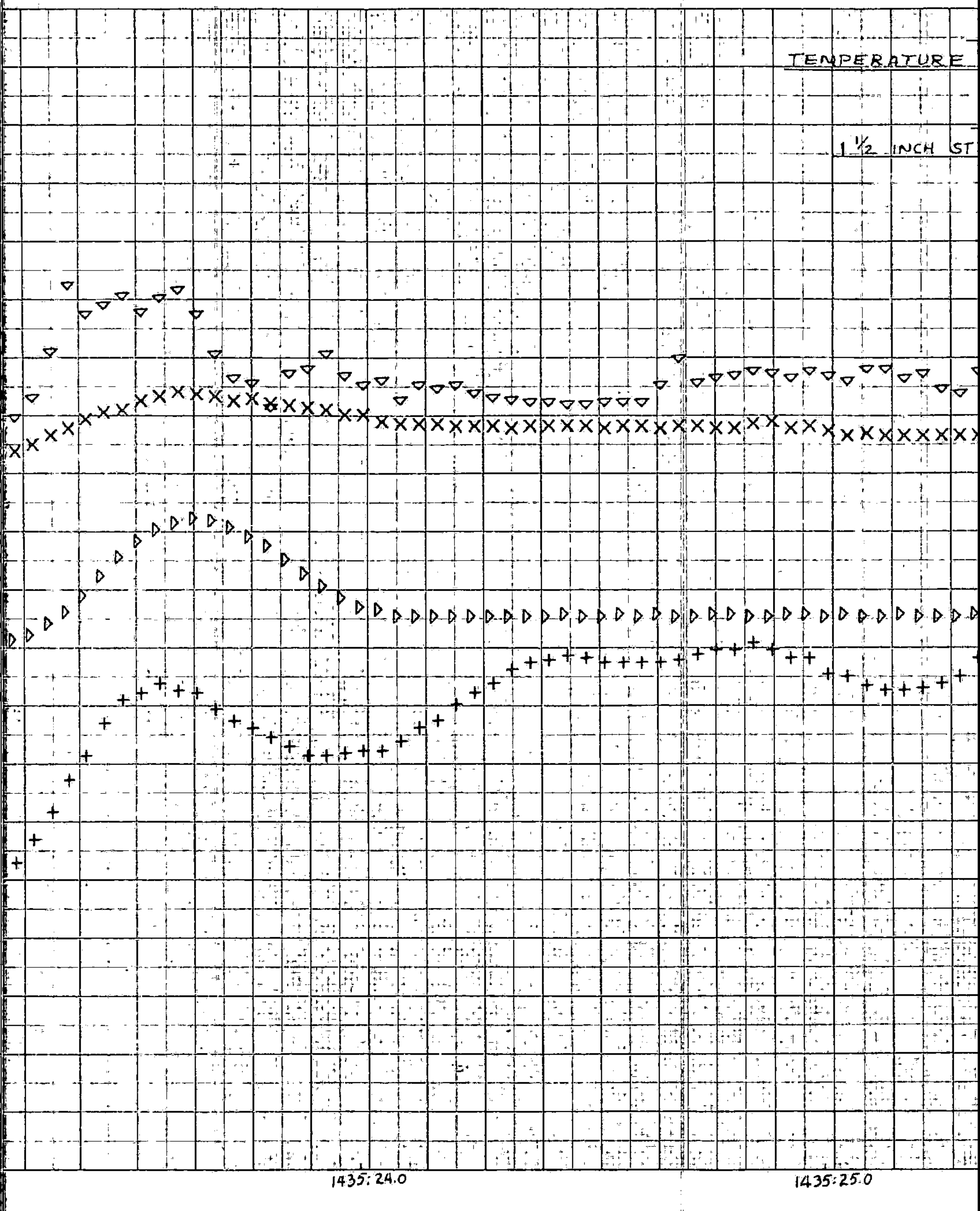
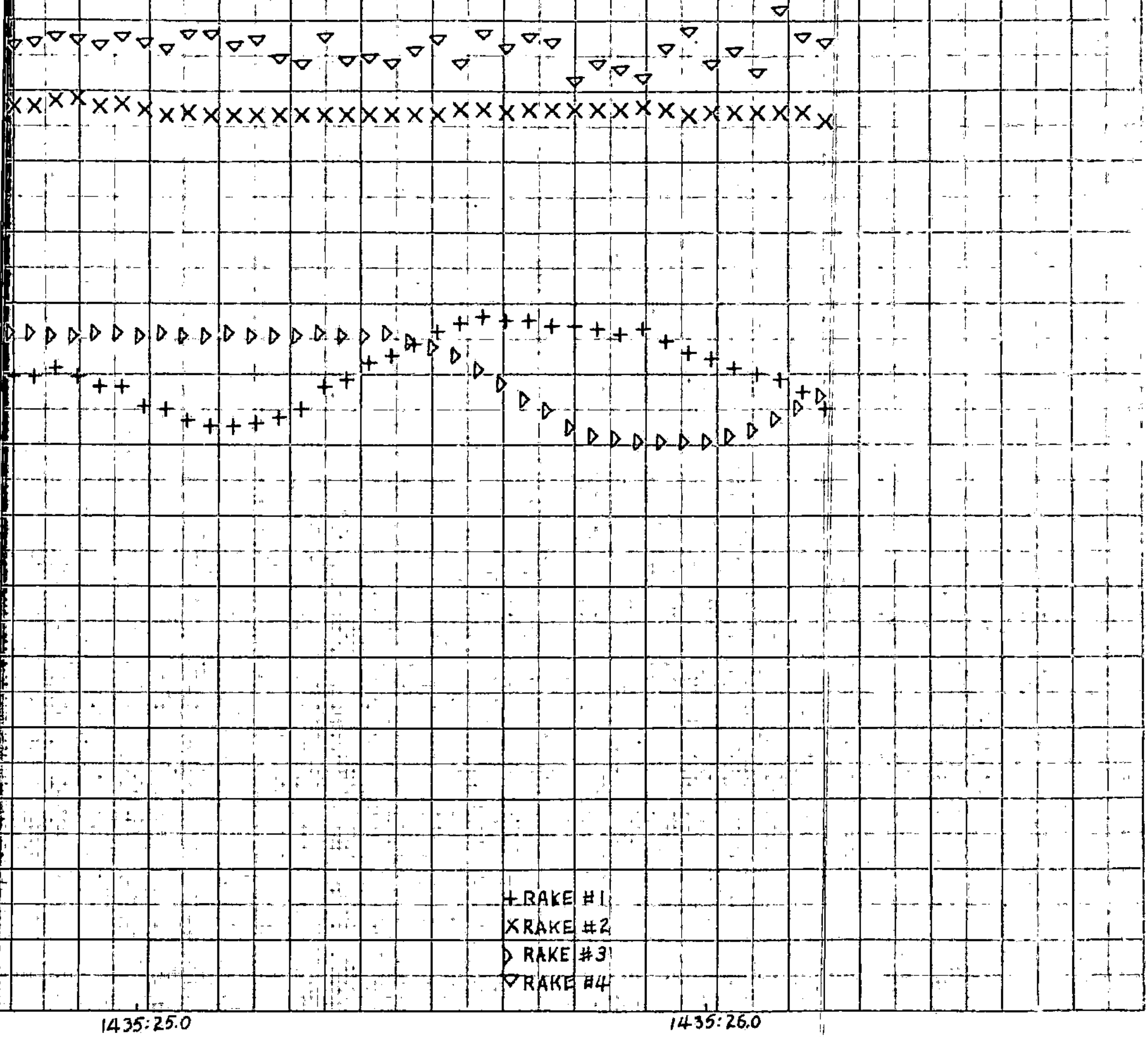

TEMPERATURE
1 1/2 INCH ST
1435:24.0
1435:25.0

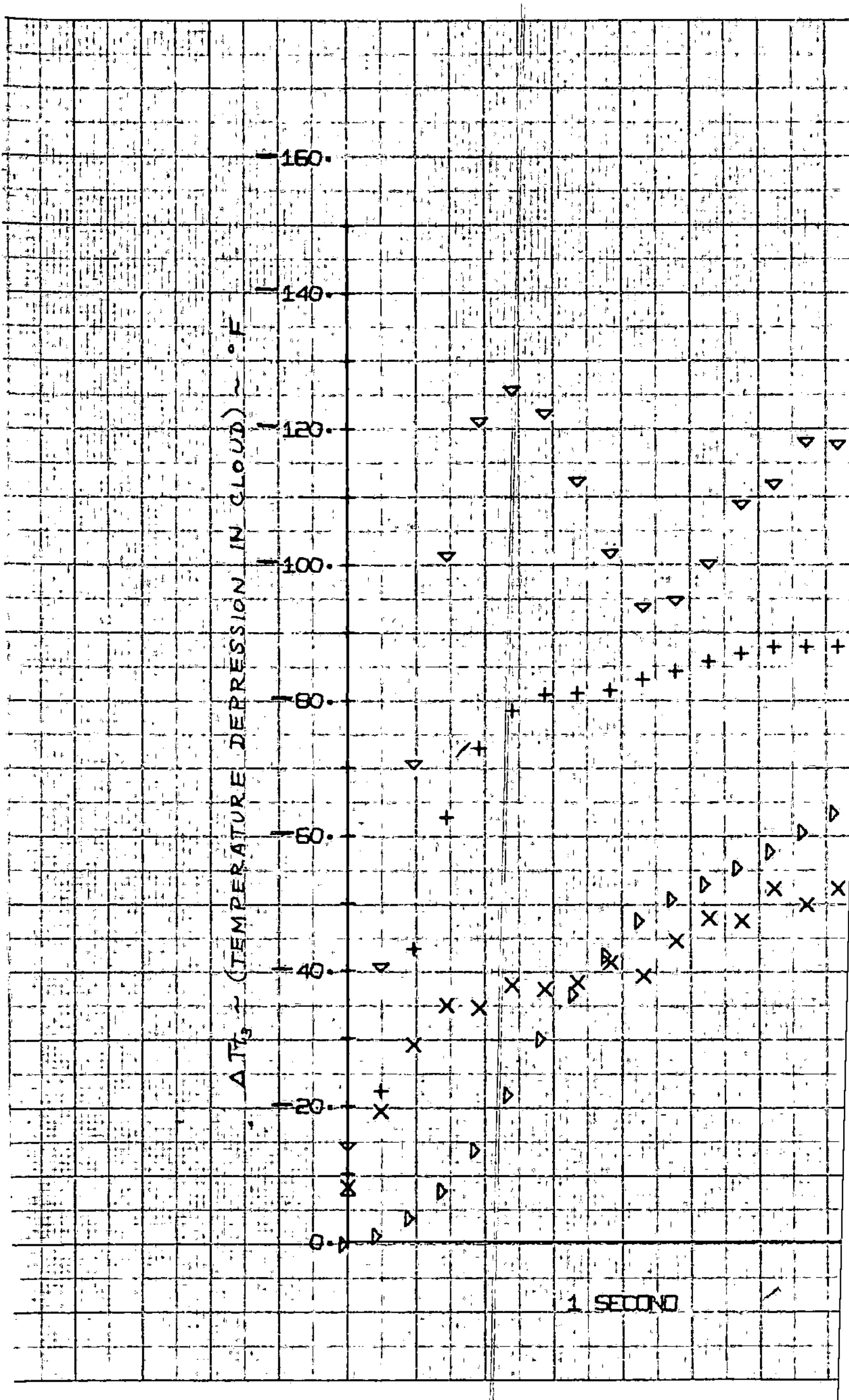
FIGURE 32

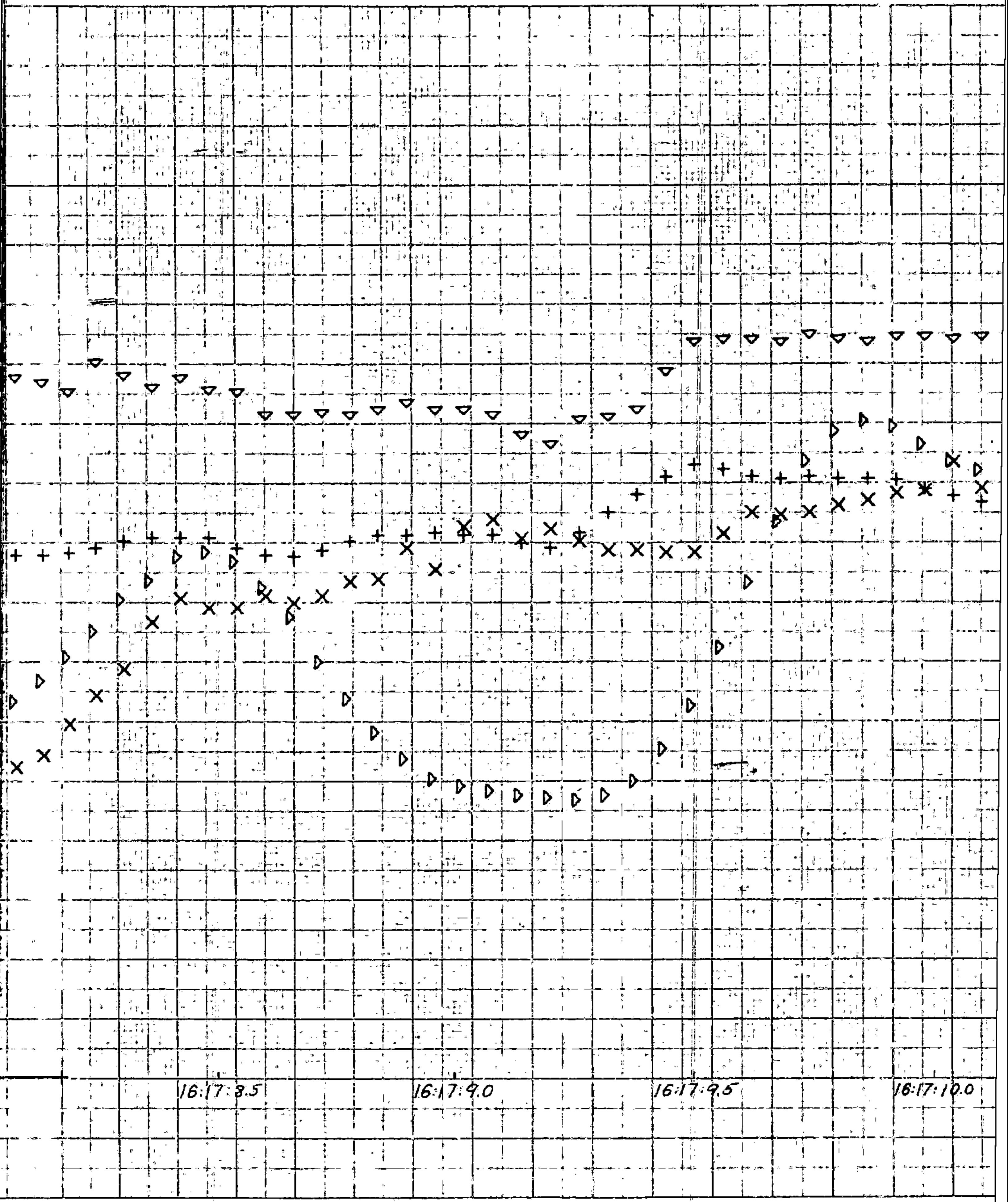
TEMPERATURE DEPRESSION (T_{T_3}) DURING CLOUD ENTRY

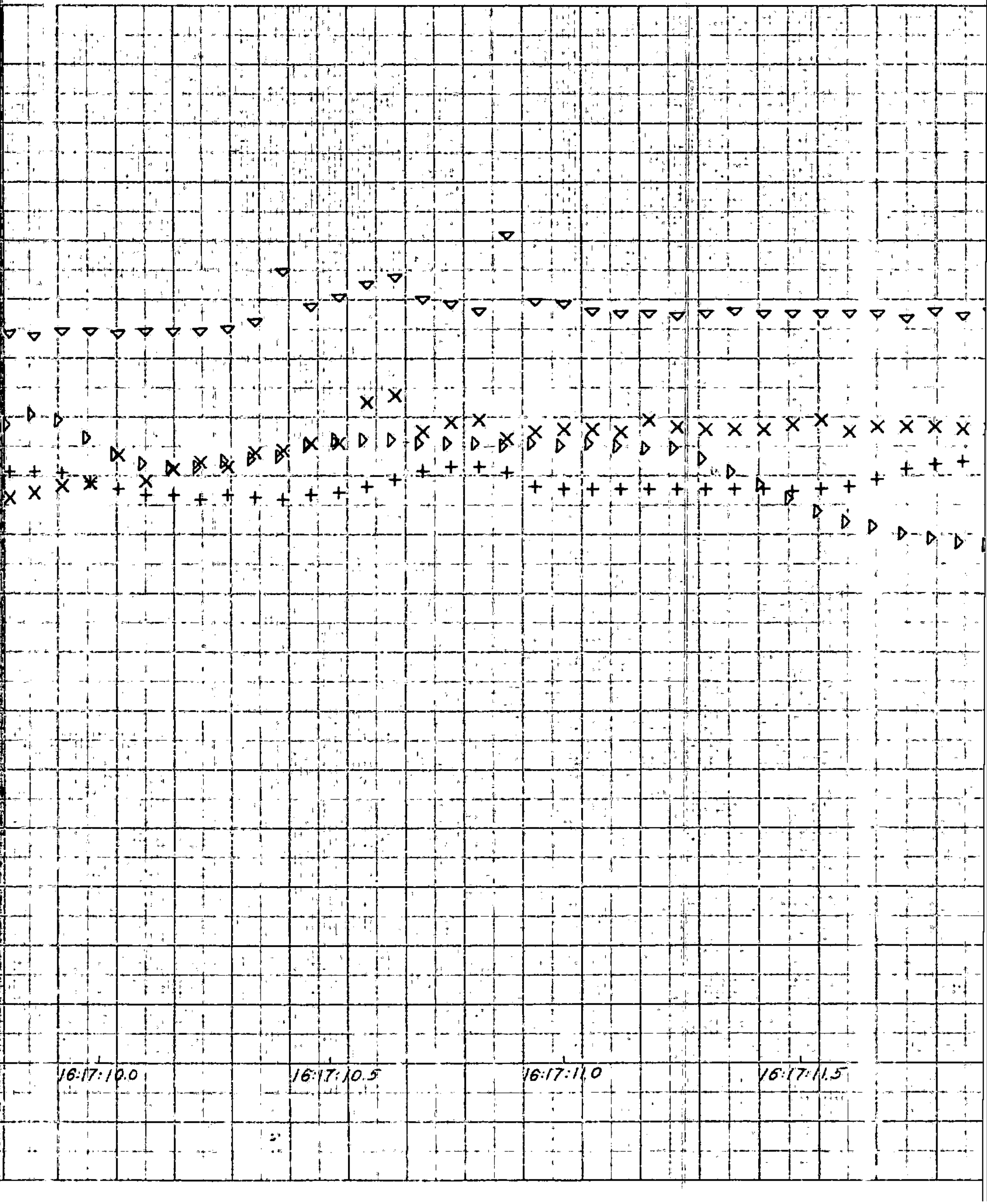
FLIGHT 75

1½ INCH STRAKES AT 2 DEGREES (REF. DUCT FLOOR)




ΔT$_{T_3}$ ~ (TEMPERATURE DEPRESSION IN CLOUD) ~ °F
160
140
120
100
80
60
40
20
0
1 SECOND


16:17:8.5
16:17:9.0
16:17:9.5
16:17:10.0


16:17:10.0
16:17:10.5
16:17:11.0
16:17:11.5

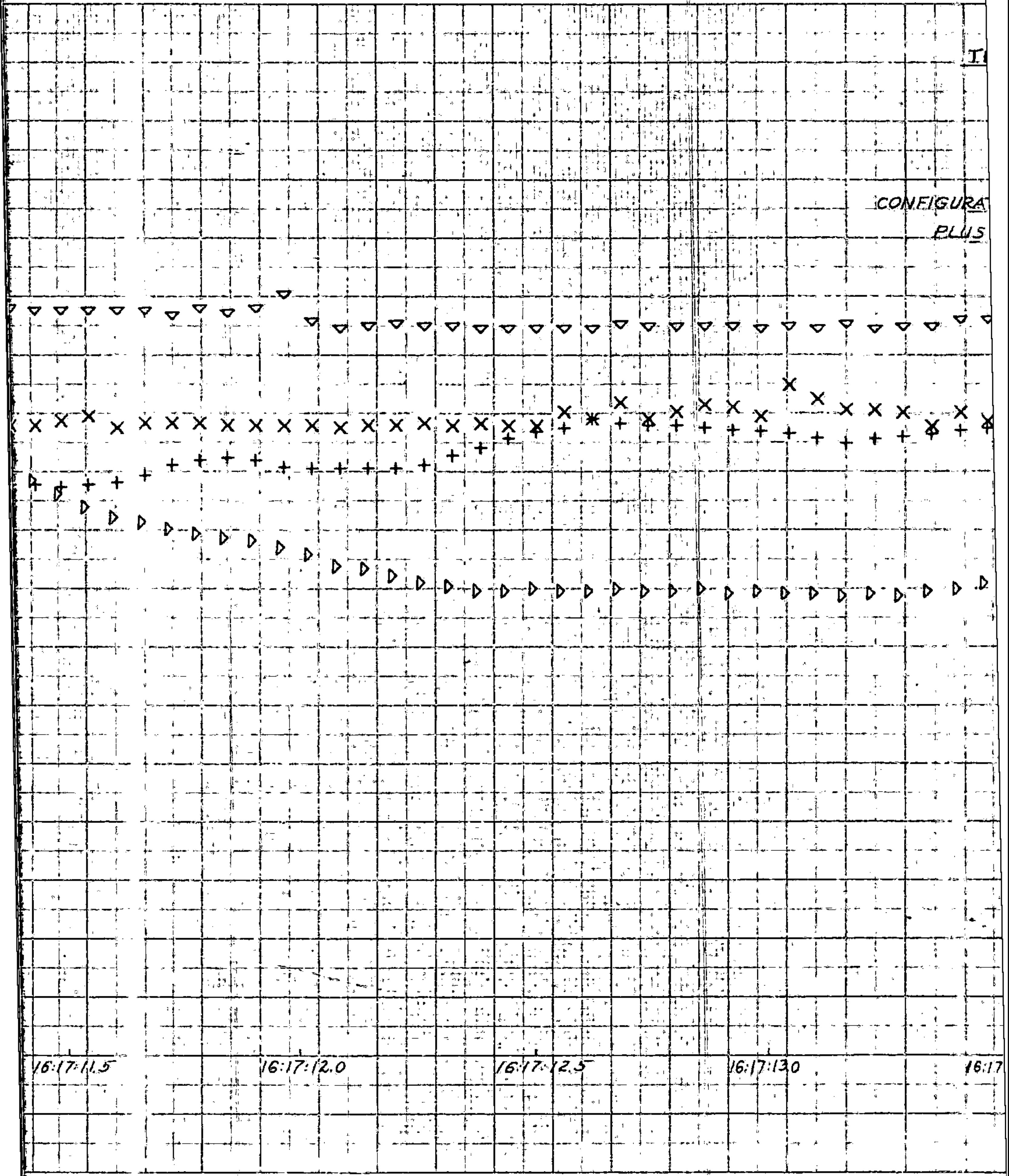

CONFIGURA
PLUS
16:17:11.5
16:17:12.0
16:17:12.5
16:17:13.0
16:17

FIGURE 33

TEMPERATURE DEPRESSION (T_{t_3}) DURING CLOUD ENTRY

FLIGHT 56

CONFIGURATION #2 AT 0 DEGREES (REF. DUCT FLOOR)

PLUS SUPPORT STRAKES FOR CONFIGURATION #1

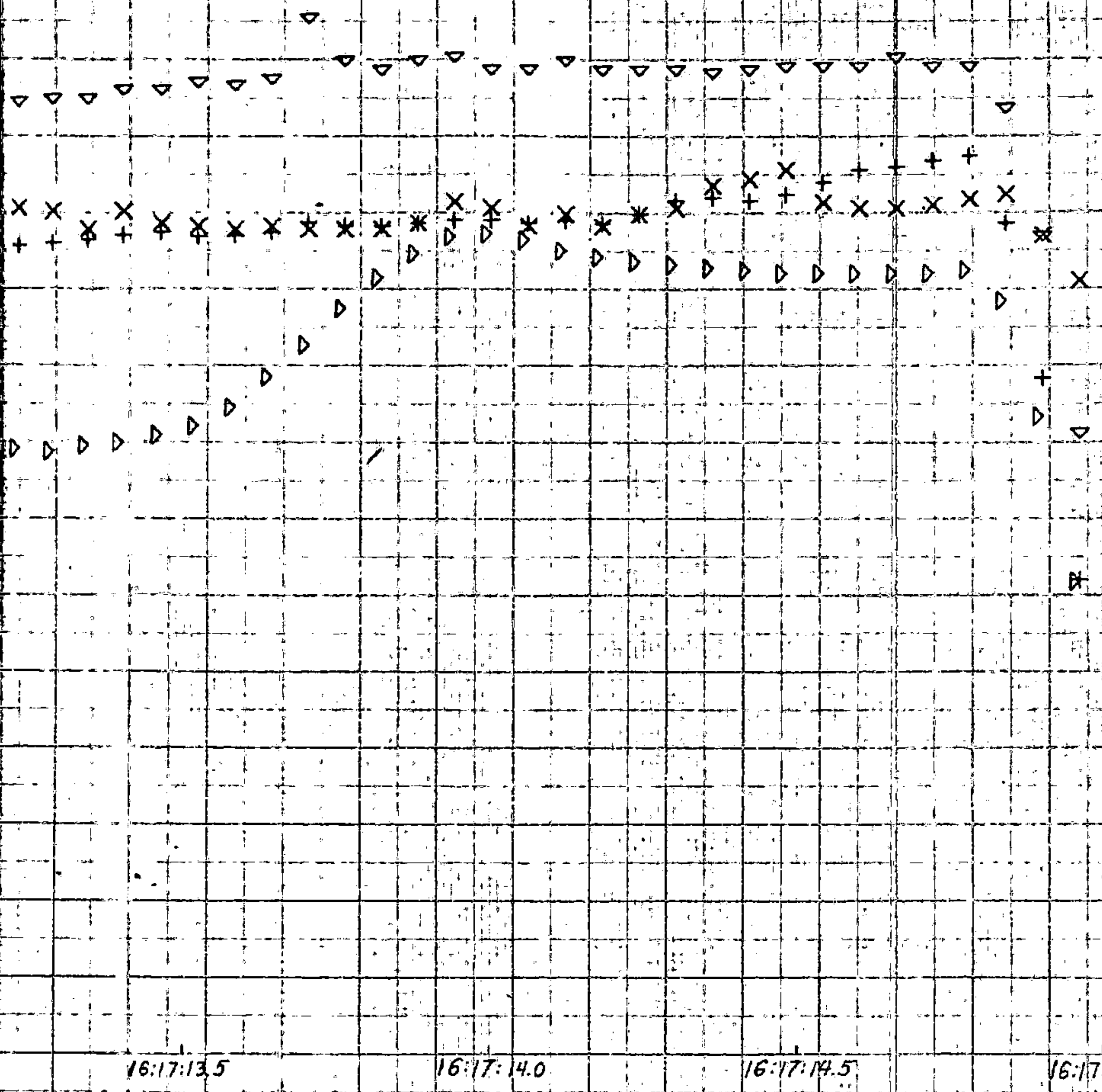


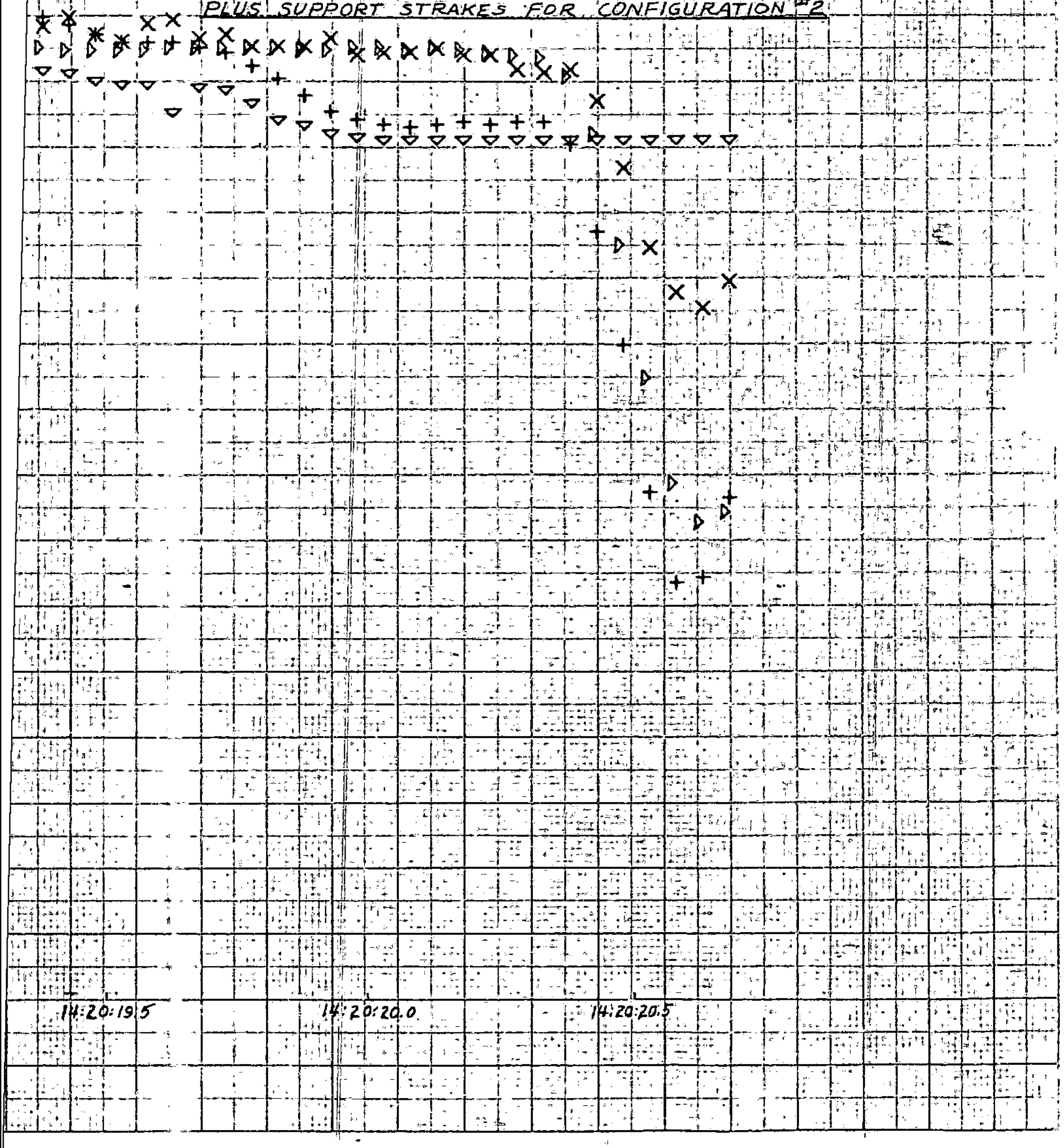
FIGURE 34

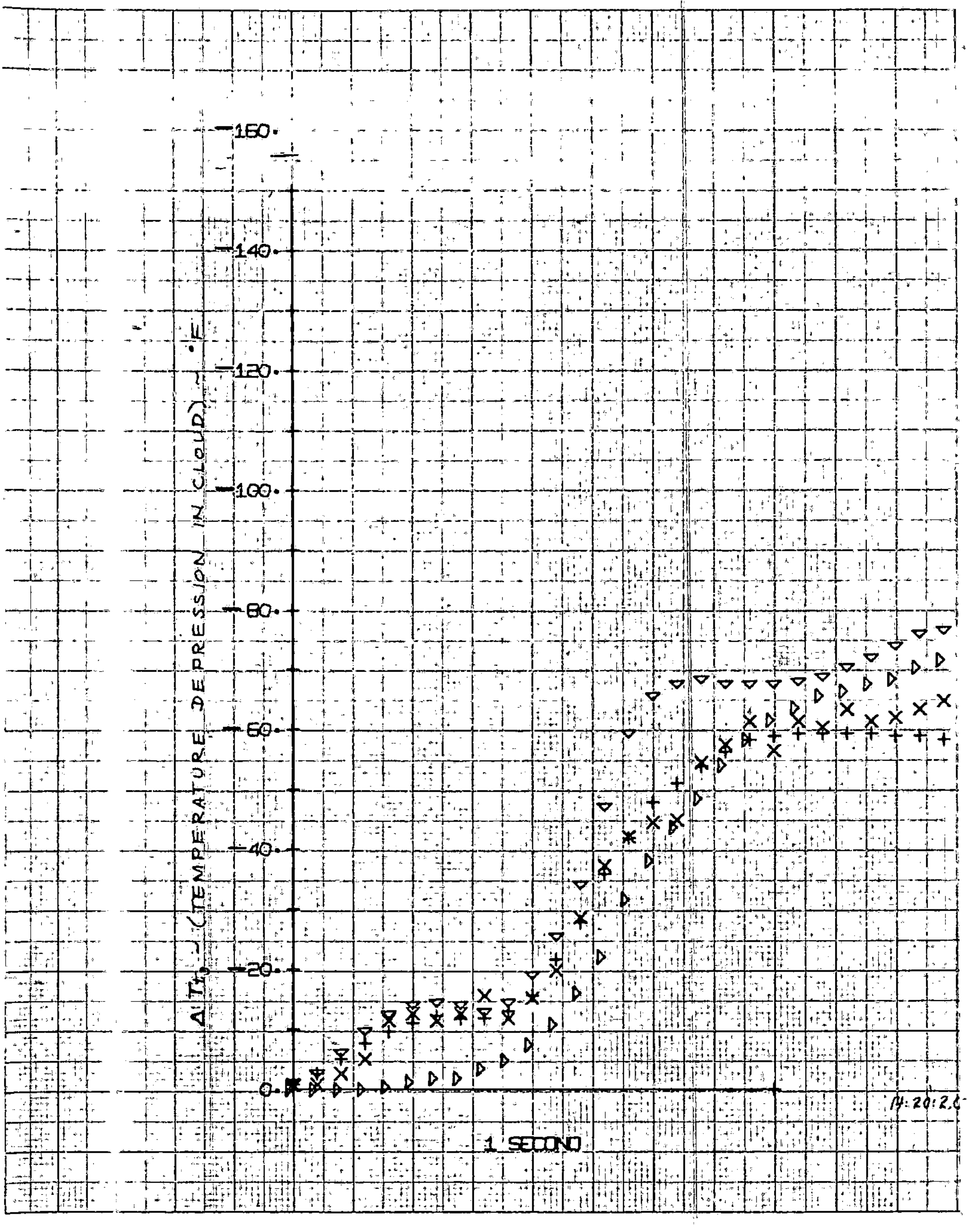
DEPRESSION (T_{t_2}) DURING CLOUD ENTRY

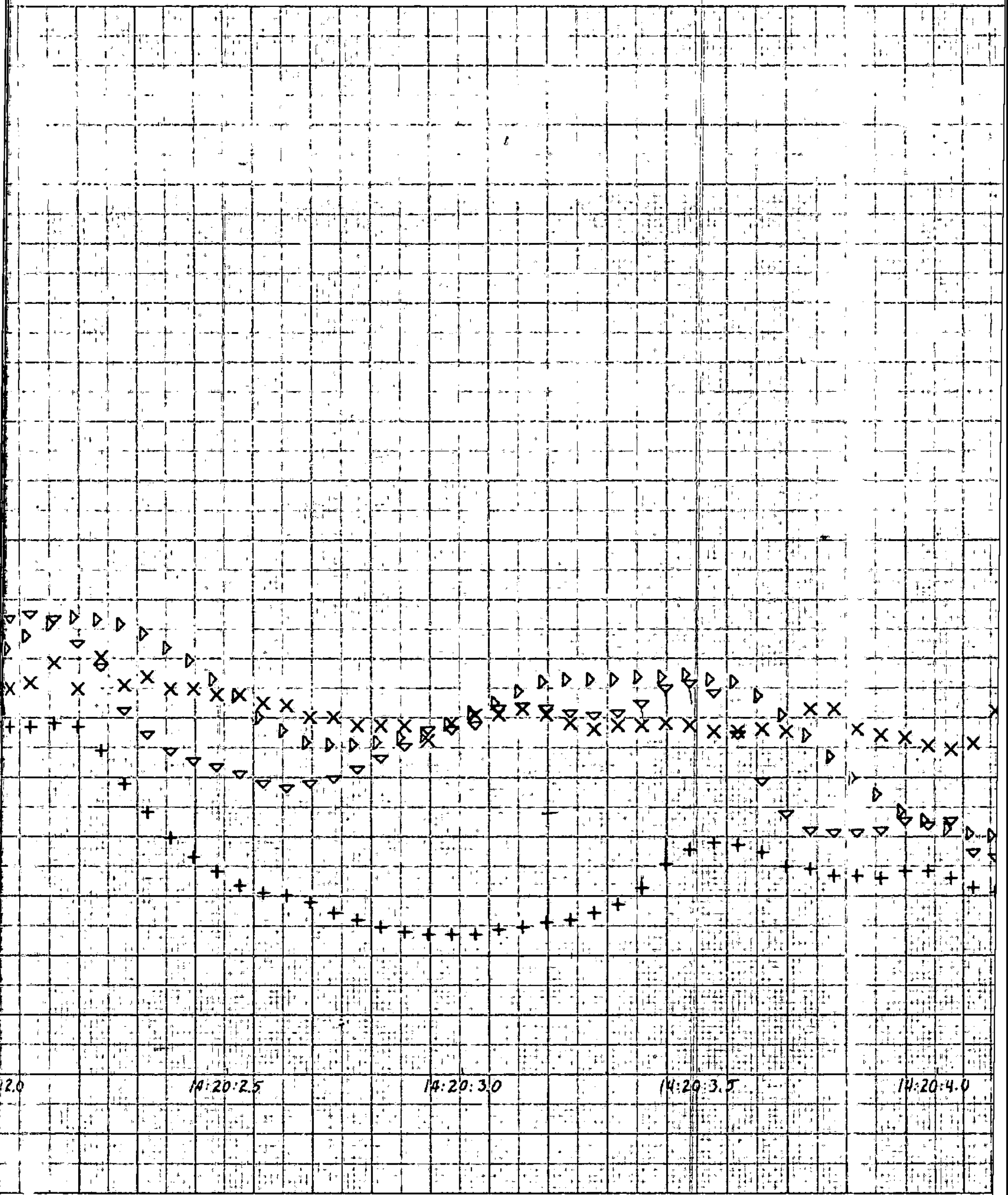
FLIGHT 64

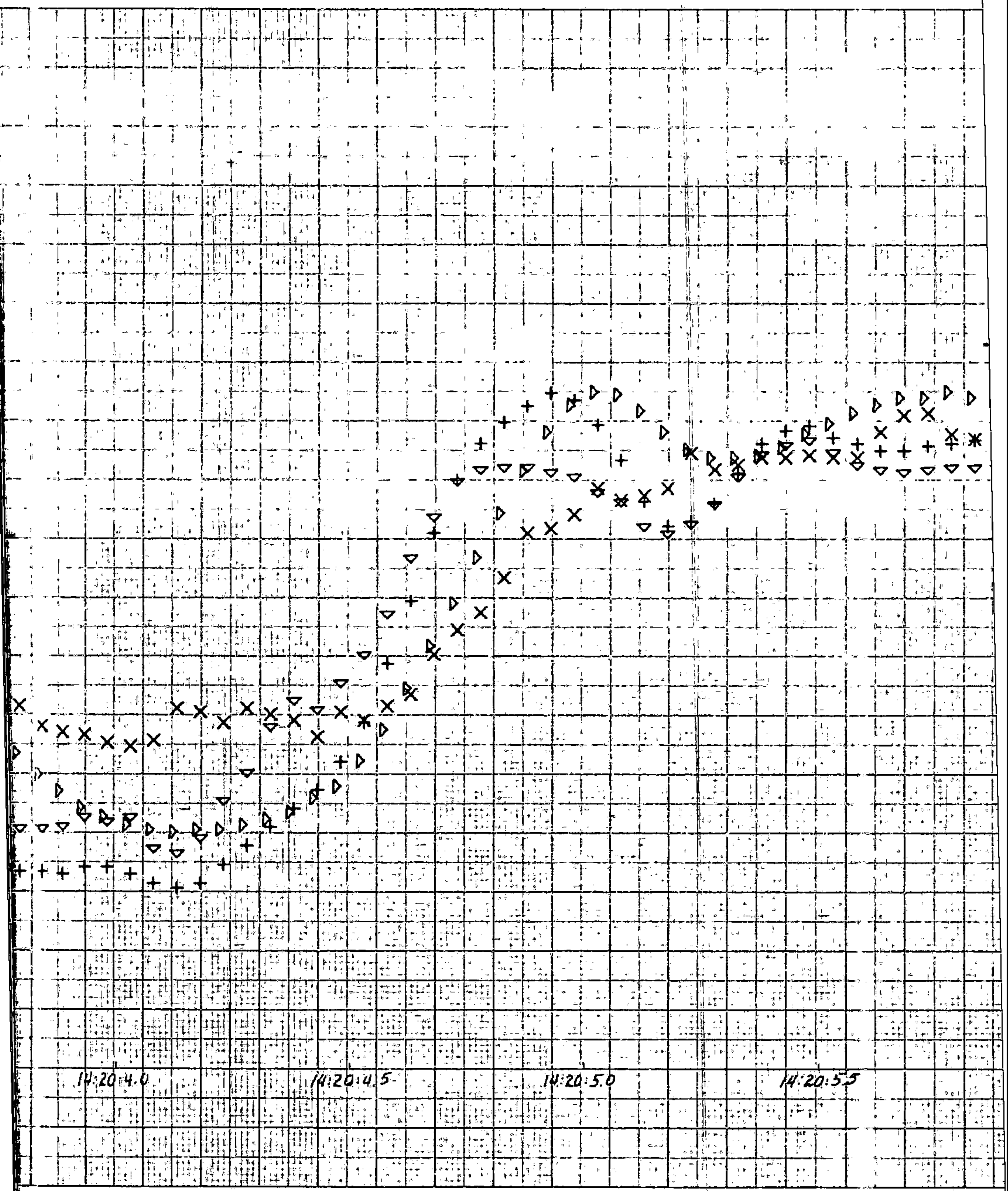
CONFIGURATION # 1 AT 6 DEGREES (REF. WATER LINE)

PLUS SUPPORT STRAKES FOR CONFIGURATION # 2




ΔT_D ~ (TEMPERATURE DEPRESSION IN CLOUD) ~ °F
160
140
120
100
80
60
40
20
0
1 SECOND
14:20:2.(


2.0
14:20:2.5
14:20:3.0
14:20:3.5
14:20:4.0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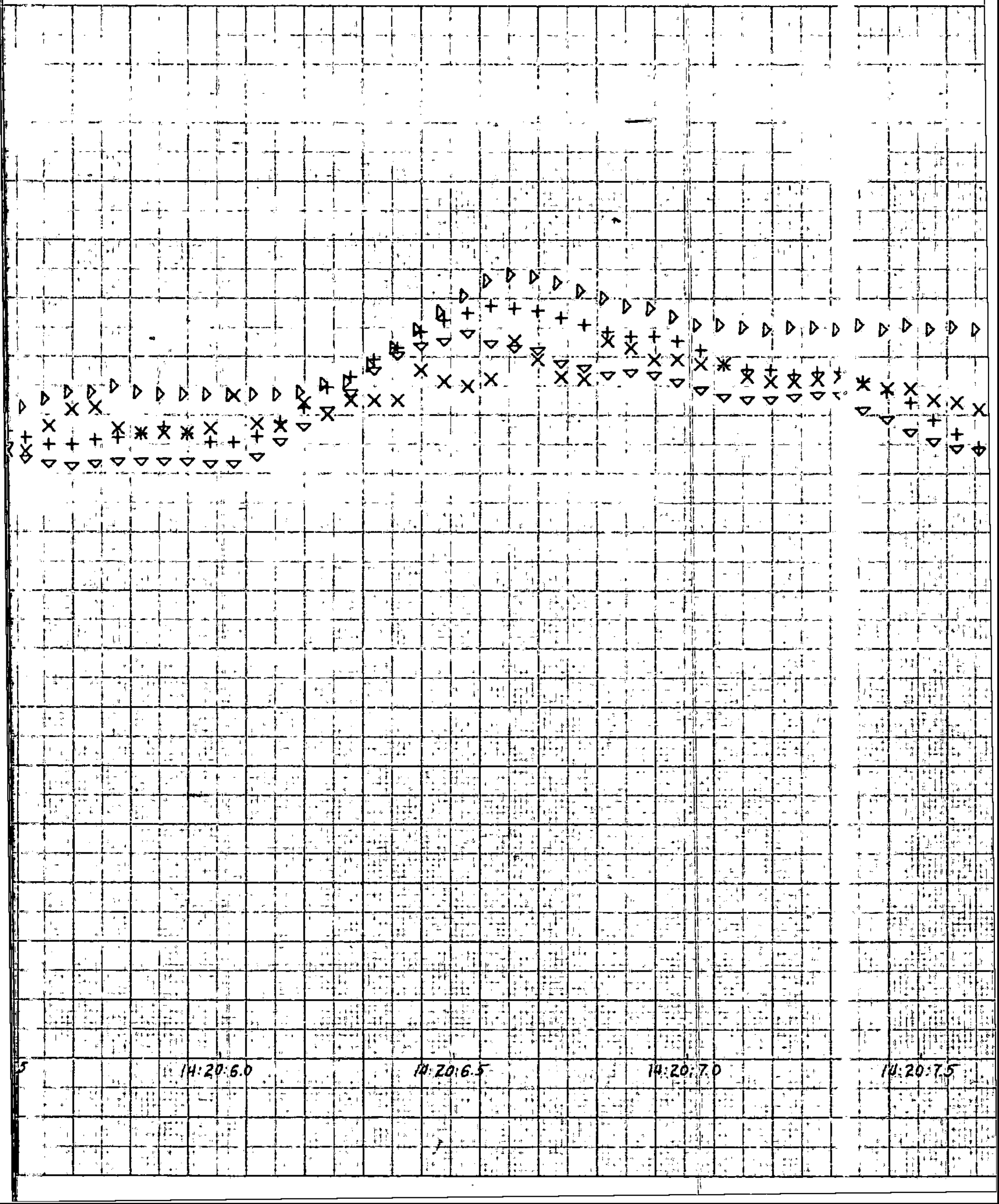
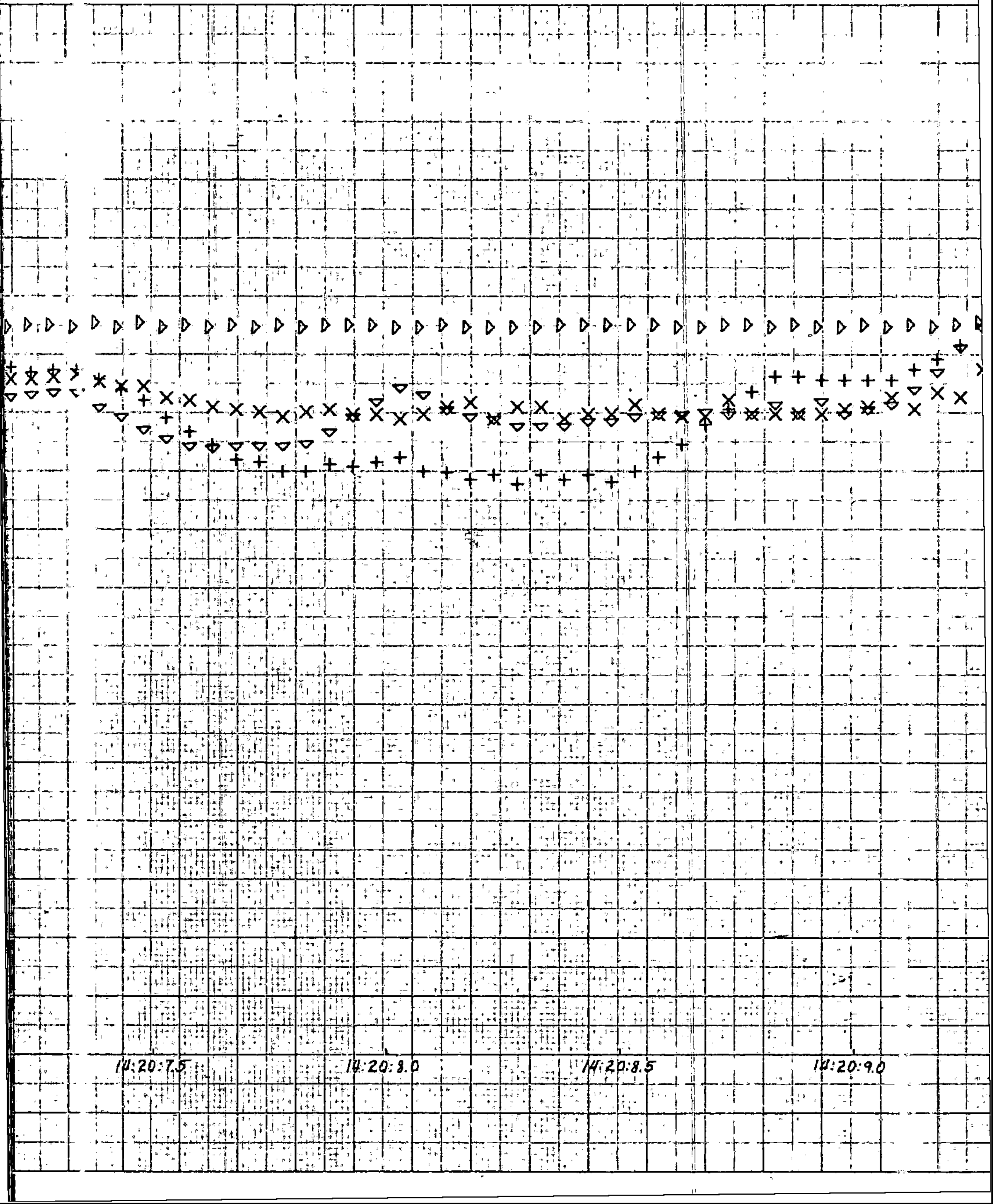
14:20:4.5
14:20:5.0
14:20:5.5


5
14:20:6.0
14:20:6.5
14:20:7.0
14:20:7.5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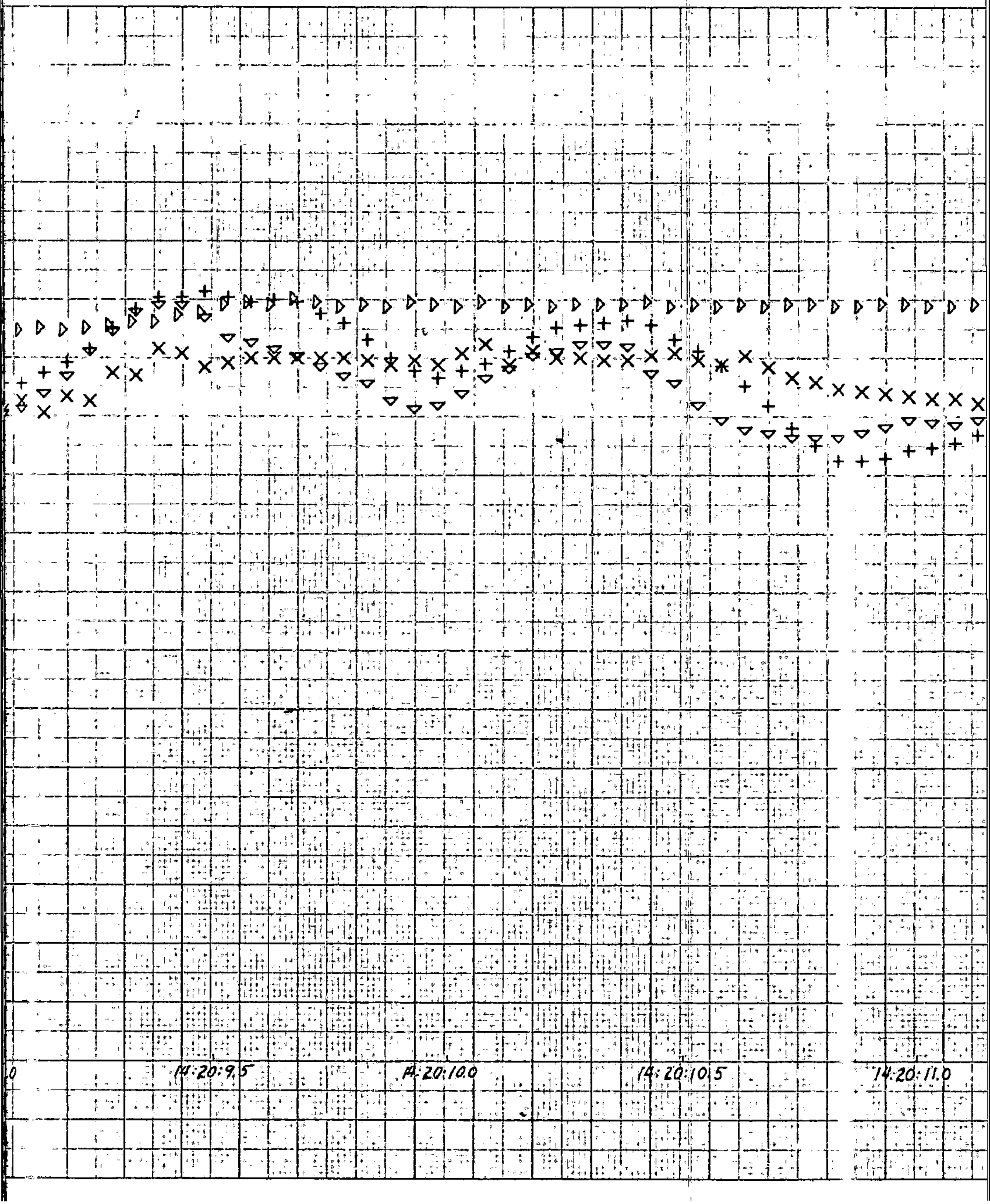
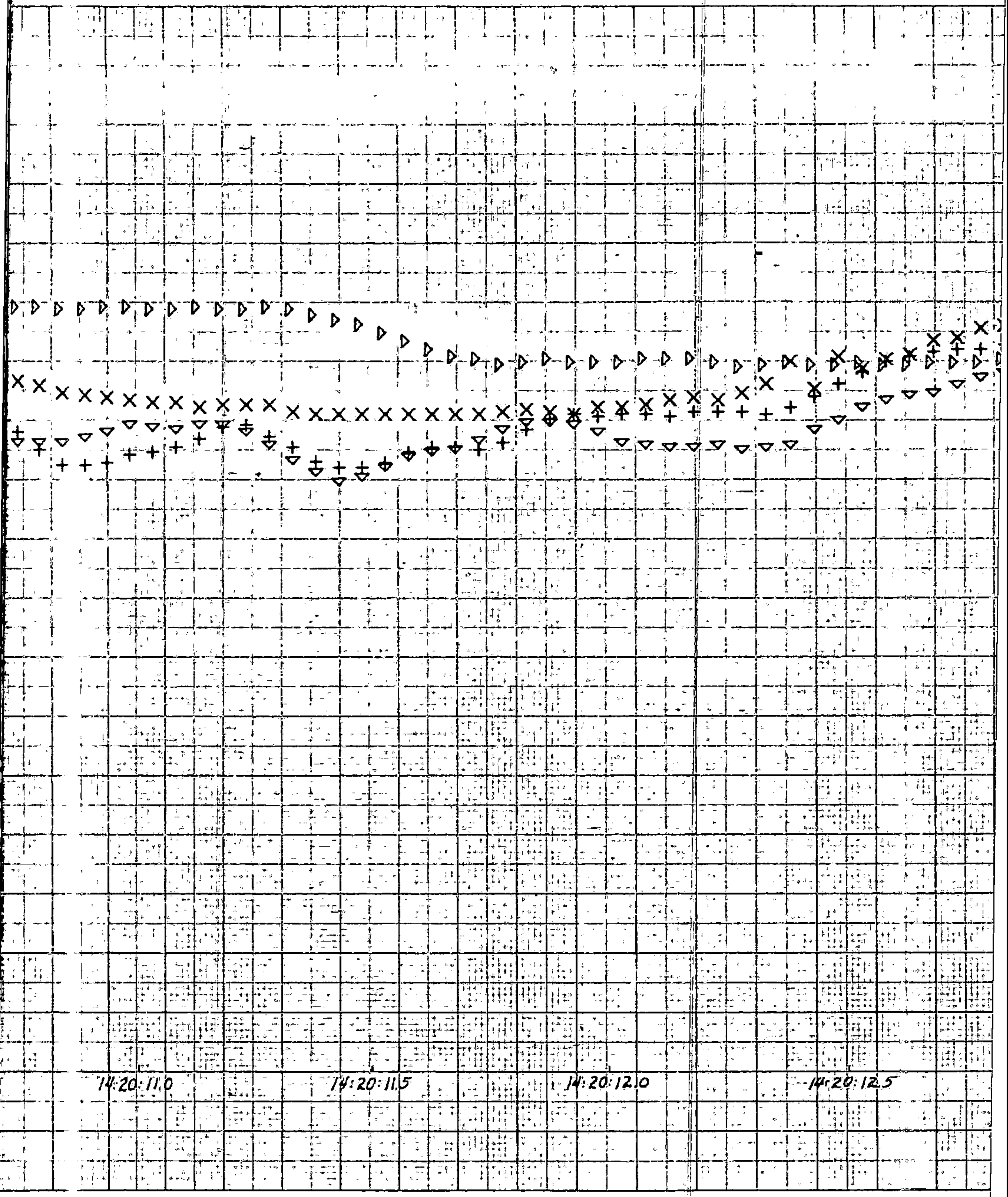
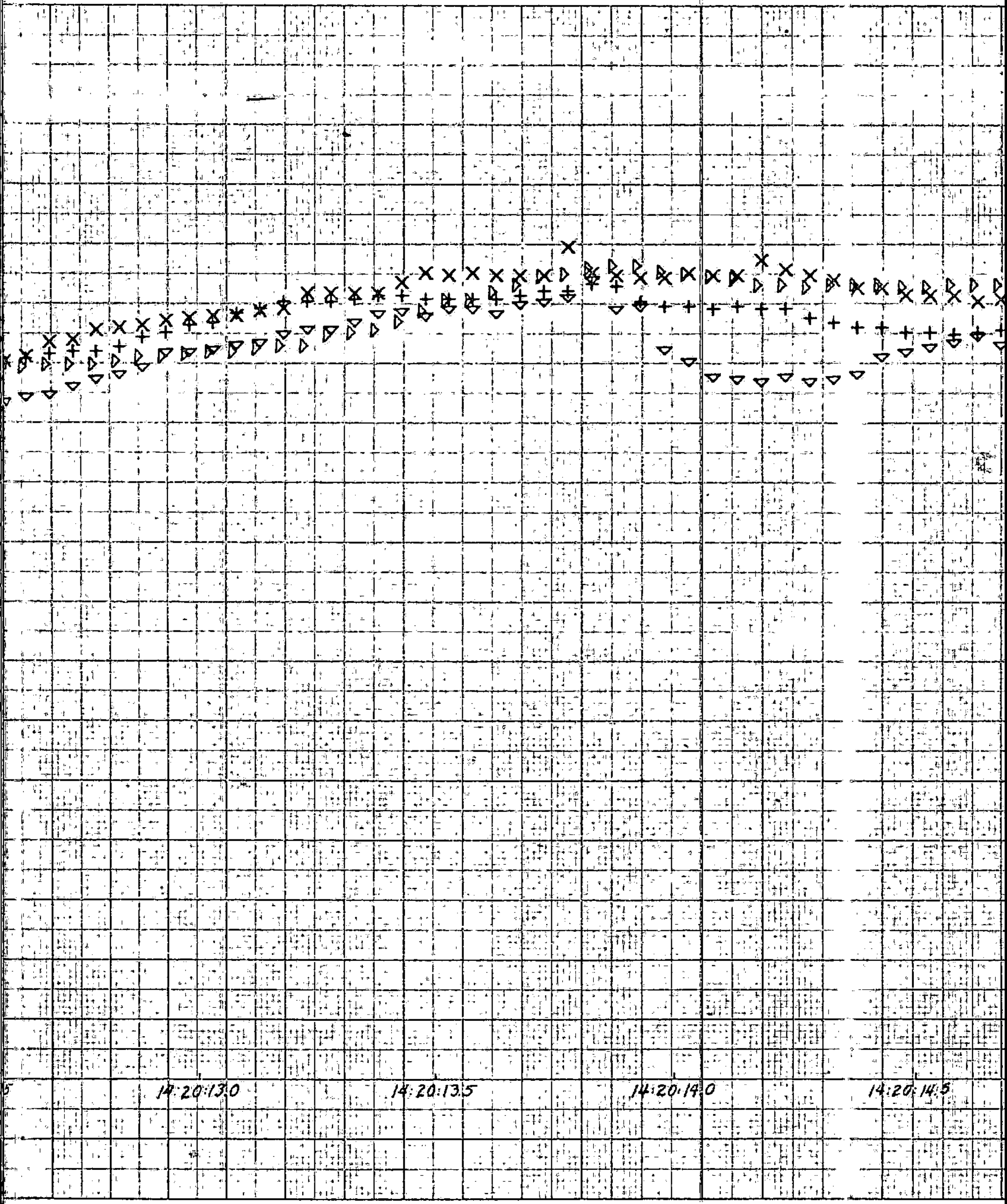
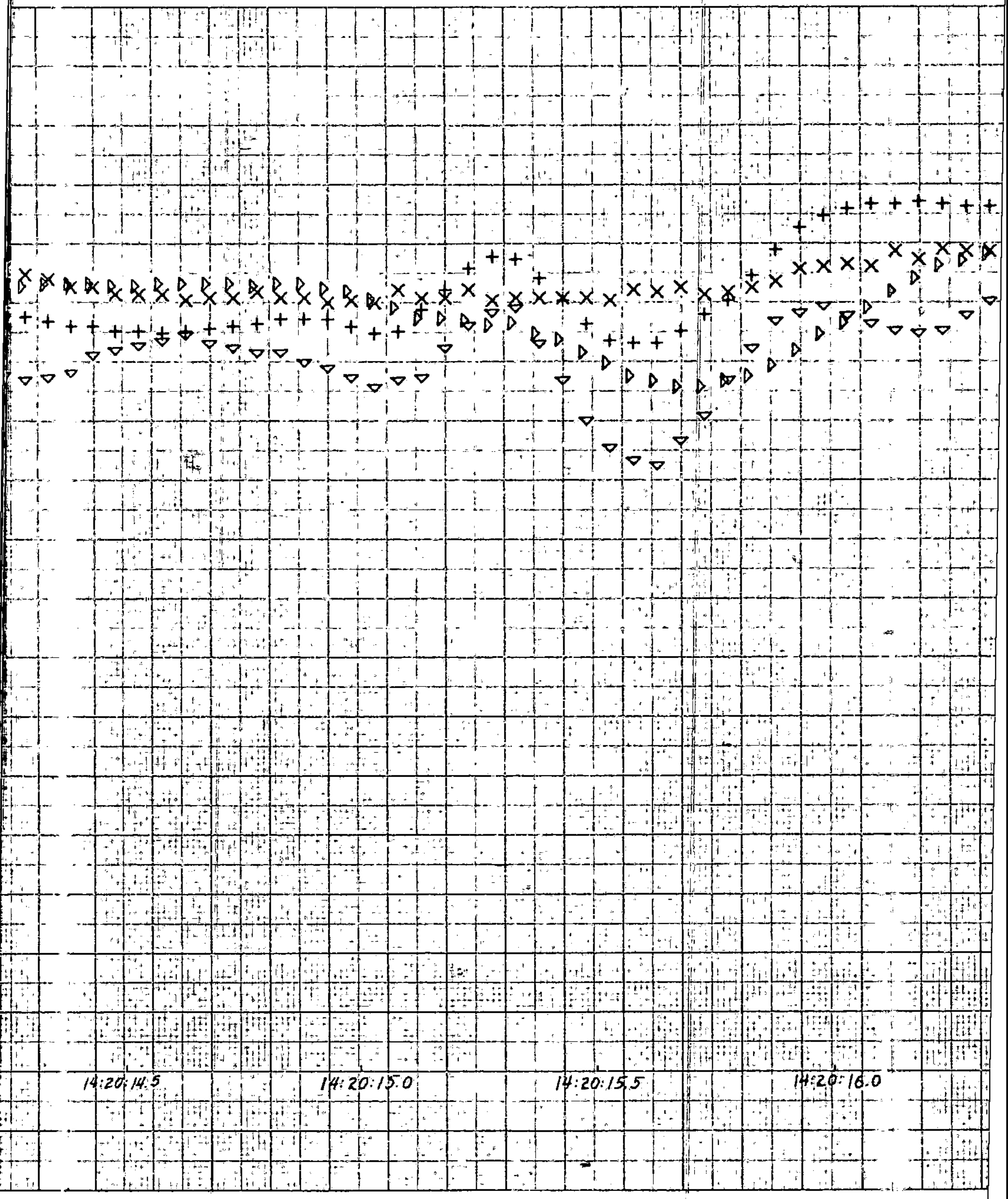
14:20:9.0


0
14:20:9.5
14:20:10.0
14:20:10.5
14:20:11.0


14:20:11.0
14:20:11.5
14:20:12.0
14:20:12.5


14:20:13.0
14:20:13.5
14:20:14.0
14:20:14.5


14:20:14.5
14:20:15.0
14:20:15.5
14:20:16.0

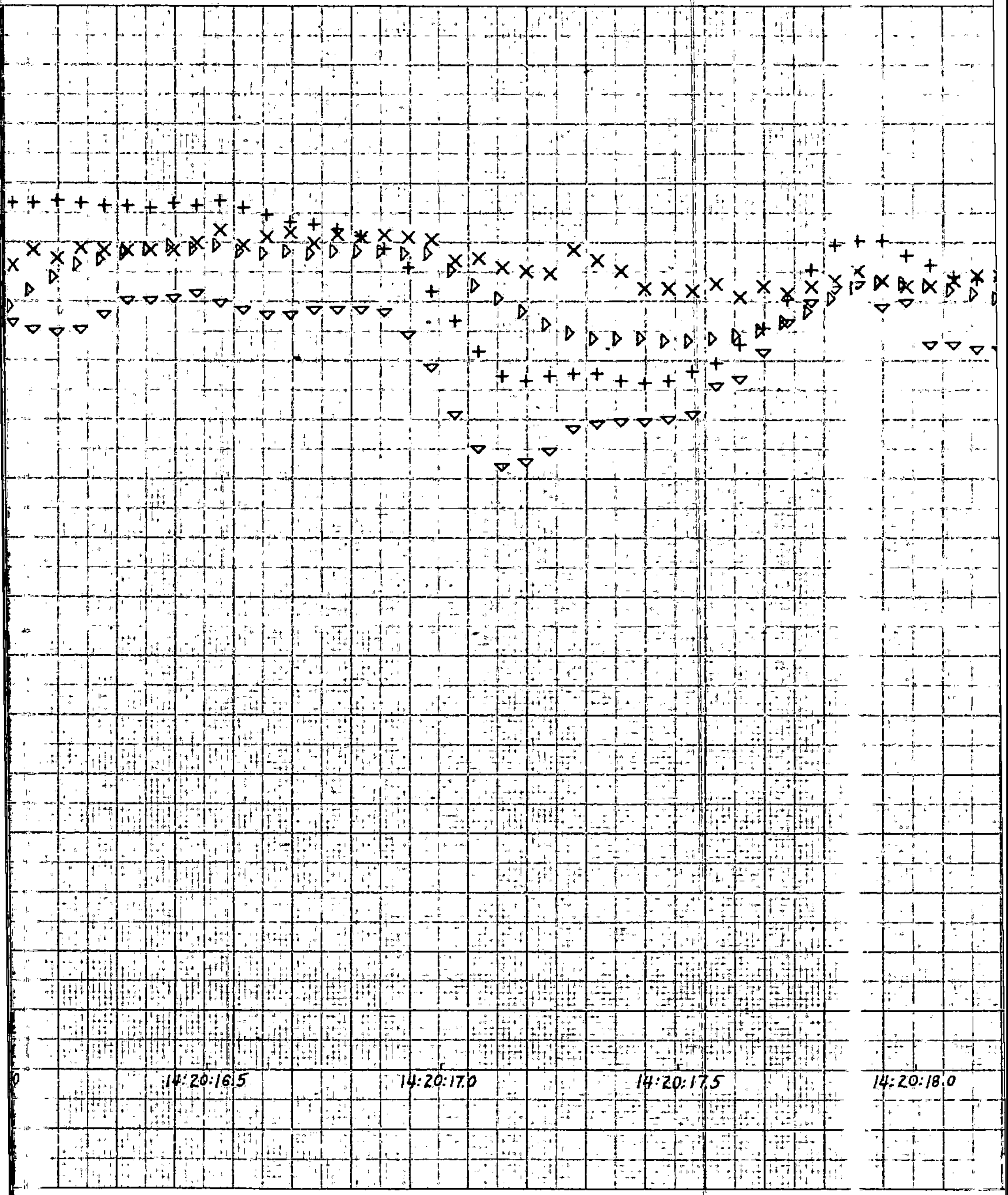

0
14:20:16.5
14:20:17.0
14:20:17.5
14:20:18.0

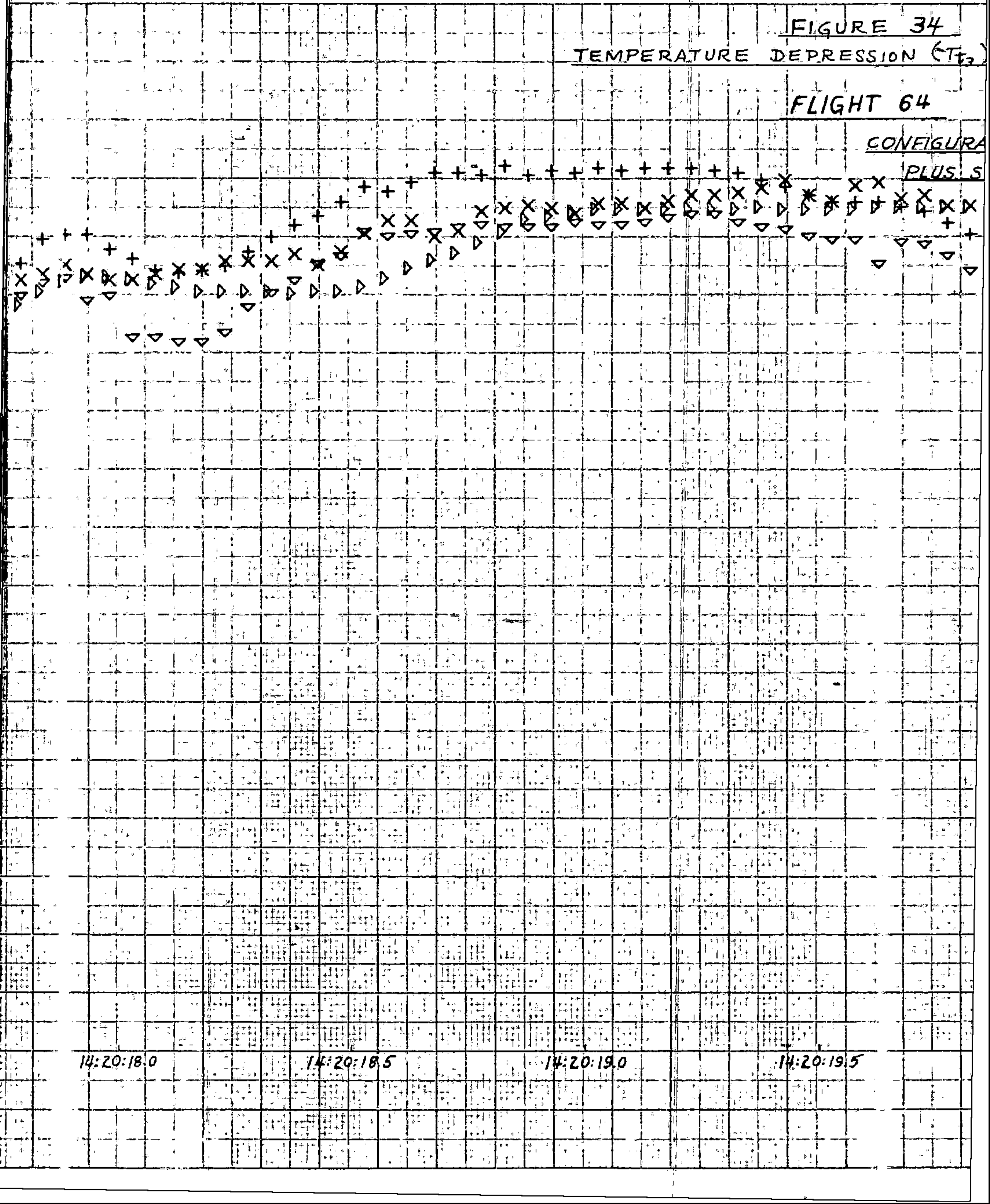
FIGURE 34

TEMPERATURE DEPRESSION (T_{t_2}

FLIGHT 64

CONFIGURA

PLUS S



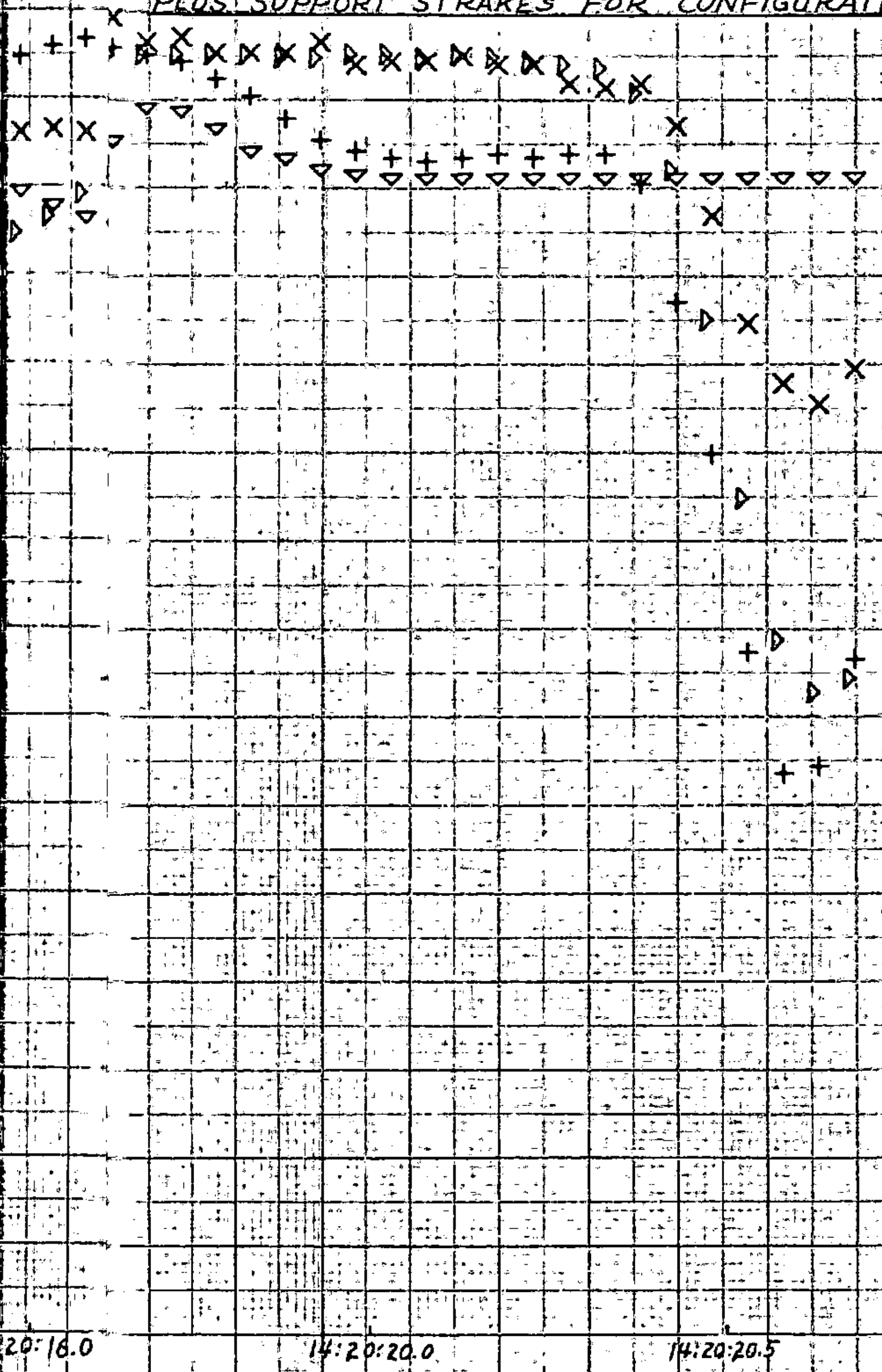
E 34

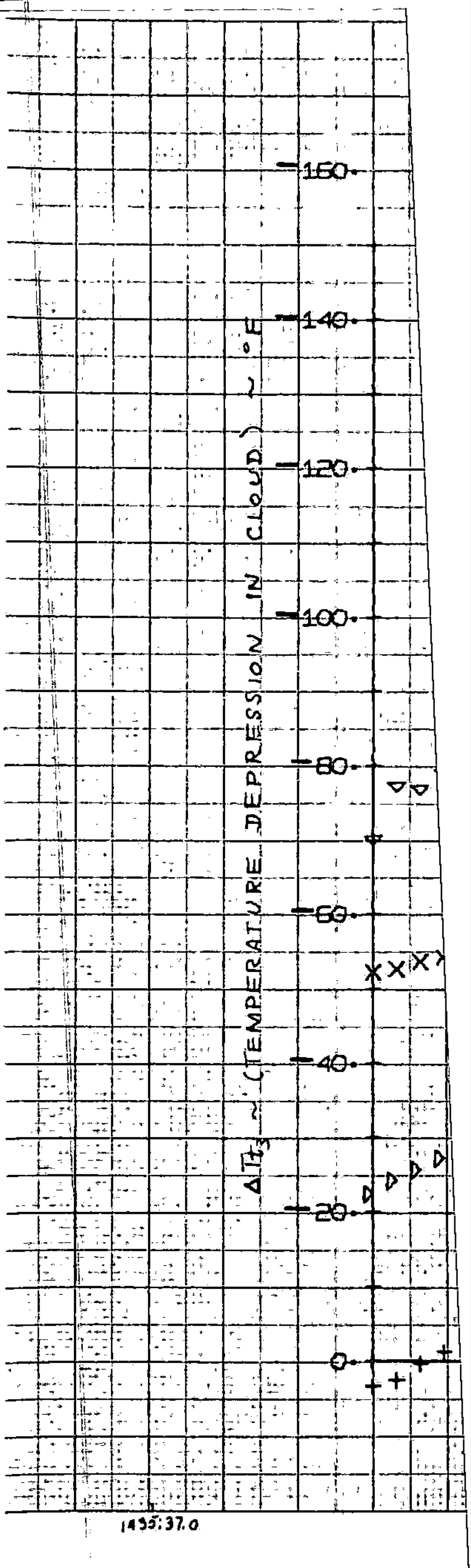
ION (ΔT_{t_2}) DURING CLOUD ENTRY

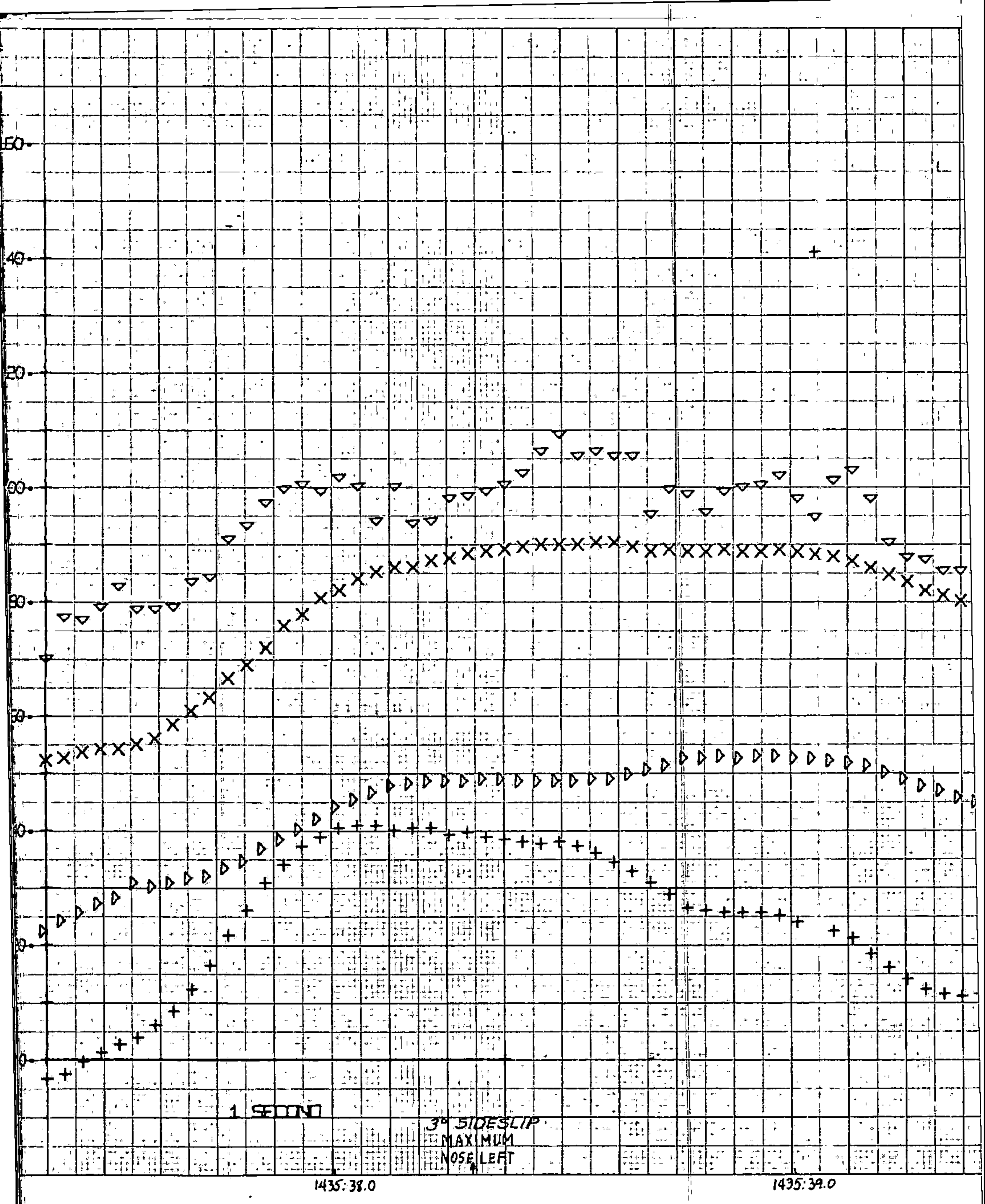
64

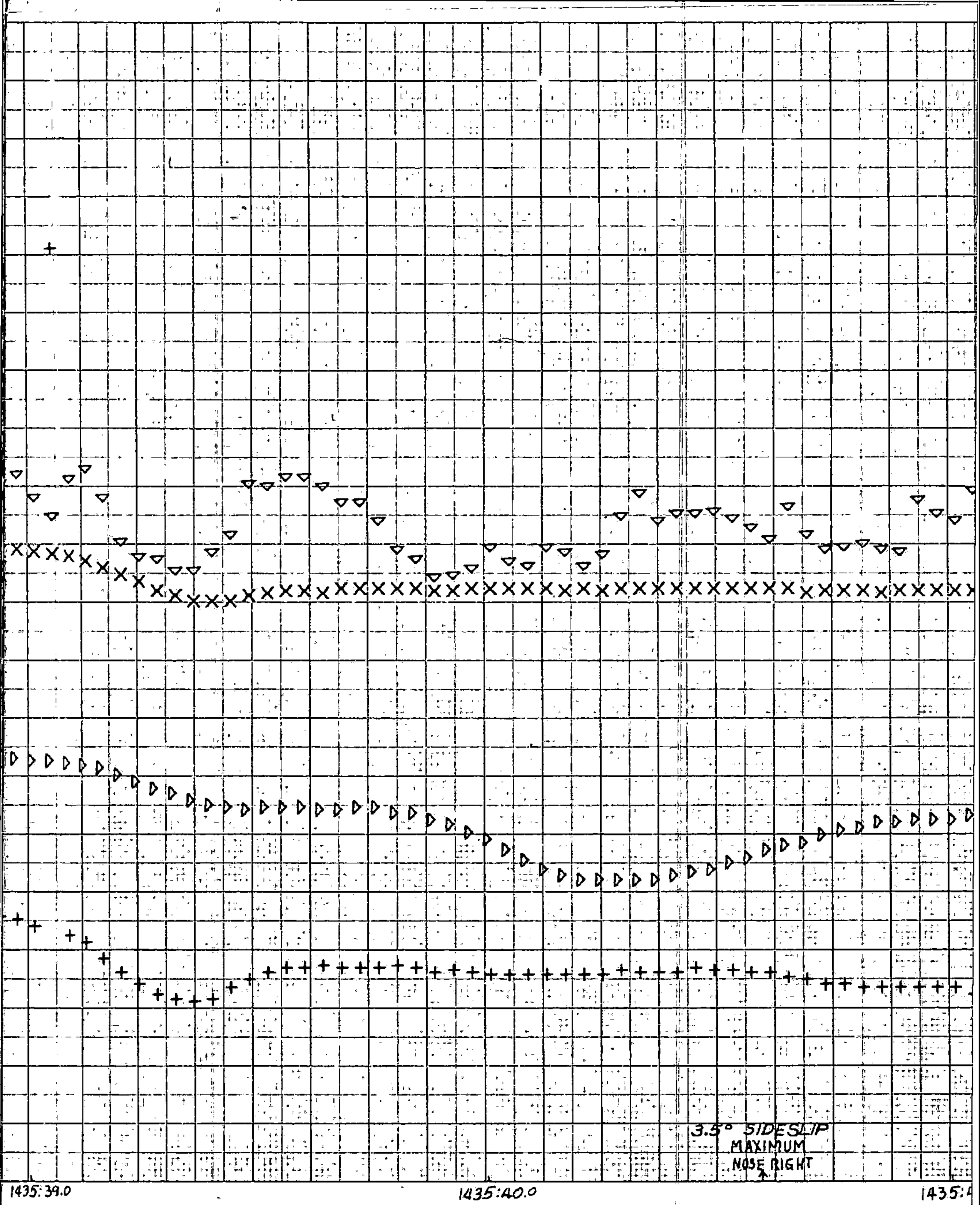
ONFIGURATION #1 AT 6 DEGREES (REF. WATER LINE)

PLUS SUPPORT STRAKES FOR CONFIGURATION #2

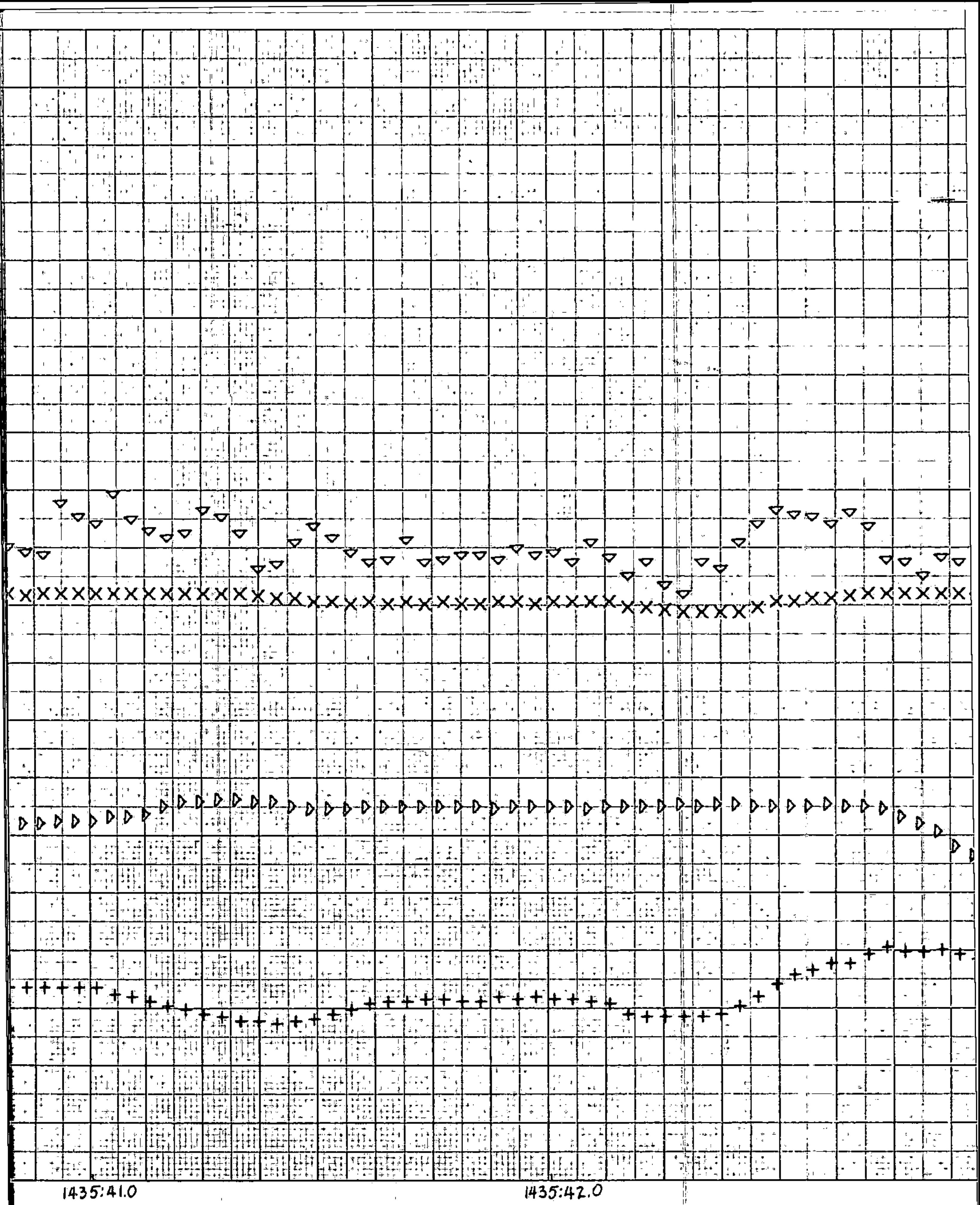


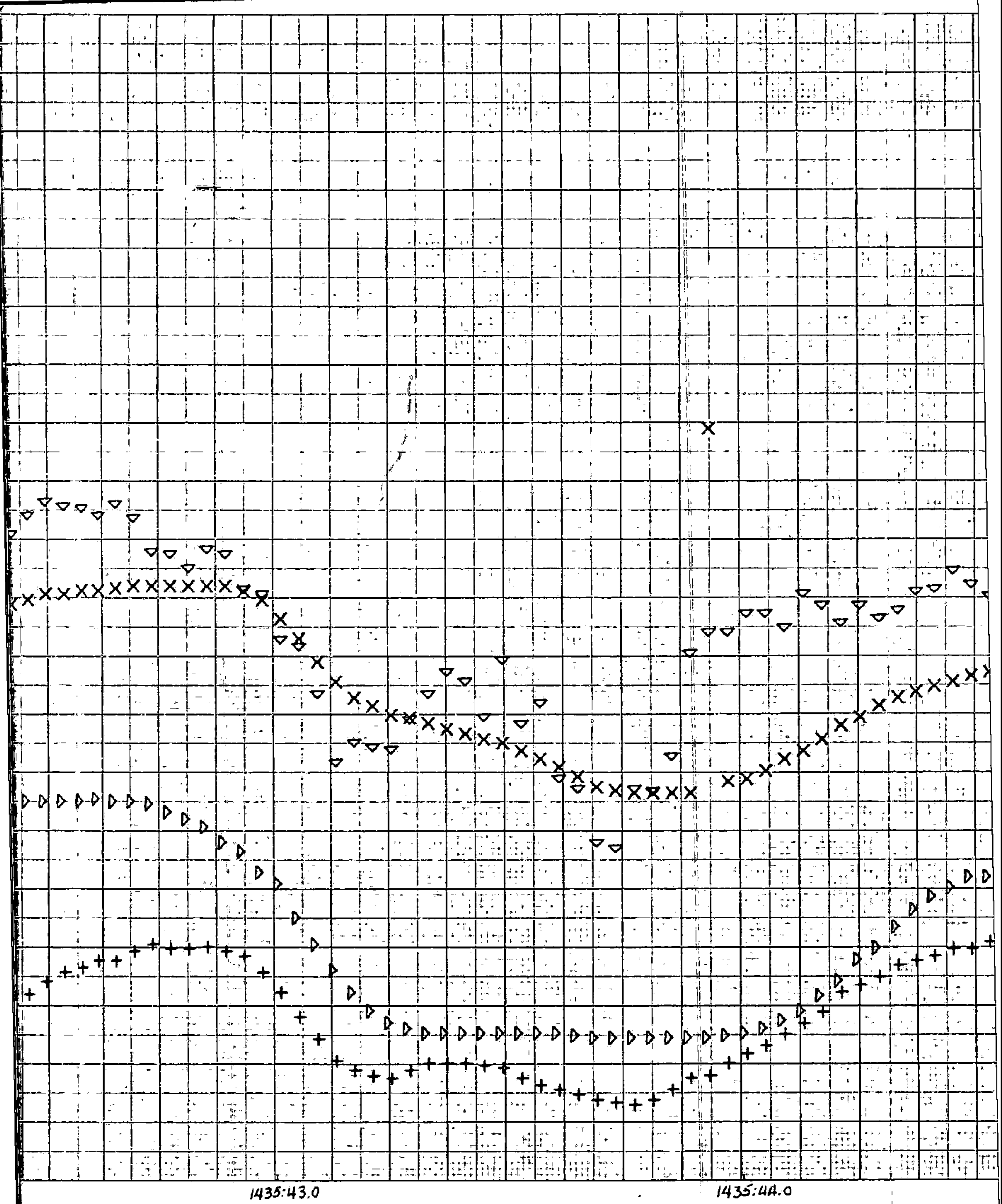

ΔTt₃ ~ (TEMPERATURE DEPRESSION IN CLOUD) ~ °F
160
140
120
100
80
60
40
20
0
1455:37.0

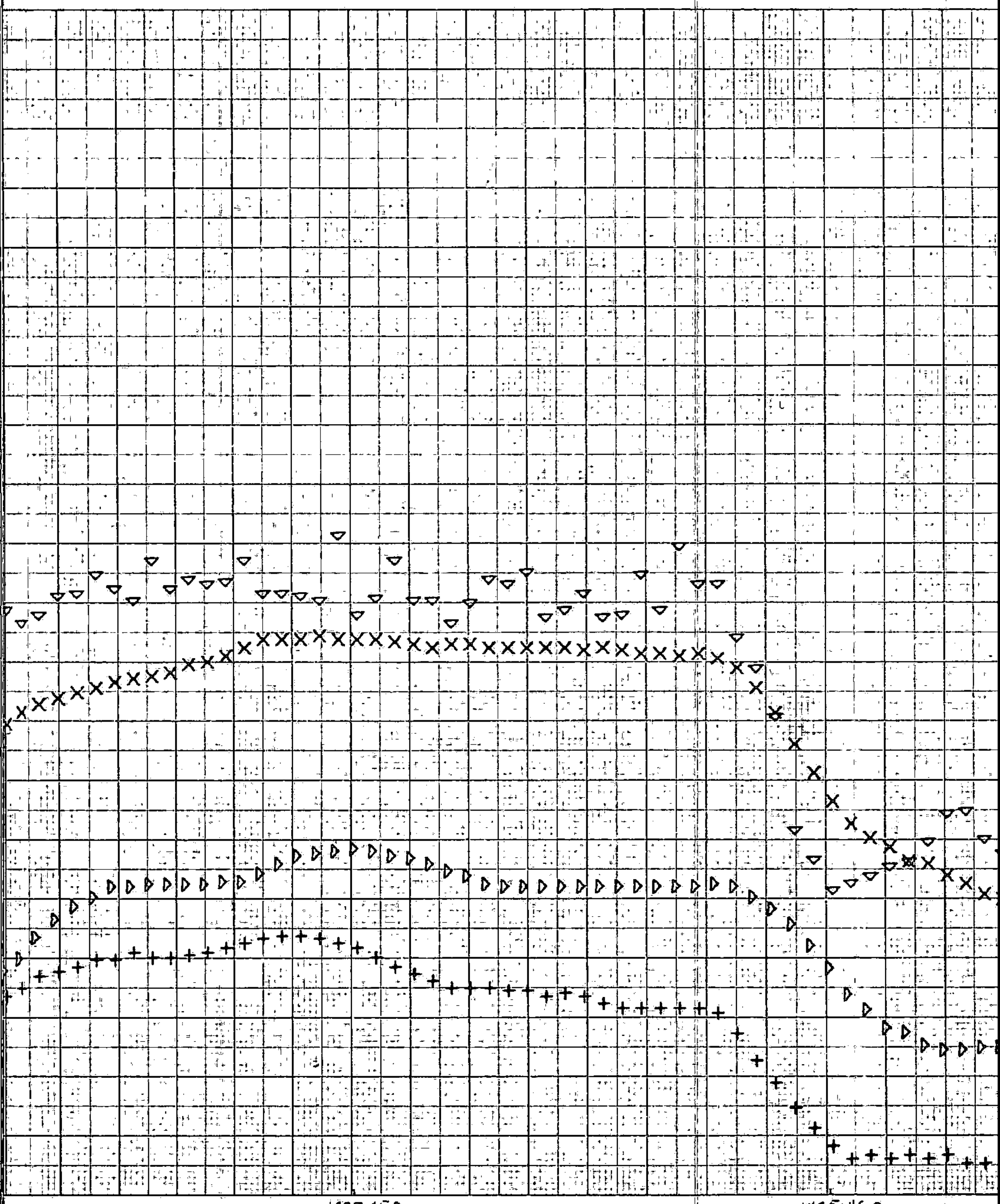

160
140
120
100
80
60
40
20
0
1 SECOND
3° SIDESLIP
MAXIMUM
NOSE LEFT
1435:38.0
1435:39.0

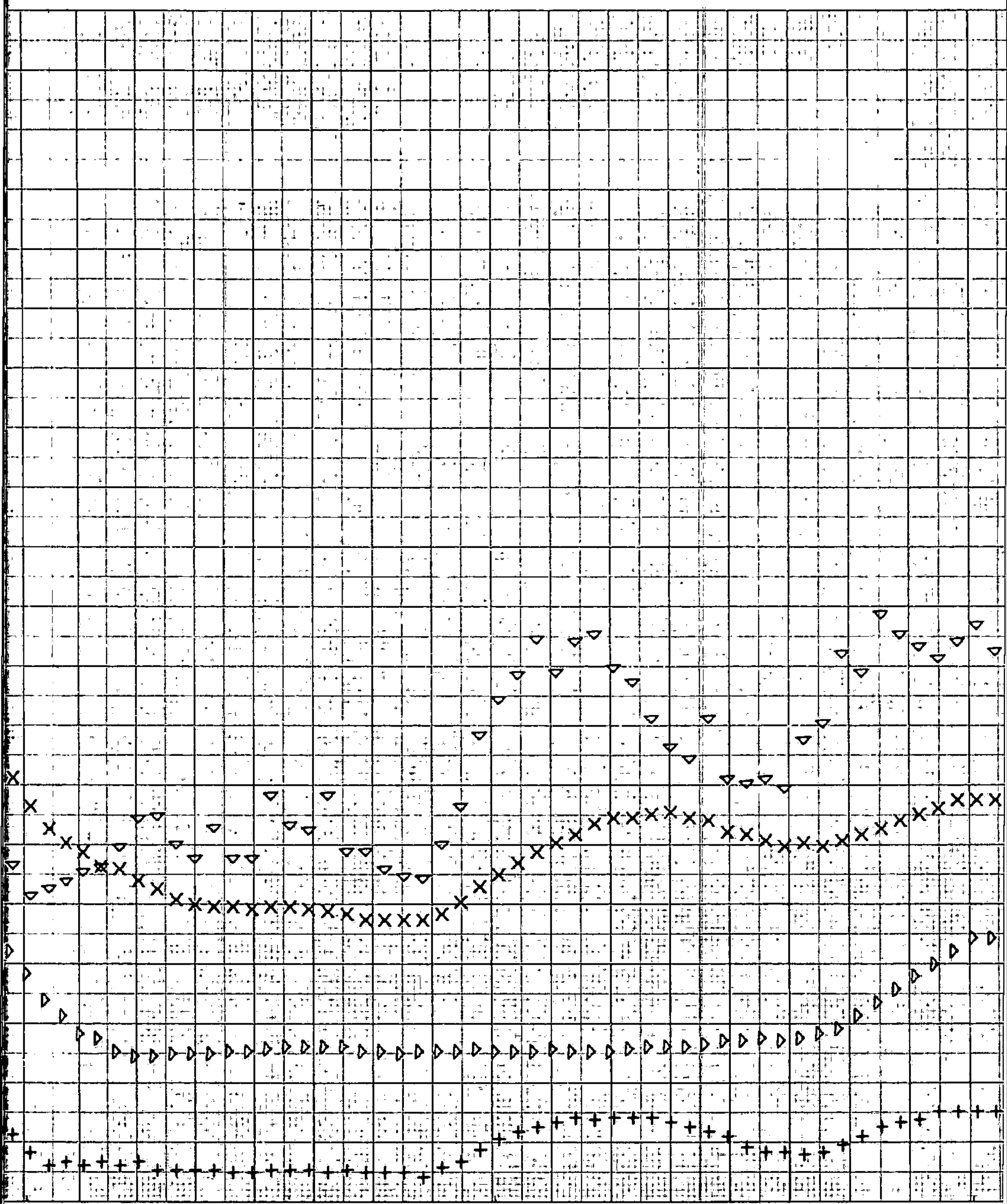


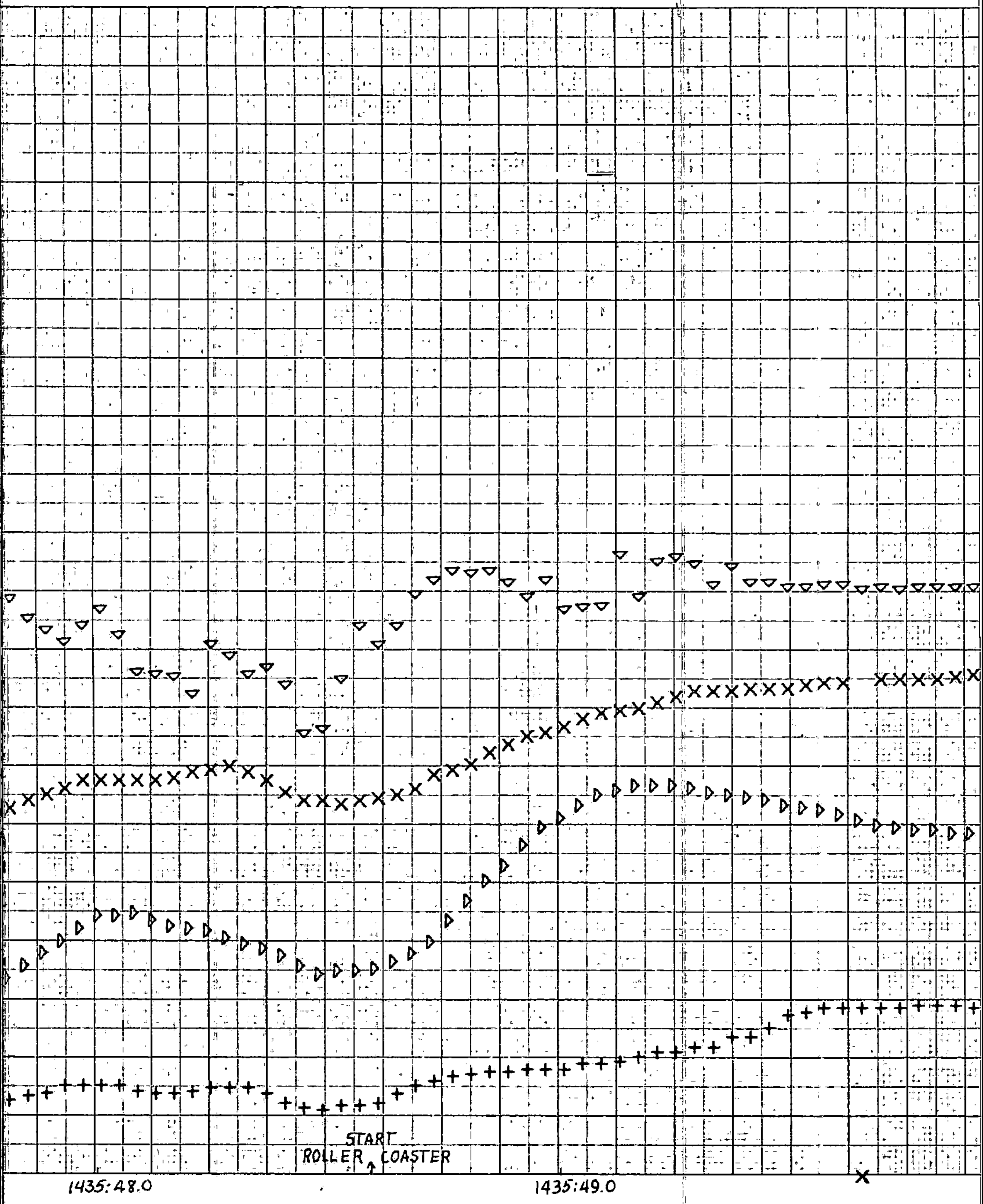
3.5° SIDESLIP
MAXIMUM
NOSE RIGHT


1435:41.0
1435:42.0

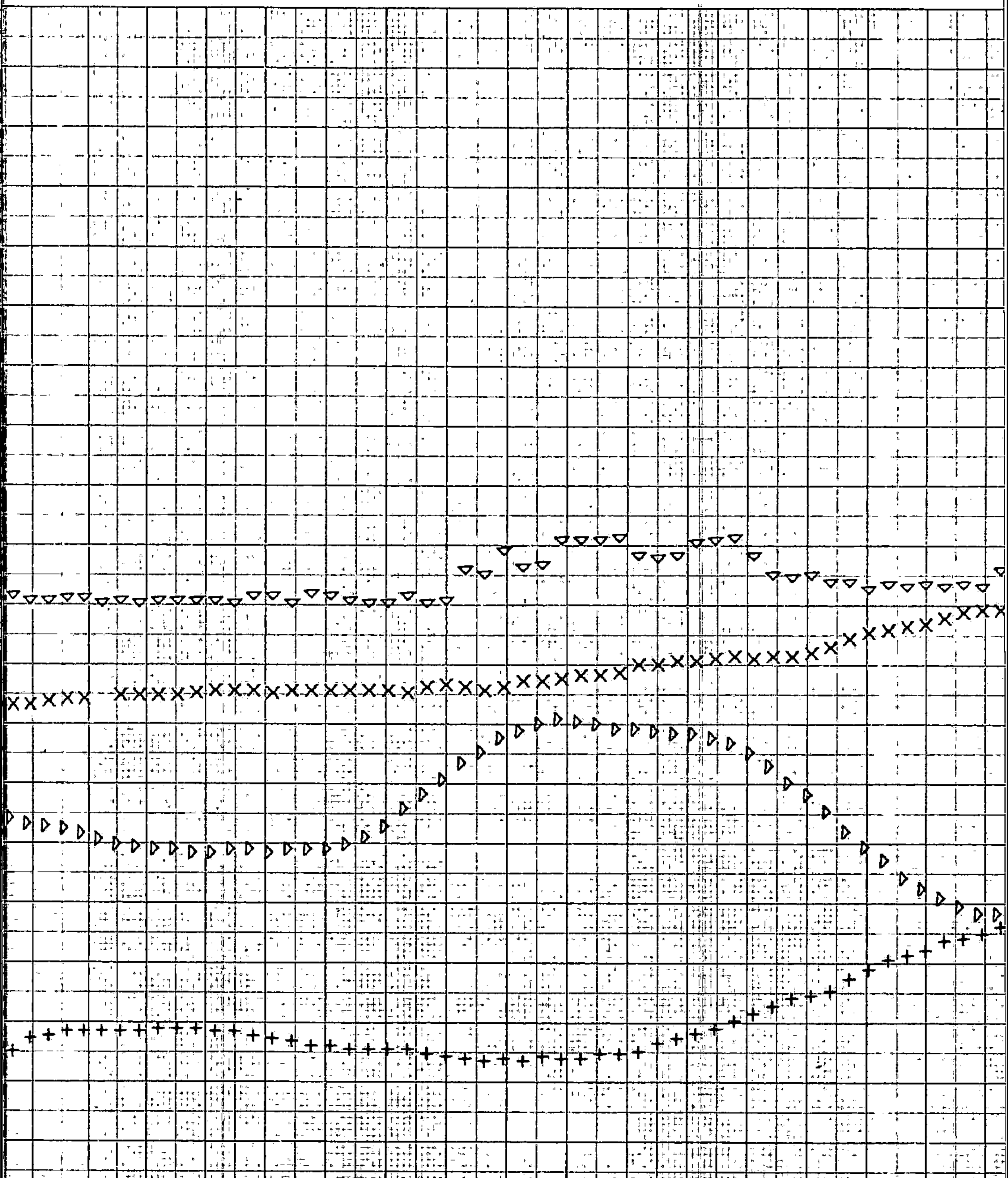

1435:43.0
1435:44.0

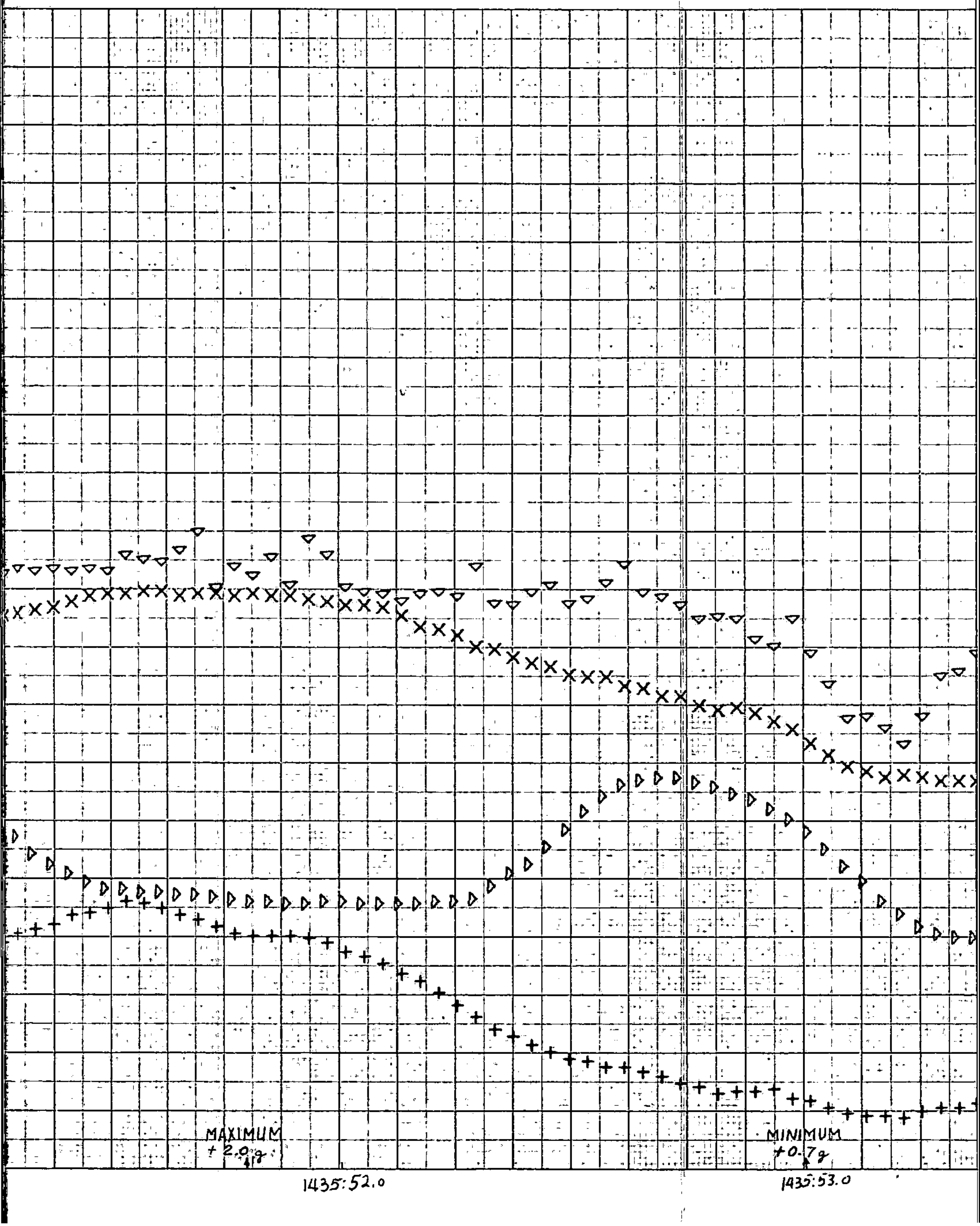

1435:45.0
1435:46.0


435:46.0
1435:47.0
1435:4



START
ROLLER COASTER
1435:48.0
1435:49.0


1435:50.0
1435:51.0


MAXIMUM
+2.0g
MINIMUM
+0.7g
1435:52.0
1435:53.0

FIG

TEMPERATURE DEPRES

(INCLUDING YAW AND RO

FLIGH

MANEU

CONFIGURATION #6

1½ INCH STRAKES AT 2 DEGREES(R

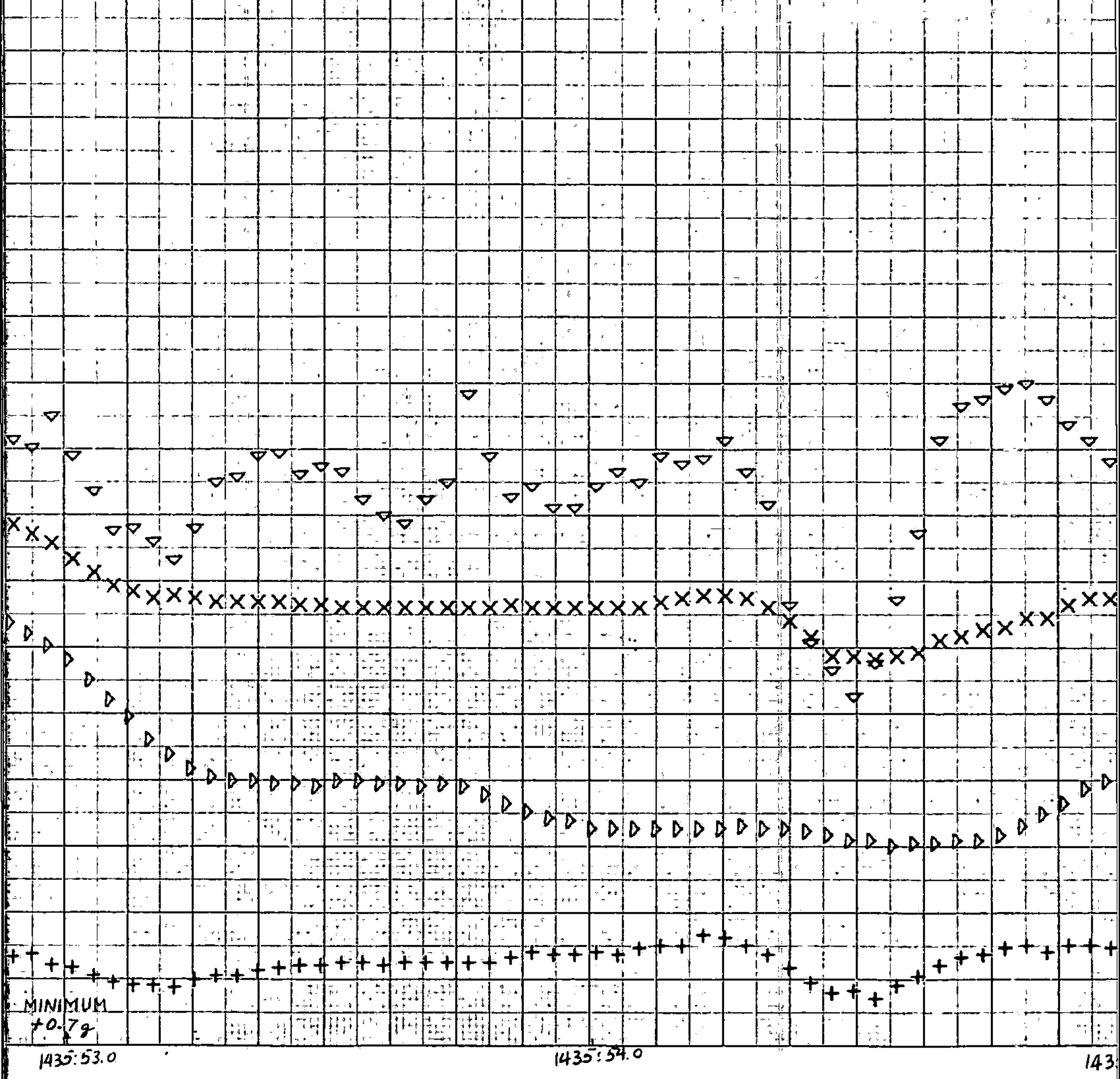


FIGURE 35

MPERATURE DEPRESSION (T_{t_2}) DURING CLOUD ENTRY

CLUDING YAW AND ROLLER COASTER MANEUVERS)

FLIGHT 75

MANEUVERS

ONFIGURATION #6

AKES AT 2 DEGREES (REF DUCT FLOOR)

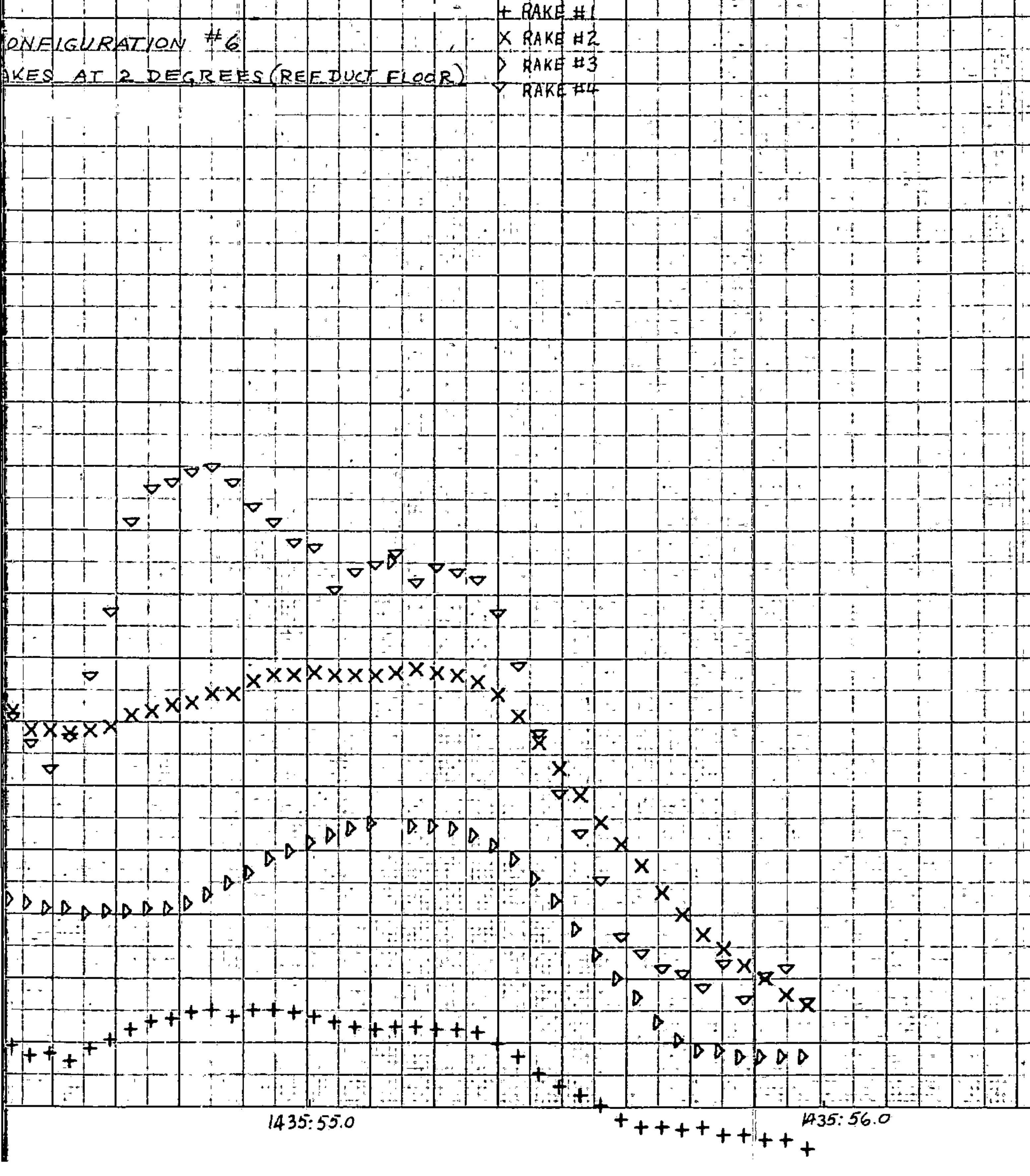
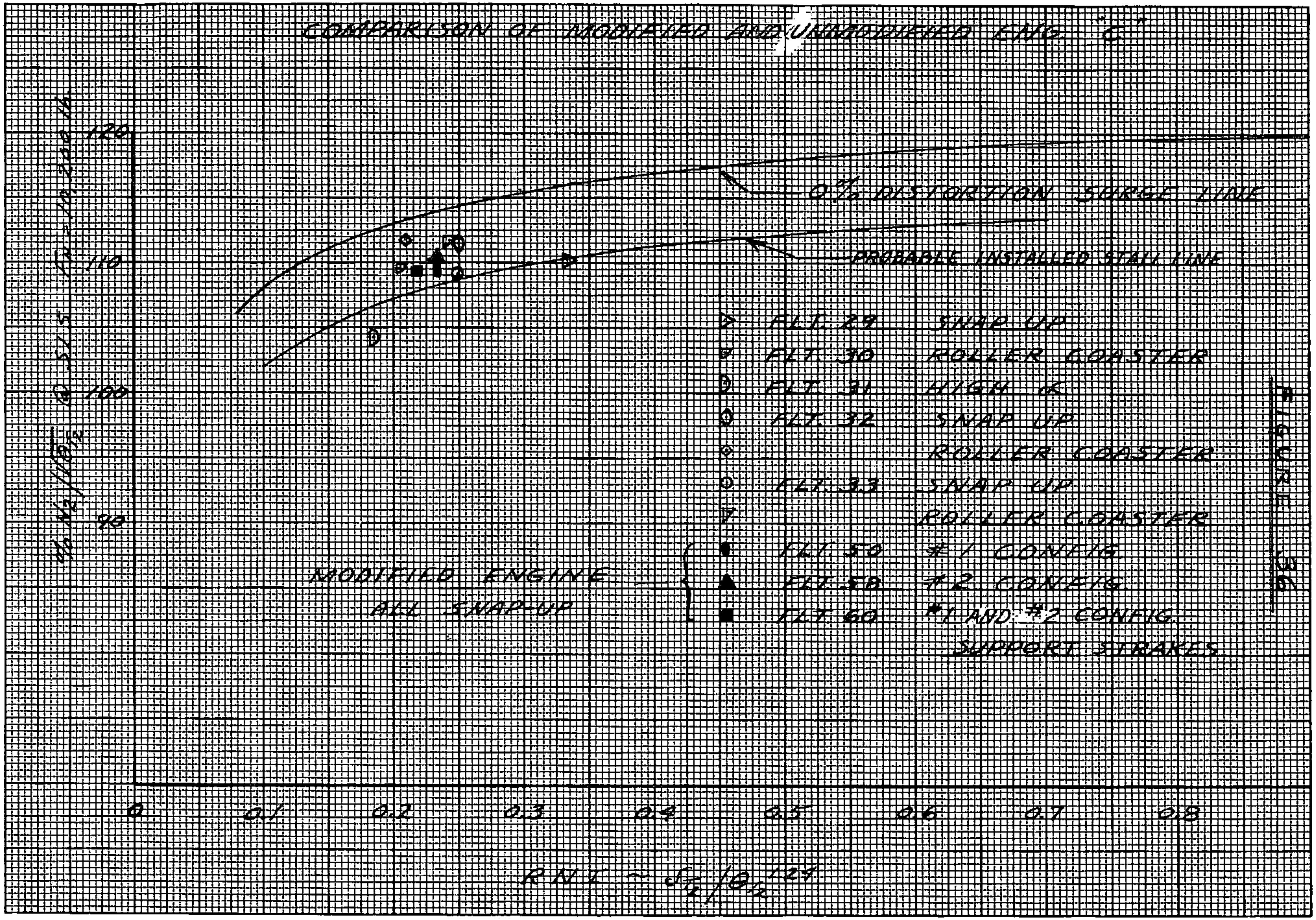


FIGURE 36

COMPARISON OF MODIFIED AND UNMODIFIED ENG "C"



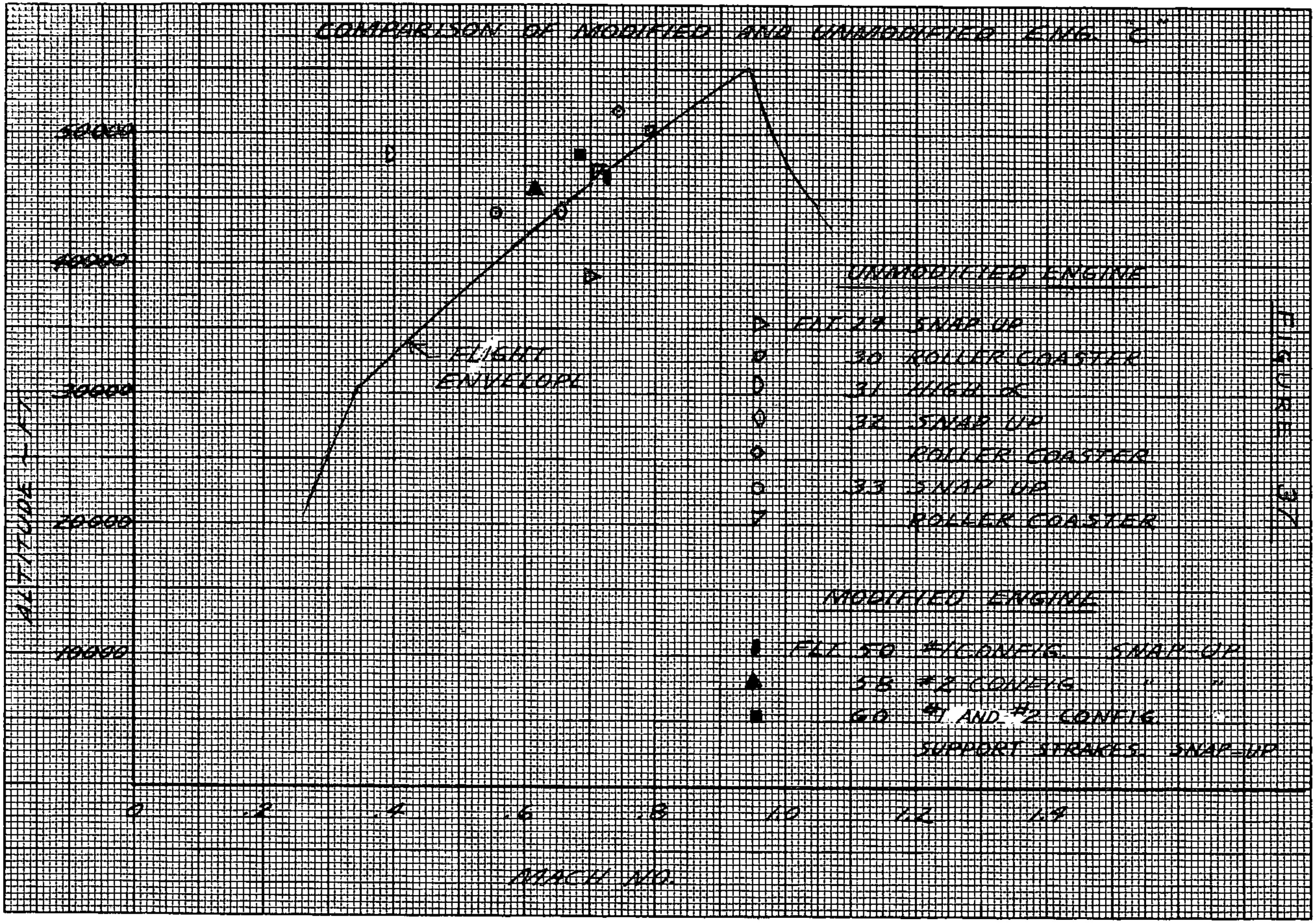


FIGURE 37

COMPARISON OF MODIFIED AND UNMODIFIED ENG. "C"

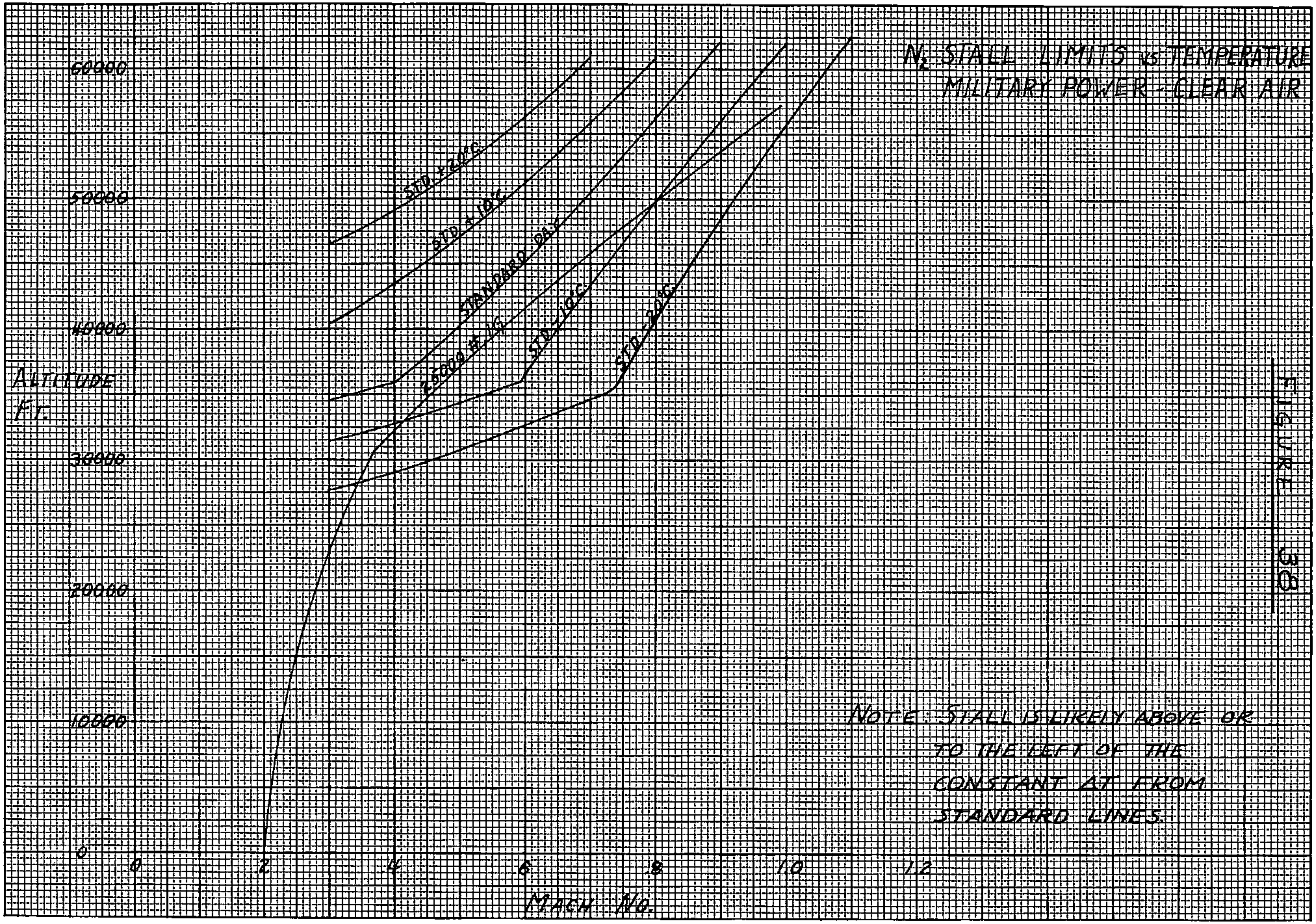
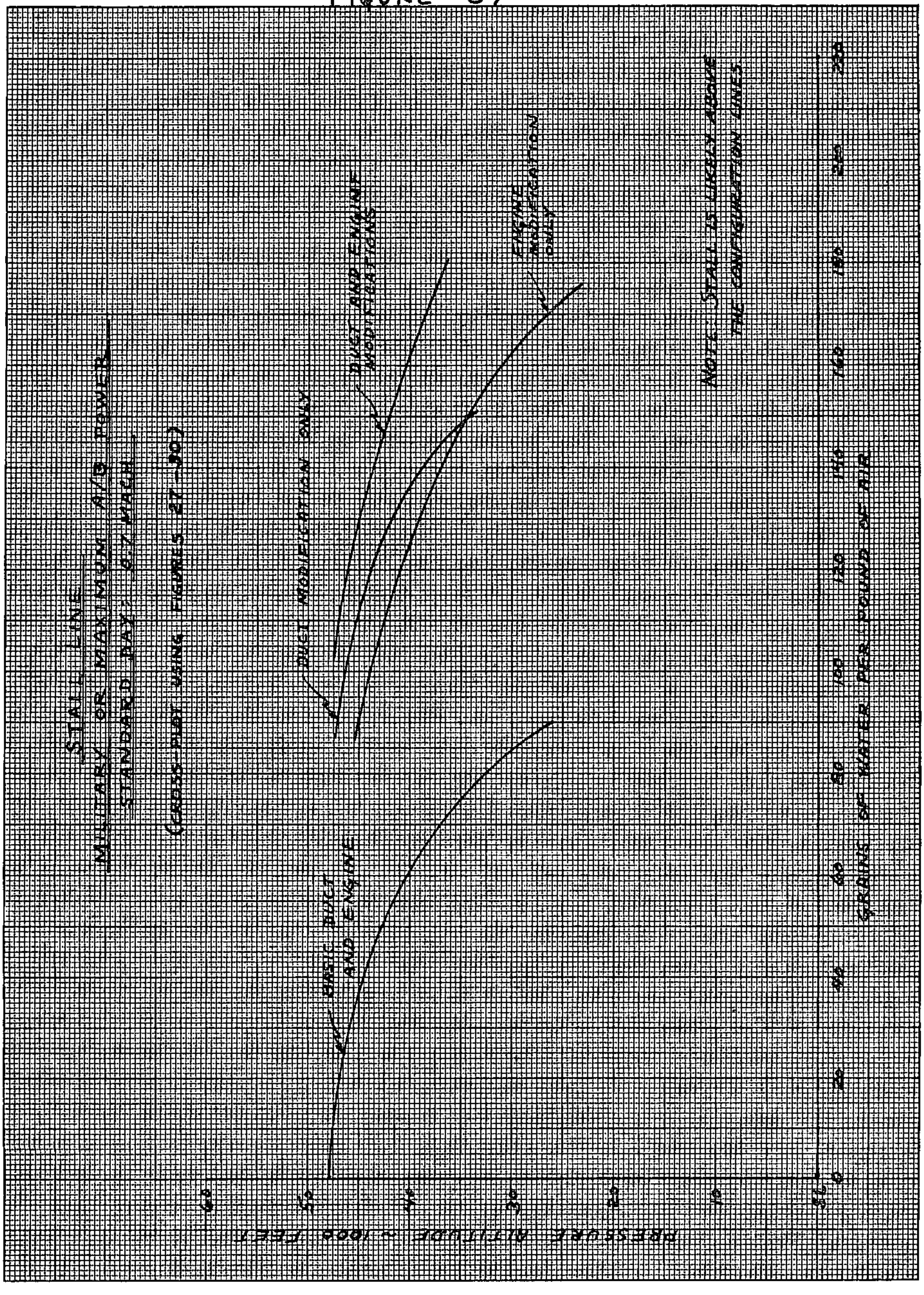


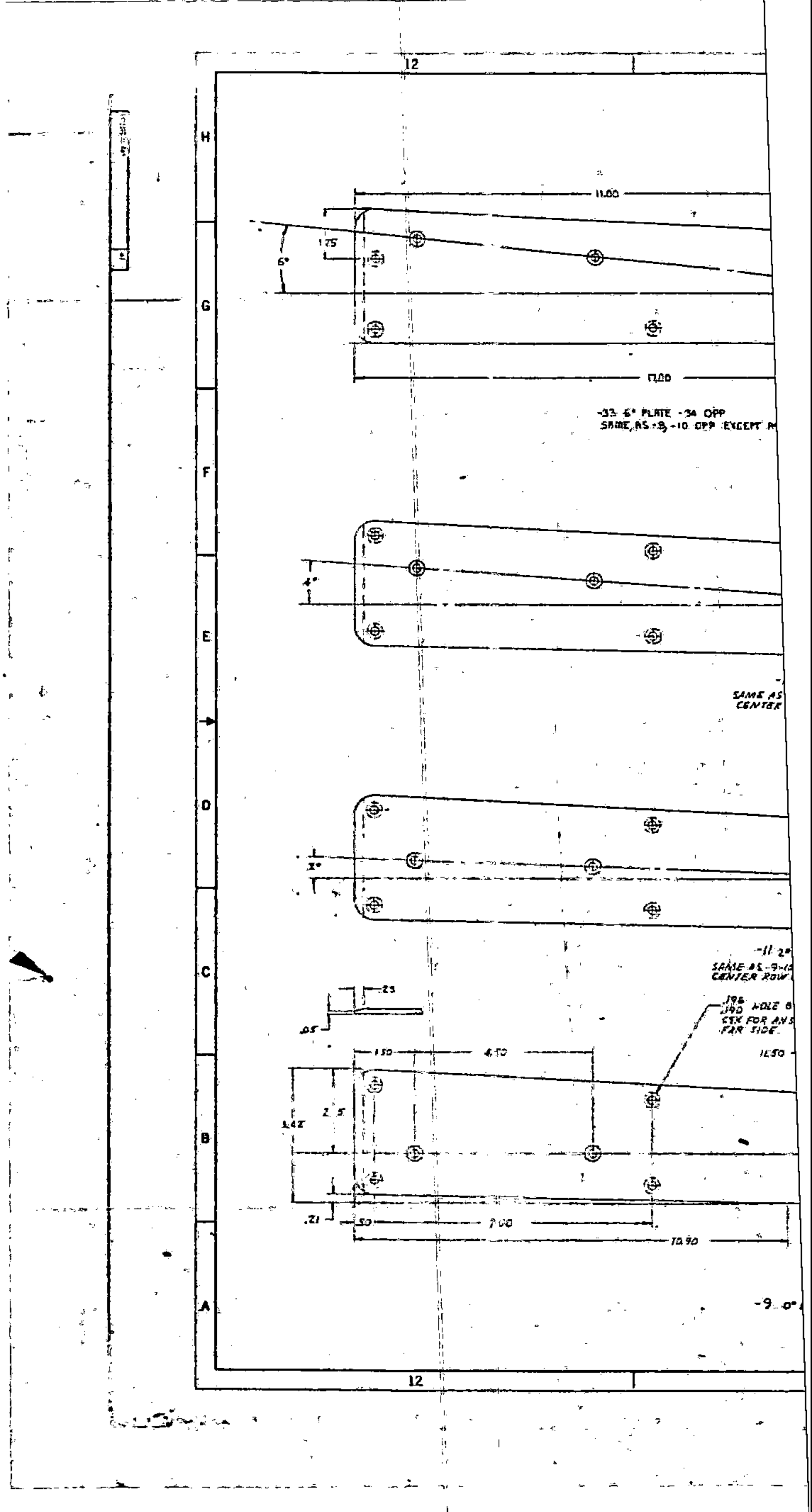
FIGURE 38

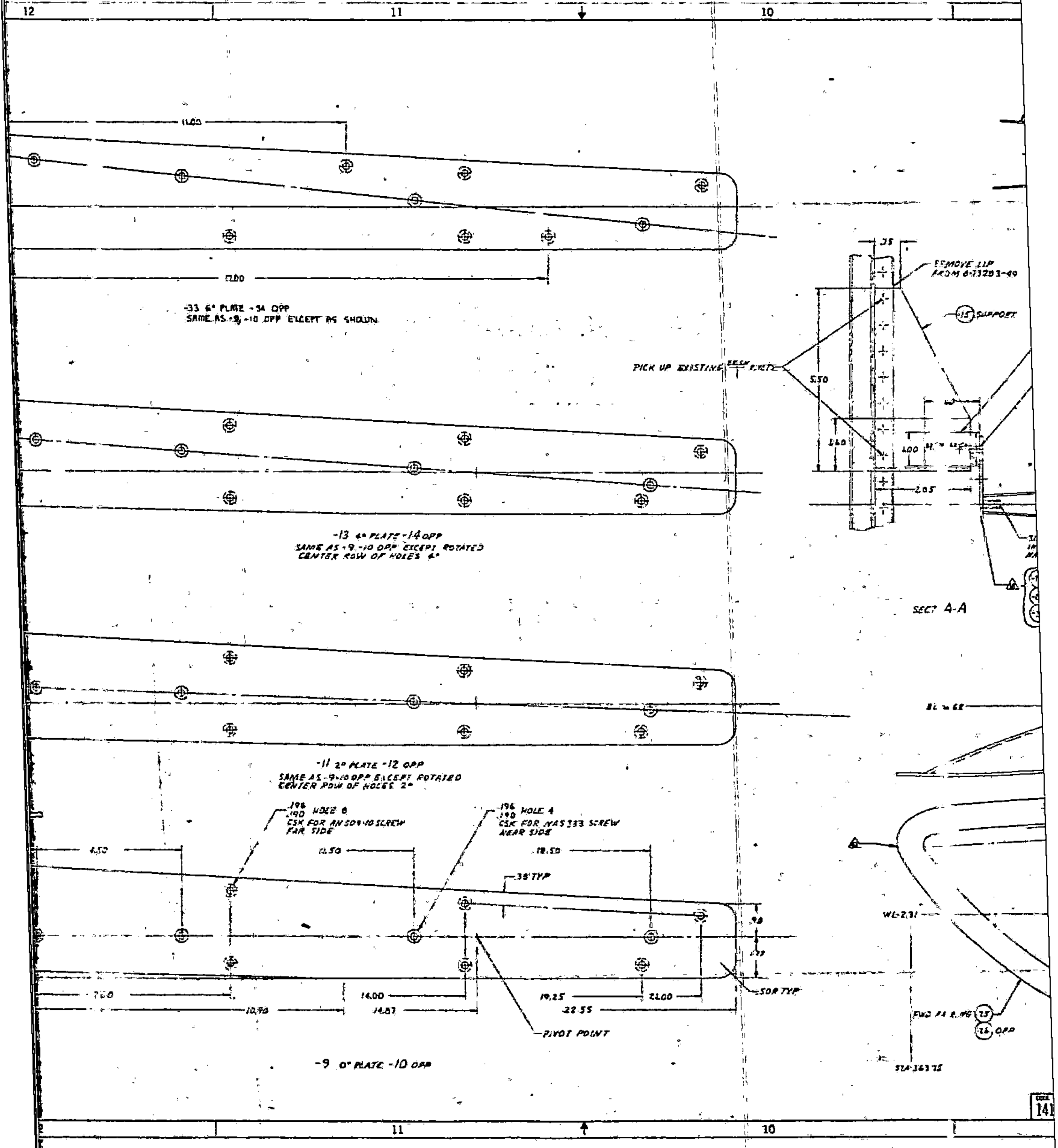
N_2 STALL LIMITS vs TEMPERATURE
MILITARY POWER - CLEAR AIR

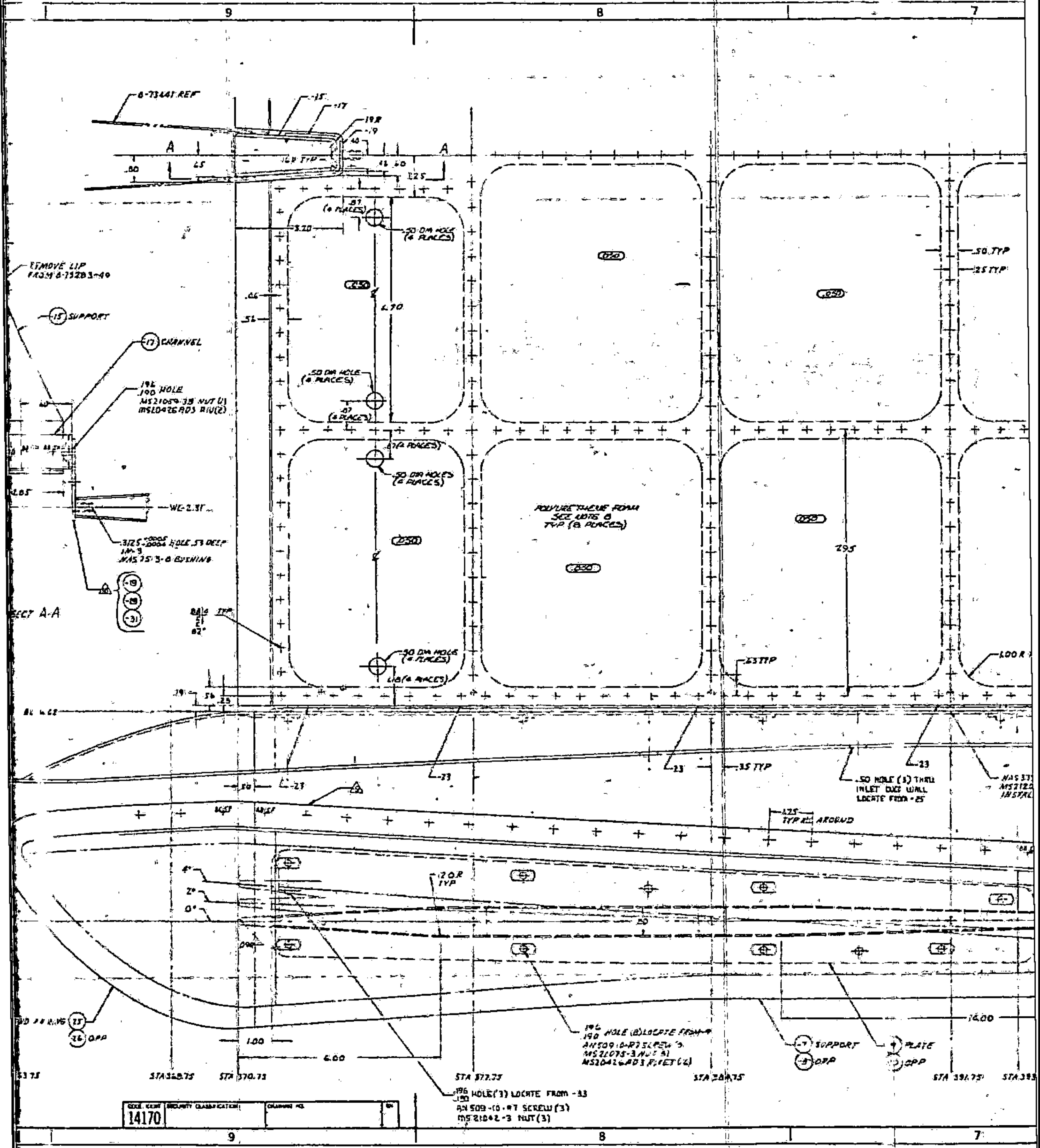
NOTE: STALL IS LIKELY ABOVE OR TO THE LEFT OF THE CONSTANT ΔT FROM STANDARD LINES.

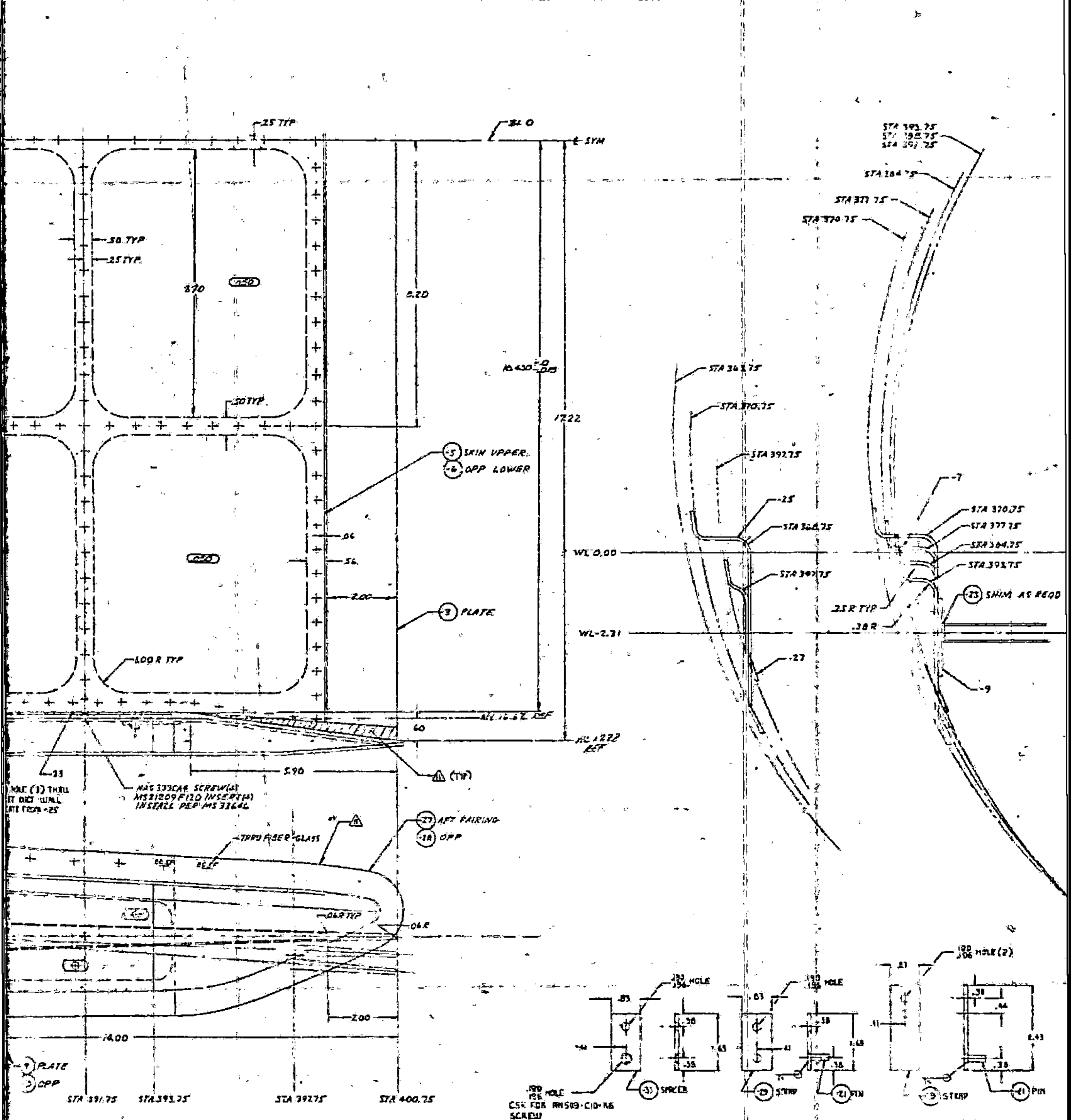
FIGURE 39


STALL LINE
MILITARY OR MAXIMUM A/B POWER
STANDARD DAY; 0.7 MACH
(CROSS-PLOT USING FIGURES 27-30)
BASIC DUCT AND ENGINE
DUCT MODIFICATION ONLY
DUCT AND ENGINE MODIFICATIONS
ENGINE MODIFICATION ONLY
NOTE: STALL IS LIKELY ABOVE THE CONFIGURATION LINES.
PRESSURE ALTITUDE ~ 1000 FEET
60
50
40
30
20
10
SL
GRAINS OF WATER PER POUND OF AIR
0
20
40
60
80
100
120
140
160
180
200
220


12
H
G
F
E
D
C
B
A
11.00
1.25
6°
11.00
-33 6° PLATE -34 OPP
SAME AS -9, -10 OPP EXCEPT
4°
SAME AS
CENTER
2°
-11 2°
SAME AS -9-1
CENTER ROW
.25
.05
.196
.190 HOLE
CSK FOR AN
FAR SIDE.
11.50
1.50
4.50
3.42
2.25
.21
.50
7.00
10.90
-9 0°
12


12
11
10
11.00
11.00
-33 6° PLATE -34 OPP
SAME AS -9, -10 OPP EXCEPT AS SHOWN
-13 4° PLATE -14 OPP
SAME AS -9, -10 OPP EXCEPT ROTATED
CENTER ROW OF HOLES 4°
-11 2° PLATE -12 OPP
SAME AS -9-10 OPP EXCEPT ROTATED
CENTER ROW OF HOLES 2°
.196
.190 HOLE 6
CSK FOR AN 509-10 SCREW
FAR SIDE
.196
.190 HOLE 4
CSK FOR NAS 333 SCREW
NEAR SIDE
4.50
11.50
18.50
.38 TYP
.98
7.00
10.90
14.00
14.87
19.25
22.55
22.00
.50R TYP
PIVOT POINT
-9 0° PLATE -10 OPP
.35
REMOVE LIP
FROM 6-73283-49
15 SUPPORT
PICK UP EXISTING RIVETS
5.50
1.60
.400
2.05
SECT A-A
BL 68
WL 2.31
FWD FR RING 23
24 OPP
STA 363.75
141
11
10


9
8
7
6-73441 REF
-15
-17
-19 R
-19
A
A
.45
.80
.12 R TYP
.40
.45 .60
1.25
REMOVE LIP FROM 6-73283-49
-15 SUPPORT
-17 CHANNEL
.196 .190 HOLE
MS21059-3B NUT (1)
MS20426AD3 RIV (2)
.40
.65
WL-2.37
.3125 HOLE .53 DEEP
MS 75 3-6 BUSHING
-19
-29
-31
SECT A-A
.87 (4 PLACES)
.50 DIA HOLE (4 PLACES)
3.20
.06
.56
.250
4.70
.50 DIA HOLE (4 PLACES)
.87 (4 PLACES)
.87 (4 PLACES)
.50 DIA HOLES (4 PLACES)
.250
.50 DIA HOLE (4 PLACES)
1.18 (4 PLACES)
.050
.050
.050
.050
.050
.050
POLYURETHANE FOAM SEE NOTE 8 TYP (8 PLACES)
.50 TYP
.25 TYP
7.95
.55 TYP
1.00 R
.39
.56
.25
-23
-23
-23
-23
.35 TYP
.50 HOLE (3) THRU INLET DUCT WALL LOCATE FROM -25
.58
-27
.125 TYP ALL AROUND
4°
2°
0°
.12 0 R TYP
.50
.090
.196 .190 HOLE (8) LOCATE FROM -7
AN509-10-R7 SCREW (5)
MS21075-3 NUT (5)
MS20426AD3 RIVET (2)
-7 SUPPORT
-8 OPP
-5 PLATE
-5 OPP
14.00
-25
-26 OPP
1.00
6.00
STA 368.75
STA 370.75
STA 377.75
STA 384.75
STA 391.75
.196 .190 HOLE (3) LOCATE FROM -33
AN509-10-R7 SCREW (3)
MS21042-3 NUT (3)
CODE IDENT
14170
SECURITY CLASSIFICATION
DRAWING NO.
SH
9
8
7



CODE IDENT
14170

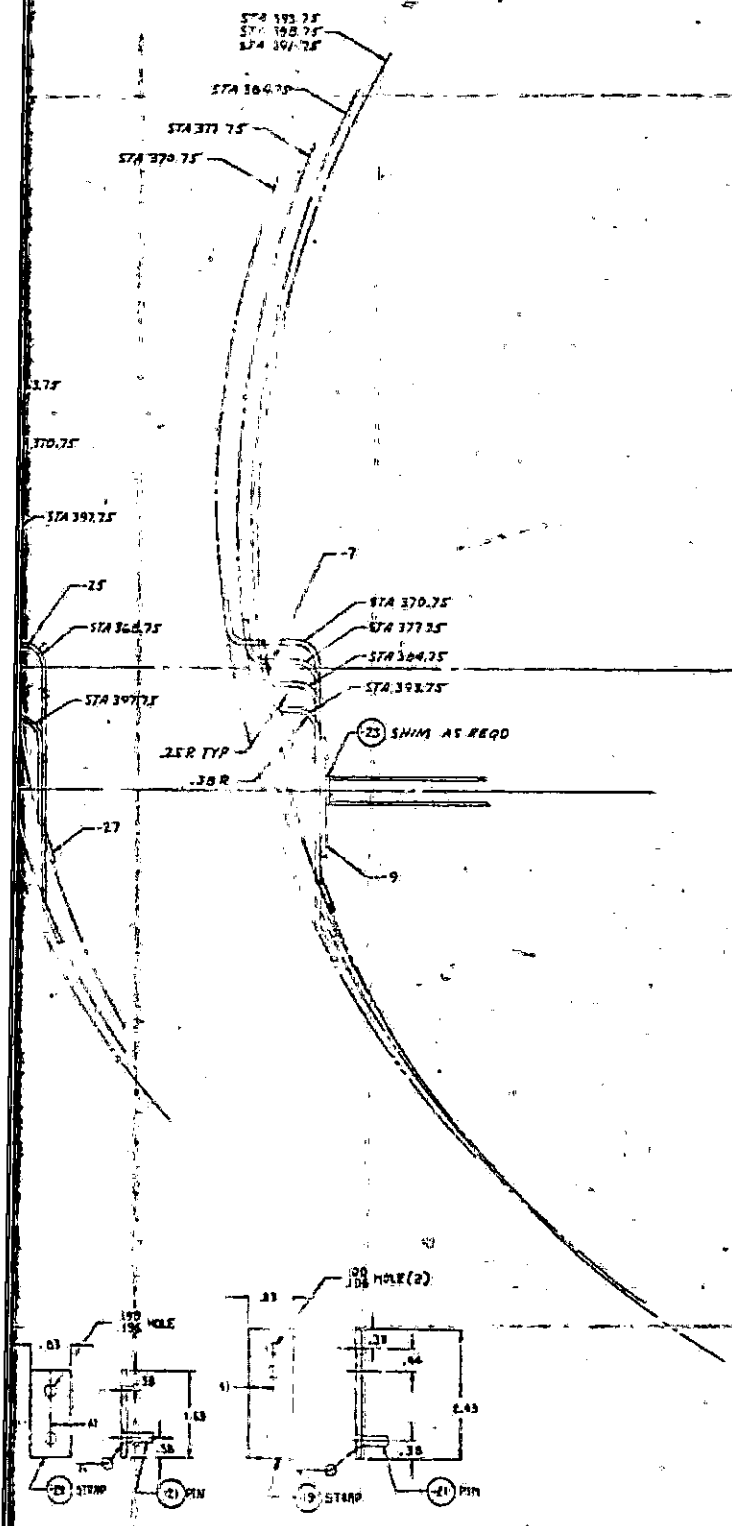
SECURITY CLASSIFICATION

DRAWING NO.
FT11-68024

SH
1

5	4	3

ZONE	LTR	DESCRIPTION	DATE	APPROVED



△11 TRIM THIS AREA ON INST'L TO CLEAR ADJACENT SUPPORT AND INLET DUCT STR

△10 ANGLE OF ATTACK (∝) INST'L
1) 0° USE: -9 AND -10 PLATES; -19 STRAP SE -17 CHANNEL THRU THE UPPER HOLE WITH
2) 2° USE: -11 AND -12 PLATES; -19 STRAP -17 CHANNEL THRU THE LOWER HOLE WITH
3) 4° USE: -13 AND -14 PLATES; -29 STRAP SE -17 CHANNEL WITH AN3H4A SCREW
4) 6° USE: -33 AND -34 PLATES; -31 SPACER TO -17 CHANNEL WITH AN509-C10 R8 -29 STRAP SECURED TO -31 SPACER BOLT AND MS21042-3 NUT

△9 APPLY EC1293 TO FAYING SURFACES

△8 FILL WITH 4-10 LB DENSITY POLYURETHA
70% MIN. CAVITY FILL ACCEPTABLE.
FILL EACH AREA THRU ONE FILL HOLE
TWO (2) POURS OF FOAM ACCEPTABLE
FLUSH FOAM AT FILL HOLES AND SEA
EPOXY RESIN (EPON 828-100 PART
HARDNER-12 PARTS; AND CAB-O-S
(REF SD4 PROG. OFFICE MEMO

7. ALTERNATE MATERIAL 7075-T6
6. ALTERNATE MATERIAL NAS[illegible]33C4
5. ALTERNATE MATERIAL 7075-T651 QQ-A-2
4. FABRICATE -25, 26, 27 & -28 FROM EPOX GLASS FABRIC (8 LAYERS) CONFORM
3. AFTER WELDING -21 PIN TO -19 & 29 STR
2. HEAT TREAT AND AGE TO T62
1. ALODINE FINISH ONLY

NOTES-

CODE IDENT	SECURITY CLASSIFICATION	DRAWING NO	SH
14170		FT11-68024	1

5	4	3

SYM	DESCRIPTION	DATE	APPROVED
A	INCORPORATED [illegible] CHG'S	[illegible]	
B	INCORPORATED FLIGHT TEST CHG'S	[illegible]	

QTY	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	STOCK SIZE, MATERIAL SPECIFICATION, NOTE TEXT
	-34	6" PLATE	5	.125 X 3.5 X 22.6 2024-T81 QQ-A-250/5 SEE NOTE 2
	-33	6" PLATE	5	.125 X 3.5 X 22.6, 2024-T81 QQ-A-250/5 SEE NOTE 2
4	MS21042-3	NUT		
1	AN3H8A	BOLT		
1	AN509-C10-R6	SCREW		
1	-31	SPACER	5	.020 X 1.0 X 1.7, 4130 STEEL, MIL-S-6758
AR	828	EPON		
AR	951	HARDNER		
AR		CAB-O-SIL		
AR	HA7303	FOAM COMPONENT A		URETHANE FOAM-HASTINGS PLASTICS,
AR	HA 7303	FOAM COMPONENT B		1704 COLORADO AVE. SANTA MONICA, CAL.
AR	MS20426-4	RIVET		
AR	MS20426AD5	RIVET		
AR	MS20426AD4	RIVET		
AR	MS20470AD3	RIVET		
8	MS21209F1-20	INSERT		
[illegible]	MS21075-3	NUT		
	MS21059-3A	NUT		
19	AN509-3R7	SCREW		
1	AN3H4A	BOLT		
8	NAS333C4	SCREW		SEE NOTE 6
AR	EC1293	SEALANT		
1	-29	STRAP	8	.050 X 1.0 X 1.7 4130 STEEL MIL-S-6758, SEE NOTE 3
1	-28	FAIRING AFT	C	SEE NOTE 4
1	-27	FAIRING AFT	C	SEE NOTE 4
1	-26	FAIRING FWD	A	SEE NOTE 4
1	-25	FAIRING FWD	A	SEE NOTE 4
AR	-23	SHIM	D	.032 X .60 X .70, 2024-T81 QQ-A-250/5
2	-21	PIN	E	[illegible] X .5 X .210, 4130 STEEL MIL-S-6758, SEE NOTE 3
1	-19	STRAP	B	.080 X [illegible] X 2.5, 4130 STEEL, MIL-S-6758, SEE NOTE 3
1	-17	CHANNEL	F	.090 X [illegible] X 4.8, 2024-0, QQ-A-250/4 SEE NOTE 2
1	-15	SUPPORT	E	.070 X 1.0 X 3.3, 2024-0 QQ-A-250/5, SEE NOTE 2
1	-14	4" PLATE	F	.125 X 3.5 X 22.6, 2024-T81, QQ-A-250/5
1	-13	4" PLATE	E	
1	-12	2" PLATE	E	
1	-11	2" PLATE	E	
1	-10	0" PLATE	A	
1	-9	0" PLATE	A	.125 X 3.5 X 22.6, 2024-T81, QQ-A-250/5
1	-8	SUPPORT	A	.080 X .90 X .250 2024-0 QQ-A-250/5 SEE NOTE 2
1	-7	SUPPORT	A	.080 X .90 X 21.0 2024-0 QQ-A-250/5 SEE NOTE 2
1	-6	SKIN	B	.050 X 8.0 X 35.0, [illegible] SEE NOTE 7
1	-5	SKIN	B	.050 X 8.0 X 35.0 2024-T81, QQ-A-250/5 SEE NOTE 8
1	-3	PLATE	D	.063 X 3.0 X 35.0, 2034-T81 QQ-A-250/4 SEE NOTE 5
—	-1	VANE INST'L		
REQUIREMENTS PER ASSEMBLY INSTALLATION	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	STOCK SIZE, MATERIAL SPECIFICATION, TYPE (TEMPER, HDCS, HEAT TREAT) NOTE TEXT

PARTS LIST

TRIM THIS AREA ON INST'L TO CLEAR ADJACENT SUPPORT AND INLET DUCT STRUCTURE

ANGLE OF ATTACK (∝) INST'L

1) 0° USES -9 AND -10 PLATES; -19 STRAP SECURED TO -17 CHANNEL THRU THE UPPER HOLE WITH AN509 SCREW

2) 2° USES -11 AND -12 PLATES, -19 STRAP SECURED TO -17 CHANNEL THRU THE LOWER HOLE WITH AN509 SCREW

3) 4° USES -13 AND -14 PLATES; -29 STRAP SECURED TO -17 CHANNEL WITH AN509 SCREW

4) 6° USES -33 AND -34 PLATES; -31 SPACER SECURED TO -17 CHANNEL WITH AN509-C10 R6 SCREW; -29 STRAP SECURED TO -31 SPACER USING AN3H8A BOLT AND MS21042-3 NUT

APPLY EC1293 TO FAYING SURFACES DURING INSTALLATION

FILL WITH 4-10 LB. DENSITY POLYURETHANE FOAM. 70% MIN CAVITY FILL ACCEPTABLE. FILL EACH AREA THRU ONE FILL HOLE. TWO (2) HOURS OF FOAM ACCEPTABLE. FLUSH FOAM AT FILL HOLES AND SEAL WITH EPOXY RESIN (EPON 828-100 PARTS, 951 HARDNER 12 PARTS, AND CAB-O-SIL AS REQD) (REF 304 PROG. OFFICE MEMO #7)

ALTERNATE MATERIAL 7075-T6

ALTERNATE MATERIAL AMS6433C4

ALTERNATE MATERIAL 7075-T651 QQ-A-250/12

FABRICATE -25, -26, 27 & -28 FROM EPON PRE PREG 181 GLASS FABRIC (5 LAYERS) CONFORMING TO MIL-R-9300A

AFTER WELDING -21 PIN TO -19 & 29 STRAP HEAT TREAT TO 125,000-150,000

HEAT TREAT AND AGE TO T62

ANODIZE FINISH ONLY

NOTES-

UNLESS OTHERWISE SPECIFIED

ALL MACHINED SURFACES 125

SEE D-70924 FOR ABBREVIATION, CODE, AND SYMBOL REQUIREMENTS

SEE D-70922 FOR STANDARD MANUFACTURING AND INSTALLATION REQUIREMENTS

ASSOCIATED LIST REQUIRED WITH DOCUMENT SAME NUMBER PREFIXED UL ☐ FOR USAGE LIST

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES

TOLERANCES

.X	.XX	.XXX
±.1	±.03	±.010

ANGULAR PER D-70988

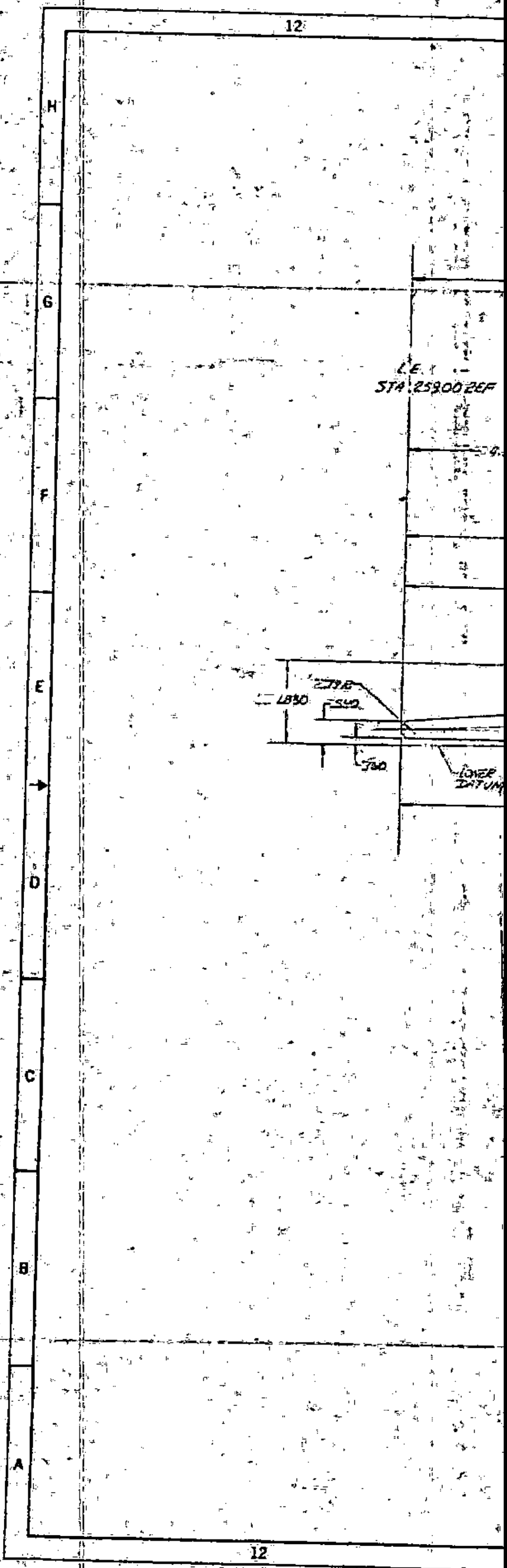
CONVAIR DIVISION OF GENERAL DYNAMICS
SAN DIEGO, CALIFORNIA

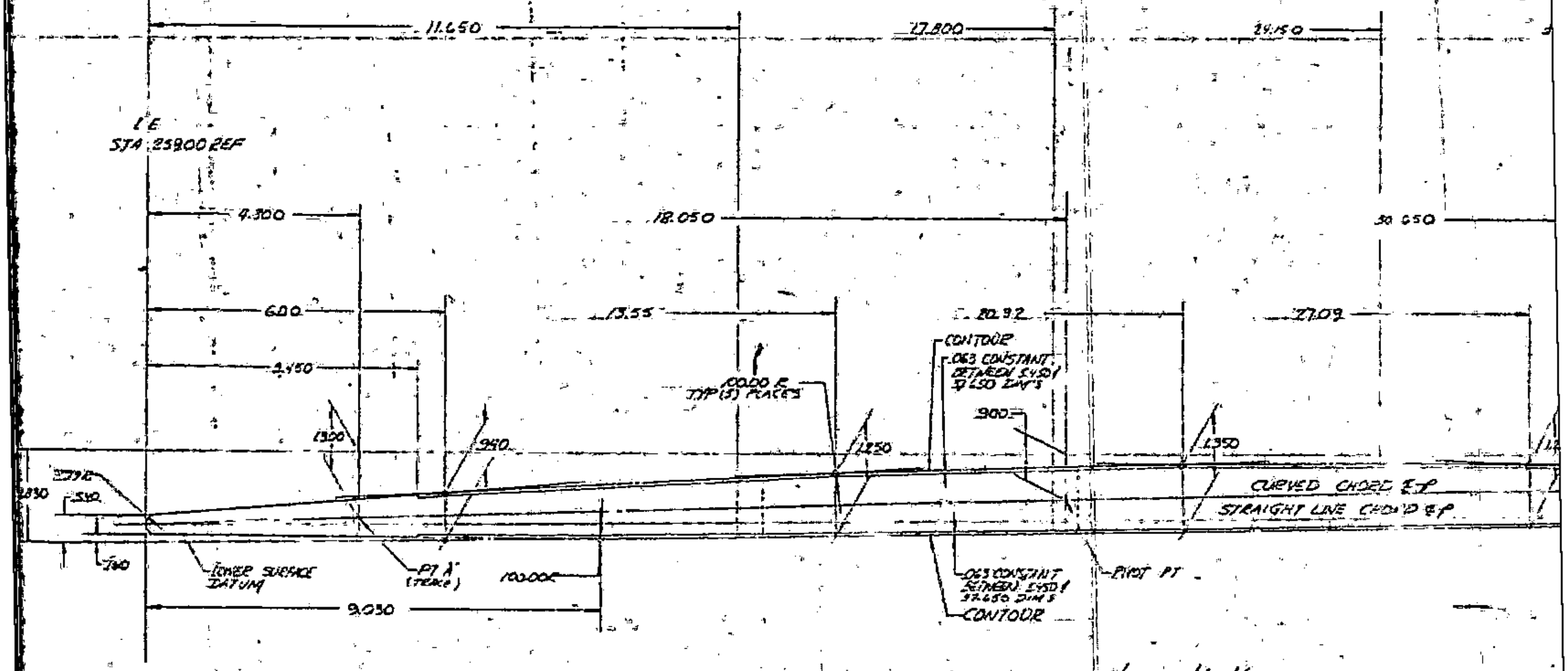
F102A INLET DUCT VANE-AFT HORIZONTAL, VERSION I

SIZE J CODE IDENT 14170 DRAWING NO. FT11-68024

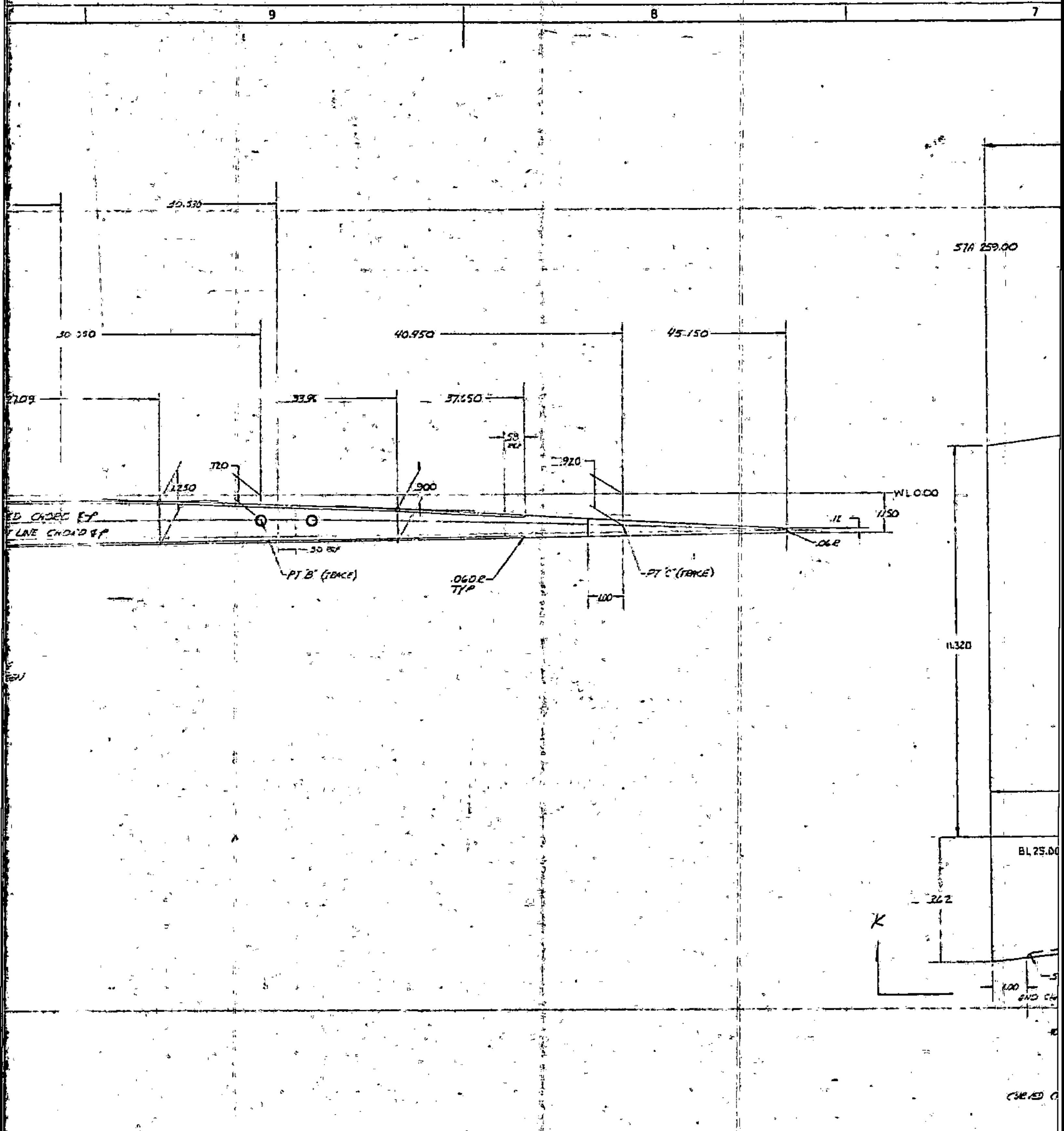
SCALE 1/1

SECURITY CLASSIFICATION

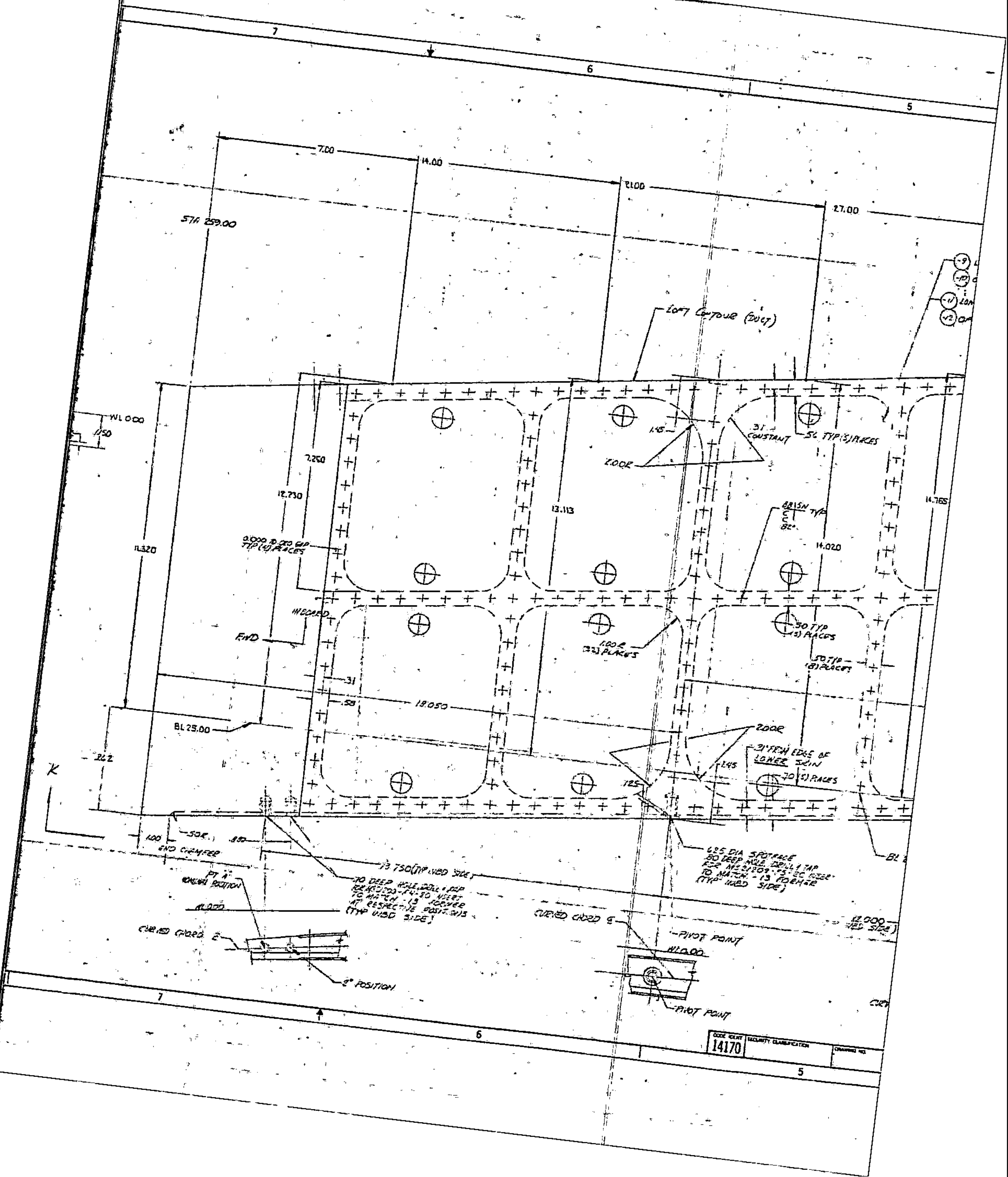

12
H
G
F
E
D
C
B
A
L.E.
STA 25900 REF
9.
.279 R
1.830
.540
.750
LOWER DATUM
12

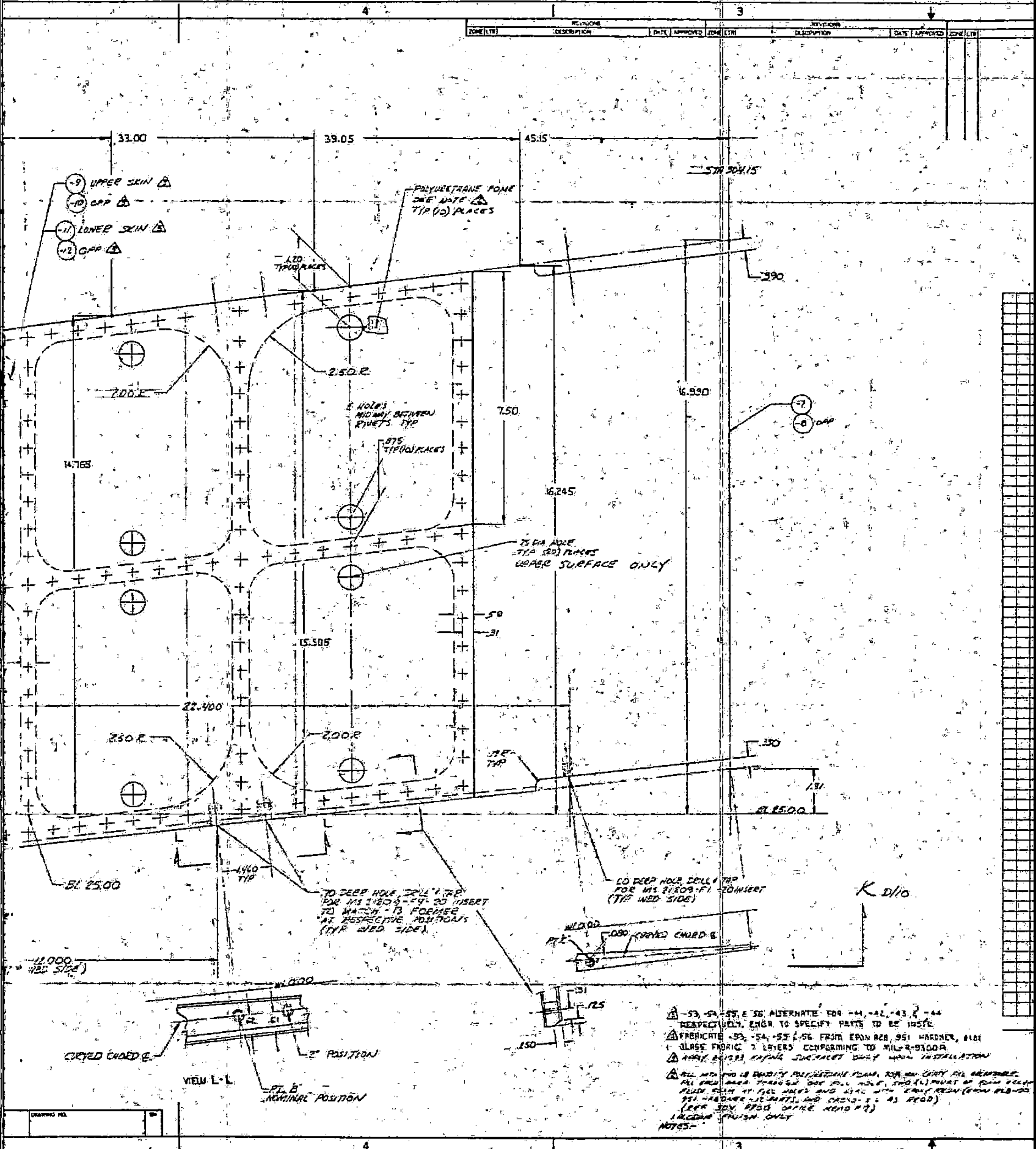


VIEW K-K
ALL CONTOUR CUTS ARE STRAIGHT LINE BETWEEN BREAK POINTS



CODE IDENT	SECURITY CLASSIFICATION	DRAWING NO.	SH
14170			


7
6
5
7.00
14.00
21.00
27.00
STA 259.00
-9
-10
-11
-12
LOFT CONTOUR (DUCT)
WL 0.00
.150
7.250
12.750
11.320
13.113
14.765
1.45
.31
CONSTANT
.56 TYP (5) PLACES
2.00R
.88 1SN TYP
82°
14.020
0.000 TO .020 GAP
TYP (4) PLACES
INBOARD
FWD
.50 TYP
(5) PLACES
1.00R
(32) PLACES
.50 TYP
(8) PLACES
.31
.58
19.050
BL 25.00
2.00R
.31" FROM EDGE OF
LOWER SKIN
2.45
.70 (5) PLACES
.125
2.62
K
1.00
.50R
.850
END CHAMFER
13.750 (TYP WEB SIDE)
.70 DEEP HOLE, DRILL & TAP
FOR MS21209-F4-30 INSERT
TO MATCH -13 FORMER
AT RESPECTIVE POSITIONS
(TYP WEB SIDE)
.625 DIA SPOTFACE
.50 DEEP HOLE, DRILL & TAP
FOR MS21209-F5-20 INSERT
TO MATCH -13 FORMER
(TYP WEB SIDE)
BL
12.000
(WEB SIDE)
PT "A"
NOMINAL POSITION
WL 0.00
CURVED CHORD E
2° POSITION
CURVED CHORD E
PIVOT POINT
WL 0.00
PIVOT POINT
CODE IDENT
14170
SECURITY CLASSIFICATION
DRAWING NO.



5 -53, -54, -55, & 56 ALTERNATE FOR -41, -42, -43, & -44 RESPECTIVELY. ENGR TO SPECIFY PARTS TO BE INSTL.

4 FABRICATE -53, -54, -55 & -56 FROM EPON 828, 951 HARDNER, #181 GLASS FABRIC 7 LAYERS CONFORMING TO MIL-R-9300A

3 APPLY EC1293 [illegible] SURFACES ONLY UPON INSTALLATION

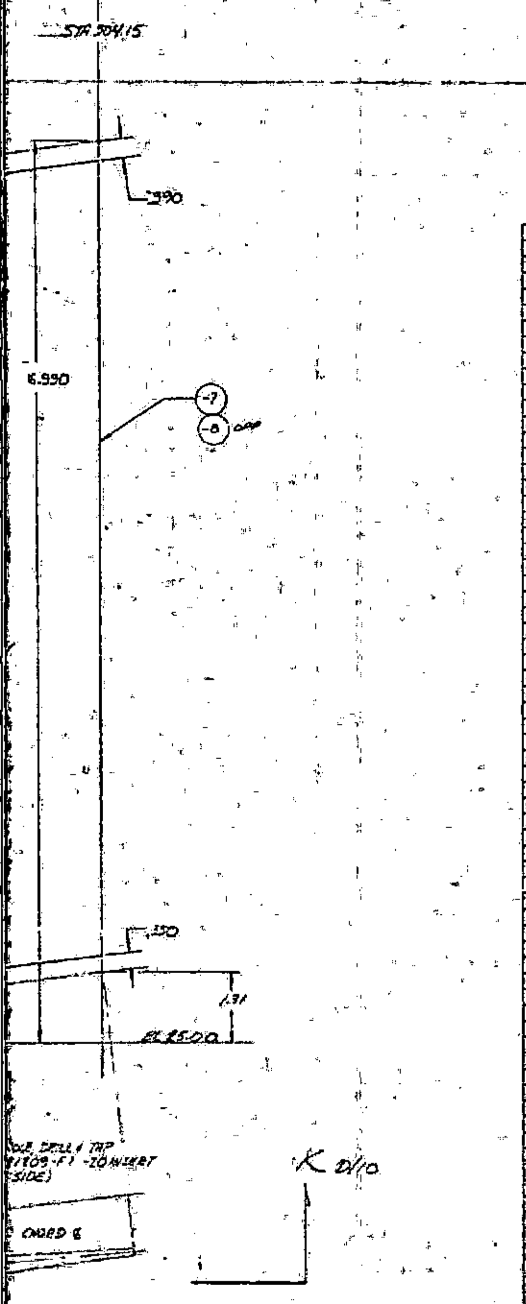
2 ALL [illegible] QUALITY POLYURETHANE FOAM, [illegible] FILL [illegible] THROUGH [illegible] HOLE, [illegible] (4) [illegible] FLUSH [illegible] AT FILL HOLES AND SEAL WITH EPOXY RESIN (EPON 828-[illegible] 951 HARDNER-[illegible] PARTS, AND [illegible] AS REQD) (SEE [illegible] MEMO #7)

1 ALODINE FINISH ONLY

NOTES:-

65

ZONE	LTR	REVISIONS DESCRIPTION	DATE	APPROVED
	A	FRONT TEST CAG'S		



-3	-1	UNIT OF MEAS	ITEM OR FIND NO.	CODE IDENT NO.	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	NO.	STOCK SIZE, MATERIAL, SPECIFICATION, TYPE (TEMPER, ETC.), HEAT TREAT, NOTE TEXT
AR	AR				MS20426AD4	RIVET			
AR	AR				MS20426AD5	RIVET			
AR	AR				MS20470AD4	RIVET			
AR	AR				MS20470AD5	RIVET			
AR	AR				MS20426DD6	RIVET			
1					-52	CLIP			.063 X 2.5 X 3.5 2024-0
	1				-51	CLIP			.063 X 2.5 X 3.5 2024-0
AR					828	EPON			
AR					951	HARDENER			
AR						CAB-O-SIL			
AR					AA-7303	FOAM COMPOUND A			LAMINATE FOAM-CASTINGS PLASTICS
AR					AA-7303	FOAM COMPOUND B			1104 COLORADO AVE. SANTA MONICA, CALIF
AR					EC1293	SEALANT			
	1				-53	FAIRING			SEE NOTE 4
3	3				AN960PD10L	WASHER			
1					-54	FAIRING			SEE NOTE 4
2	2				AN509-10R10	SCREW			
	1				-55	FAIRING			SEE NOTE 4
2	2				MS21042L4	NUT			
1					-56	FAIRING			SEE NOTE 4
2	2				HL[illegible]-8-[illegible]	HI-LOK			
1	1				NAS464-4-3	BOLT			ALTERNATE AN4-4A
AR	AR				181	FIBERGLASS FABRIC			
2	2				NAS625-6A	BOLT			ALTERNATE AN5[illegible]
2	2				NAS1004-4A	BOLT			
2	2				NAS1003-3A	BOLT			
2	2				MS[illegible]-5-40	BUSHING			
4	4				MS77-[illegible]	BUSHING			
2	2				MS21209-F1-15	INSERT			
9	8				MS21209-F1-20	|			
2	2				MS21209-F1-15	|			
2	2				MS21209-F1-20	INSERT			
1					-50	CLIP			.063 X 2.5 X 3.5 2024-0
	1				-49	|			.063 X 2.5 X 3.5 2024-0
1					-48	|			.063 X 2.5 X 2.5 2024-0
	1				-47	|			|
1					-46	|			|
	1				-45	CLIP			.063 X 2.5 X 2.5 2024-0
1					-44	LE WEDGE			[illegible] X 2.0 X 6.0 2024-T4 ALCLAD QQ-A-250/5
	1				-43	LE WEDGE			[illegible] X 2.0 X 6.0
1					-42	LE WEDGE			[illegible] X 2.0 X 6.0
	1				-41	LE WEDGE			[illegible] X 2.0 X 6.0
1					-40	LINK			[illegible]
	1				-39	LINK			[illegible]
1					-38	LINK			[illegible]
	1				-37	LINK			[illegible]
1	1				-35	PLATE			[illegible]
1	1				-33	PLATE			[illegible] X 2.0 2024-T351
4	4				-31	FILLER			.020 X 2.0 X 6.0 2024-T3
2	2				-29	PLATE			[illegible]
2	2				-27	PLATE			[illegible] X .25
1	1				-25	PLATE			.50 X [illegible]
1	1				-23	PLATE			[illegible] 2024-T351
1	1				-21	CHANNEL			[illegible] X 5.5 X 5.5 2024-0
1	1				-19	|			[illegible]
1	1				-17	|			[illegible]
1					-15	|			[illegible]
	1				-15	PANEL			[illegible]
1					-14	LONGERON			[illegible]
	1				-13	LONGERON			[illegible] 2024-0
1					-12	SKIN			[illegible] 2024-T81
	1				-11	SKIN			[illegible]
1					-10	SKIN			[illegible]
	1				-9	SKIN			[illegible] 2024-T[illegible]
1					-8	PLATE			[illegible] 2024-T351
	1				-7	PLATE			[illegible] 2024-T351 AL CLAD QQ-A-250/5
—	—				-2	[illegible] ASSY			
—	—				-1	[illegible] ASSY			

REQUIREMENTS PER ASSEMBLY INSTALLATION

PARTS LIST

NOTE SYMBOL: E–END ITEM, G–GENERAL NOTE, L–LIST, M–MATERIAL, R–REFERENCE DESIGNATION, S–STOCK NO., U–USAGE DATA

5. -53, -54, -55, & 56 ALTERNATE FOR -44, -42, -43 & -44 RESPECTIVELY. ENGR TO SPECIFY PARTS TO BE INSTL

4. FABRICATE -53, -54, -55 & 56 FROM EPON 828, 951 HARDENER, 181 GLASS FABRIC 7 LAYERS CONFORMING TO MIL-R-9300A

3. APPLY EC1293 FAYING SURFACES ONLY WHEN INSTALLATION

2. FILL WITH 2 LB DENSITY POLYURETHANE FOAM. 3 1/2 [illegible] [illegible] FILL [illegible]. [illegible] AREA THROUGH [illegible] FILL HOLE. TWO (2) PLIES OF [illegible] FLUSH [illegible] AT FILL HOLES [illegible] WITH EPOXY RESIN (EPON 828-100 PARTS, 951 HARDENER-10 PARTS, AND [illegible] 45 REQD) (REF 3DV PROD OFFICE MEMO #7)

1. ALODINE FINISH ONLY

0785-

UNLESS OTHERWISE SPECIFIED ALL MACHINED SURFACES ✓ 125

SEE D-70914 FOR ABBREVIATION, CODE AND SYMBOL REQUIREMENTS

SEE D-70900 FOR STANDARD MANUFACTURING AND INSTALLATION REQUIREMENTS

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES
TOLERANCES .X ±.1 .XX ±.03 .XXX ±.010
ANGULAR PER D-70900

ASSOCIATED LIST REQUIRED SEE DOCUMENT SAME NUMBER PREFIXED UL FOR USAGE LIST

CONTRACT NO.

CONVAIR DIVISION OF GENERAL DYNAMICS
SAN DIEGO, CALIFORNIA

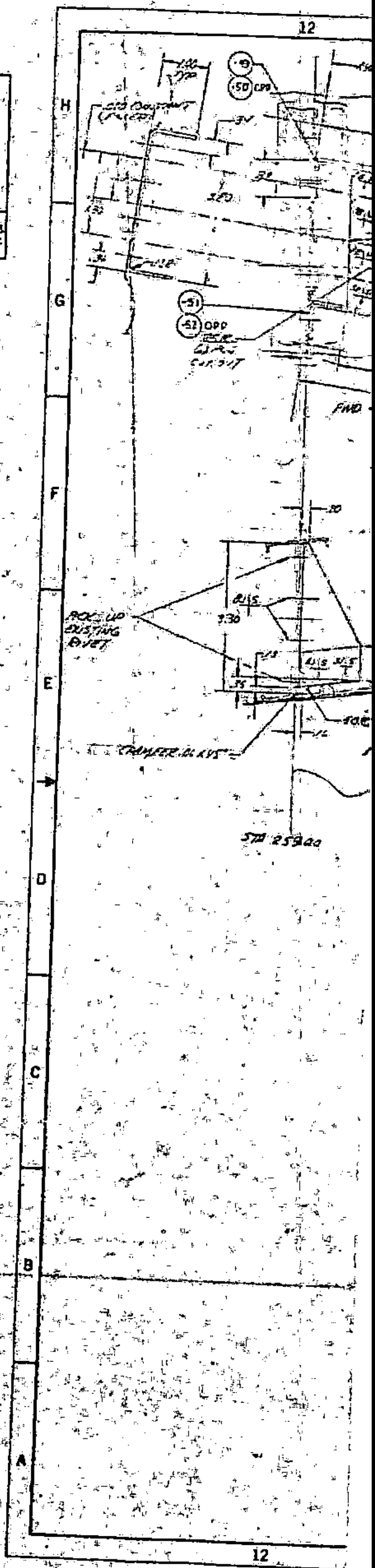
F-111A INLET DUCT VANE FWD HORIZONTAL, VERSION II

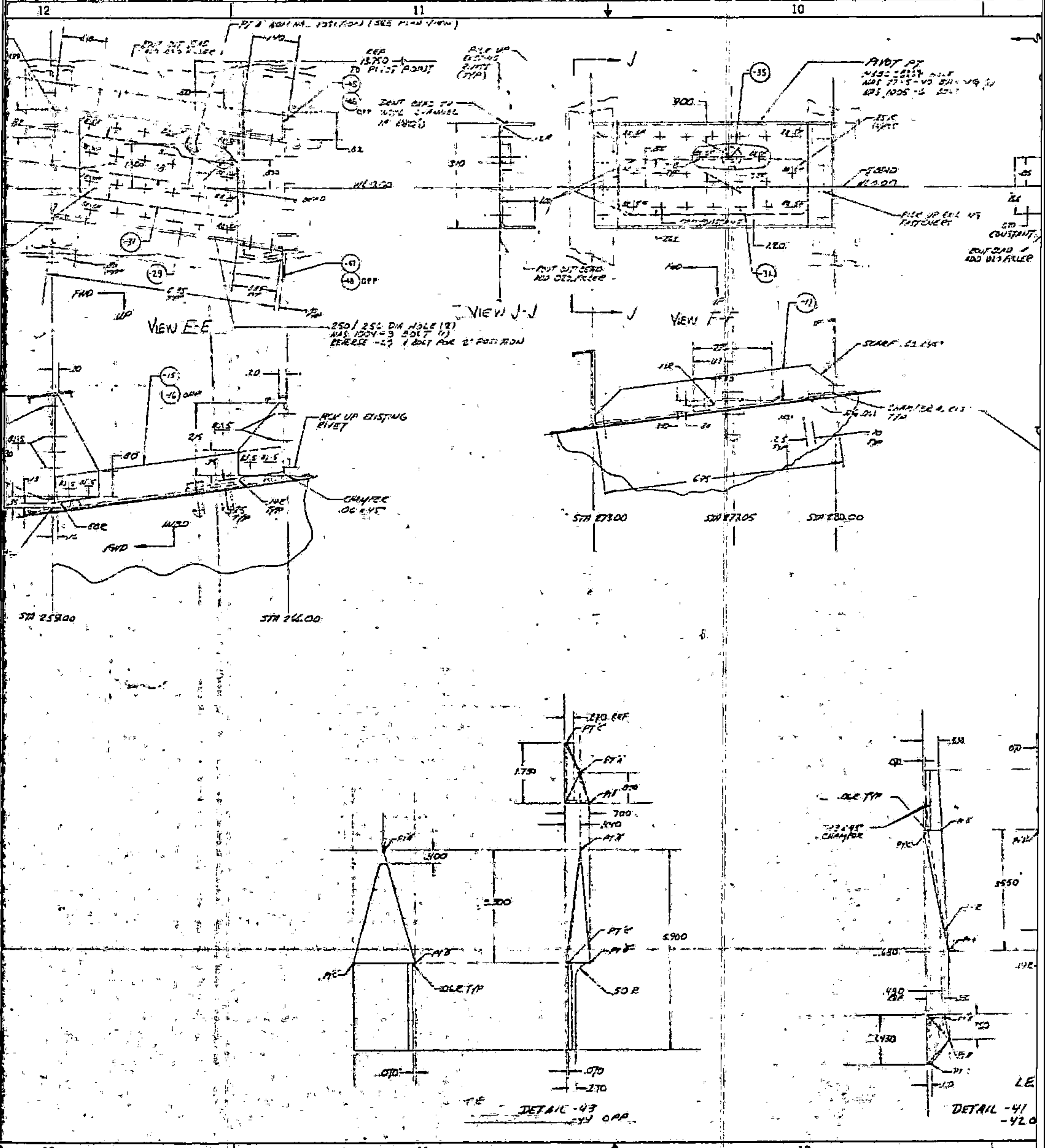
SIZE	CODE IDENT NO.	DRAWING NO.
J	14170	FT11-68025

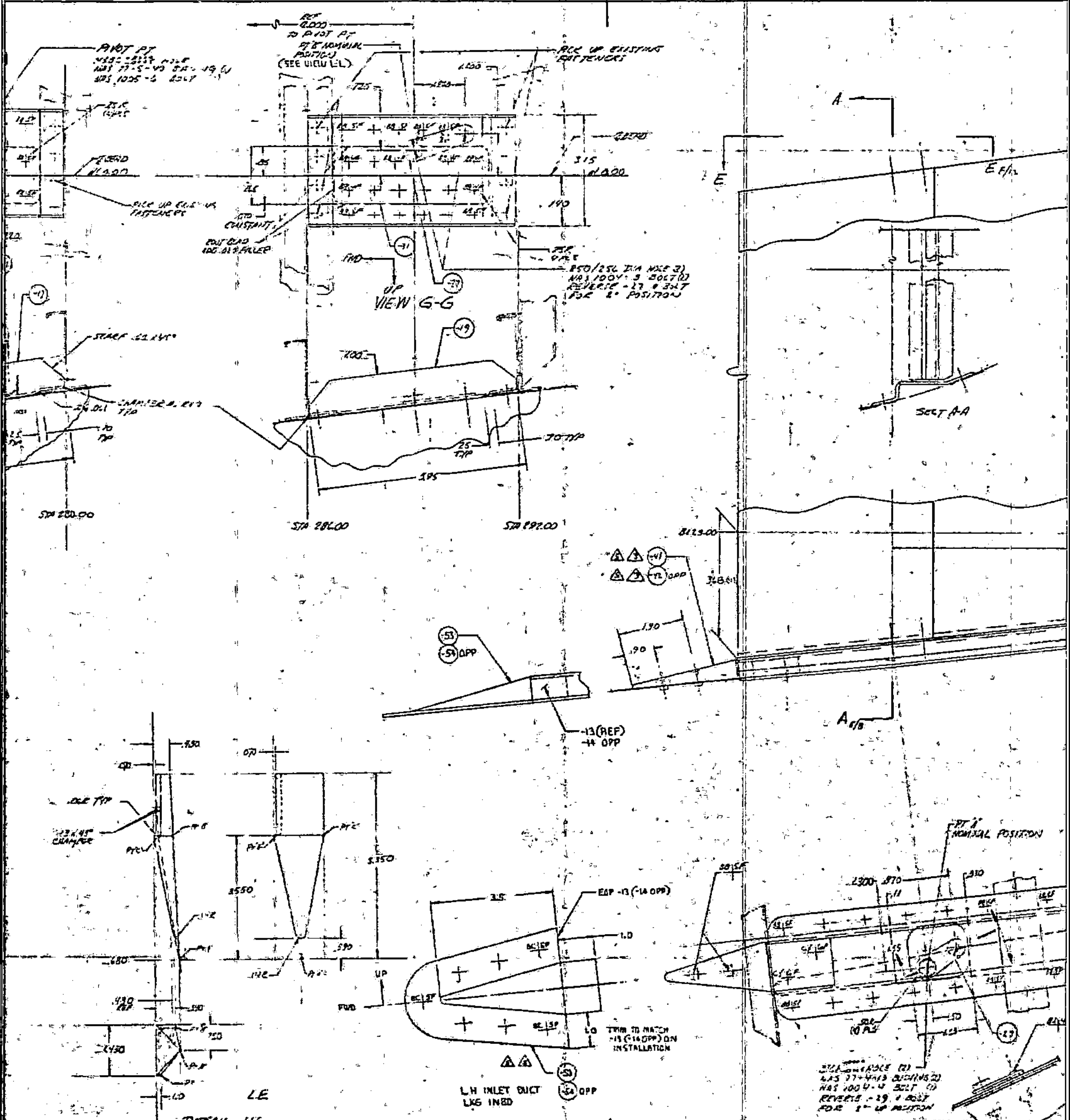
SCALE ½ RELEASE SHEET 1 OF 2

PACKAGE NO.

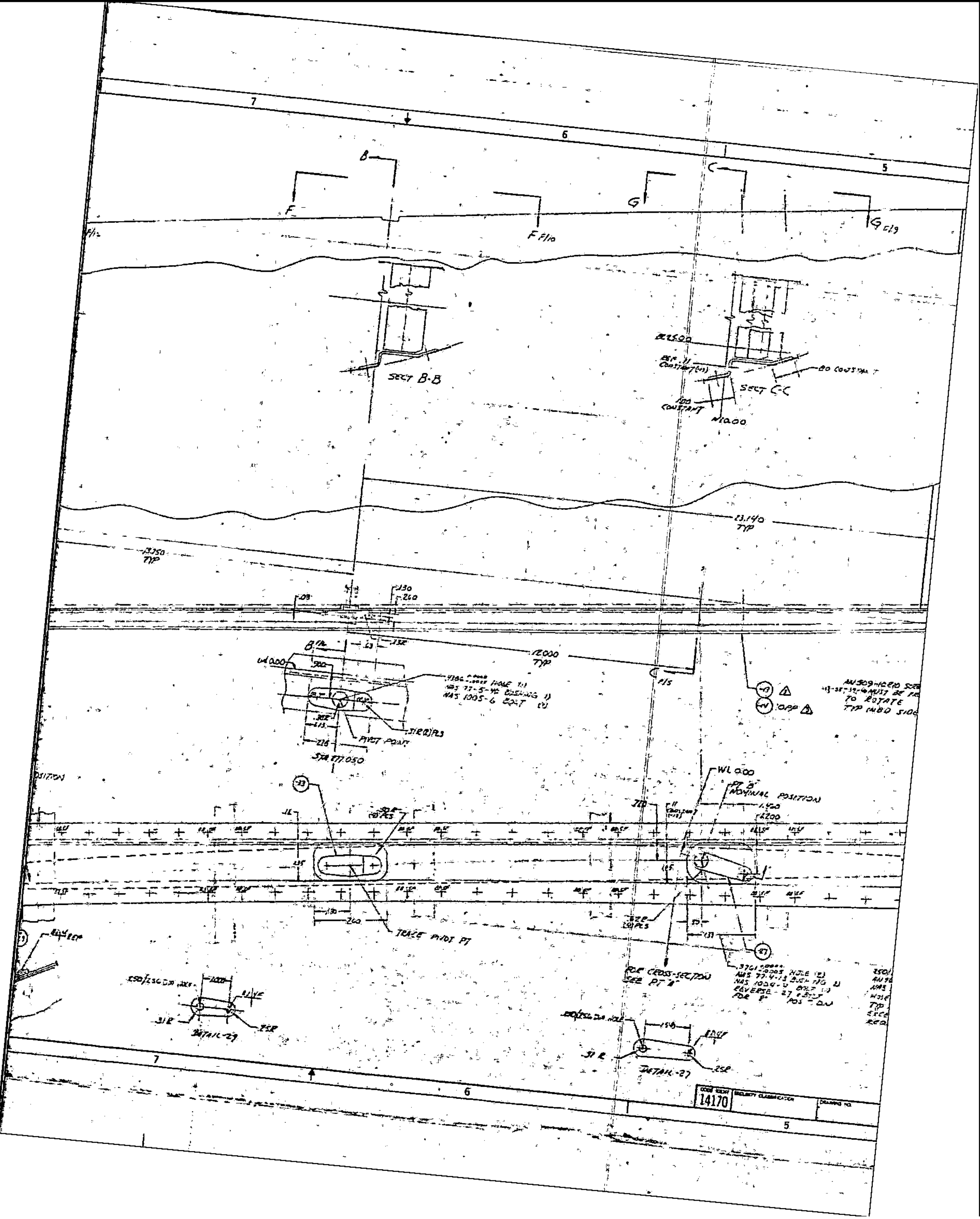
ROCKWELL INTERNATIONAL

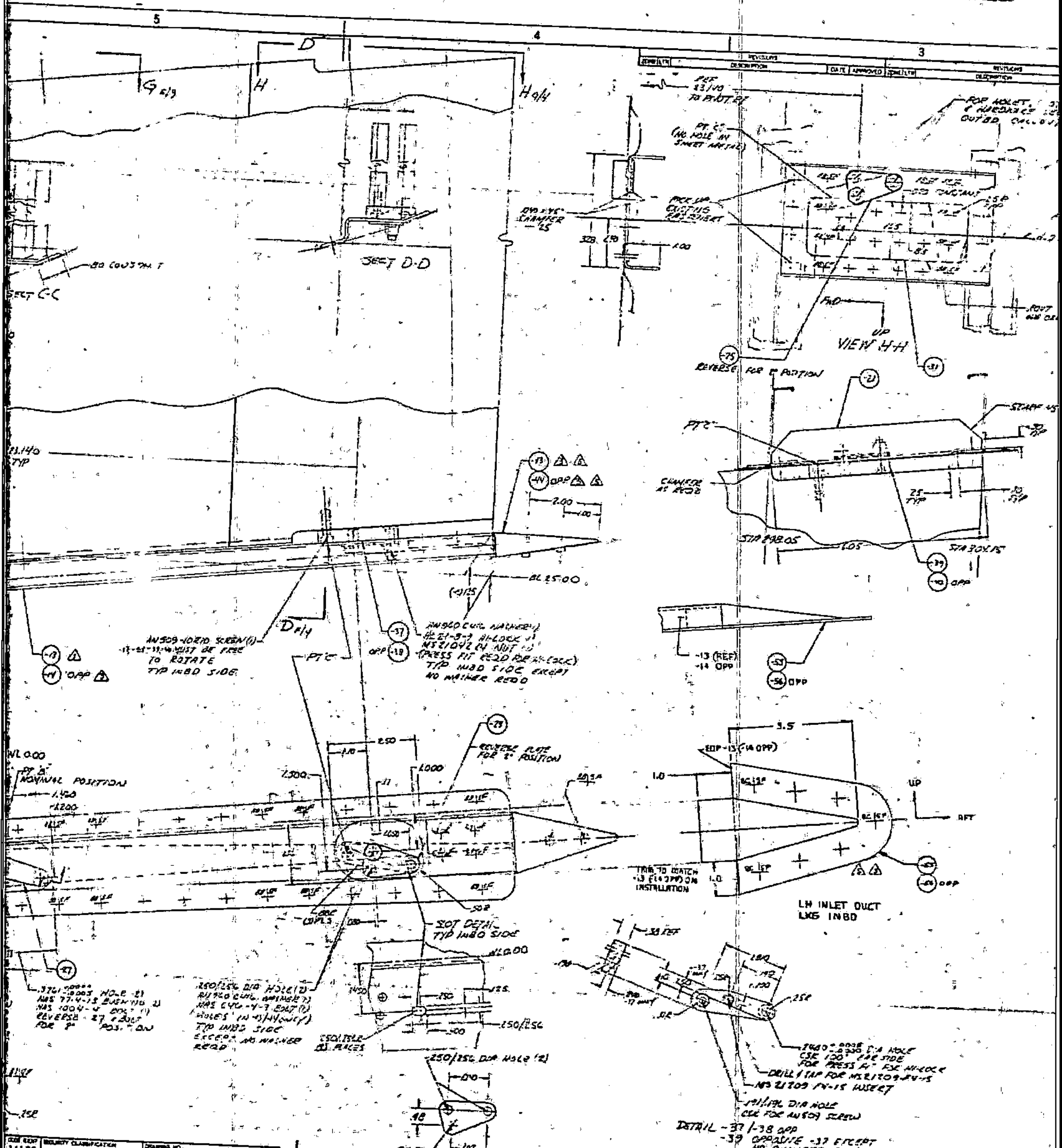



12
11
10
PT A NOMINAL POSITION (SEE PLAN VIEW)
.140
REF 13.750 TO PIVOT POINT
-45
-46 OPP
BENT ENDS TO NOTE CHANNEL
.82
.250
WL 3.00
.310
-31
-29
-47
-48 OPP
FWD
UP
VIEW E-E
.250/.252 DIA HOLE (2)
NAS 1004-3 BOLT (1)
J
VIEW J-J
.900
-35
PIVOT PT
NAS 1005-4 BOLT
.220
-34
PICK UP EXIST NAS FASTENERS
CONSTANT
VIEW F-F
-11
SCARF
CHAMFER
.25 TYP
.10 TYP
STA 273.00
STA 277.05
STA 280.00
-15
-16 OPP
.20
PICK UP EXISTING RIVET
.80
CHAMFER .06 X 45°
.60R
FWD
STA 259.00
STA 266.00
.270 REF
PT C
PT A
1.750
.700
.240
PT B
.400
2.300
5.900
PT C
OLE TYP
.50 R
.070
.270
DETAIL -43
-44 OPP
OLE TYP
CHAMFER
3.550
.680
.490
2.430
.750
LE
DETAIL -41
-42



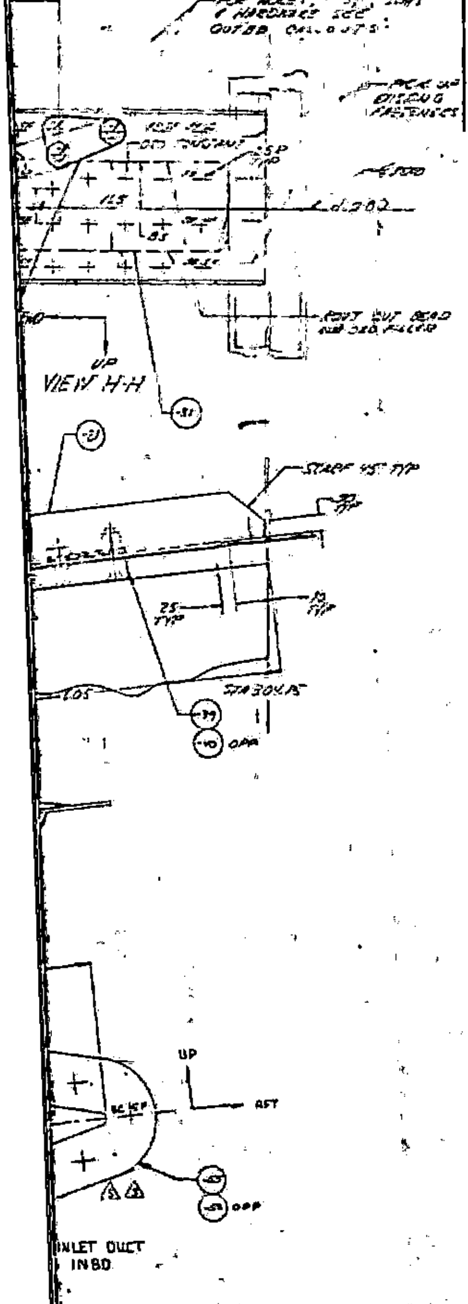
CODE IDENT	SECURITY CLASSIFICATION	DRAWING NO.	SH
14170			


7
6
5
B
C
F
G
F F/10
G C/9
SECT B-B
SECT C-C
BL25.00
WL 0.00
N10.00
23.140 TYP
13.250 TYP
12.000 TYP
.130
.260
STA 777.050
PIVOT POINT
.31R (2) PLS
NAS 77-5-45 BUSHING
NAS 1005-6 BOLT
F/S
AN509-10R10 SCREW
MUST BE FREE TO ROTATE
TYP INBD SIDE
OPP
PT "B" NOMINAL POSITION
TRACE PIVOT PT
FOR CROSS-SECTION SEE PT "B"
NAS 77-4-13 BUSHING
REVERSE -27 & BOLT FOR "B" POS
DETAIL-27
.31 R
.25R
EDGE
CODE IDENT
14170
SECURITY CLASSIFICATION
DRAWING NO.


5
4
3
D
G 5/3
H
H 4/4
REVISIONS
ZONE LTR DESCRIPTION DATE APPROVED
REF 33.40 TO PIVOT PT
PT. "C" (NO HOLE IN SHEET METAL)
FOR HOLES & HARDWARE SEE OUTBD CALLOUT
PICK UP EXISTING FAB INSERT
.090 X 45° CHAMFER .25
-80 COUNTERSINK
SECT C-C
SECT D-D
FWD
UP
VIEW H-H
-75
REVERSE FOR 2° POSITION
-31
-21
PT "C"
CHAMFER AS REQ'D
.25 TYP
STA 298.05
STA 301.75
-39
-40 OPP
21.140 TYP
-13
-14 OPP
2.00
1.00
BL 25.00
AN509-10R10 SCREW (1) -13-14-33-34 MUST BE FREE TO ROTATE TYP INBD SIDE
D 4/4
PT "C"
-37
OPP -38
AN960 C10L WASHER (1) HL21-5-7 HI-LOCK (1) MS21042 04 NUT (1) (PRESS FIT REQ'D FOR HI-LOCK) TYP INBD SIDE EXCEPT NO WASHER REQ'D
-13 (REF) -14 OPP
-53
-56 OPP
WL 0.00
PT "B" NOMINAL POSITION
1.500
.250
1.000
-29
REVERSE PLATE FOR 2° POSITION
EDP -13 (-14 OPP)
3.5
1.0
UP
AFT
TRIM TO MATCH -13 (-14 OPP) ON INSTALLATION
-55
-56 OPP
LH INLET DUCT LKG INBD
SLOT DETAIL TYP INBD SIDE
-27
.3761-.3764 HOLE (2) NAS 77-4-12 BUSHING (2) NAS 1004-4 BOLT (1) REVERSE -27 & BOLT FOR 2° POSITION
.250/.256 DIA HOLE (2) AN960 C10L WASHER (1) NAS 6404-7 BOLT (1) (HOLES IN -13/-14 ONLY) TYP INBD SIDE EXCEPT NO WASHER REQ'D
.250/.256
.300
CSOLTSLE (3) PLACES
.250/.256 DIA HOLE (2)
.48
.25R TYP
DETAIL -23
-25 SAME EXCEPT FOR GAGE & .090 X 45° CHAMFERS 3/U
.38 REF
.25R
.2460-.2530 DIA HOLE CSK 100° FAR SIDE FOR PRESS FIT FOR HI-LOCK
DRILL & TAP FOR MS21209-F4-15
MS 21209 F4-15 INSERT
.191/.196 DIA HOLE CSK FOR AN509 SCREW
DETAIL -37 / -38 OPP
-39 OPPOSITE -37 EXCEPT NO CHAMFER REQ'D
-40 OPP -39
CODE IDENT 14170
SECURITY CLASSIFICATION
DRAWING NO.
SH

66

ZONE	LTR	DESCRIPTION	DATE	APPROVED
	A	INCORPORATED FLIGHT TEST CHG'S		


FOR HOLES, STS, DIM'S & HARDWARE SEE OUTBD CALLOUTS
PICK UP DIMPLING FASTENERS
.202
VIEW H-H
UP
STAGE 45° TYP
.25 TYP
STA 304.75
OPP
UP
AFT
INLET DUCT INBD

...00-.0025 DIA HOLE
... 100° FAR SIDE
... PRESS FIT FOR HI-LOCK
... FOR MS21209-FV-15
...V-15 INSERT

...OLE
... SCREW

...37 EXCEPT
...R REQ'D

UNLESS OTHERWISE SPECIFIED
ALL MACHINED SURFACES √

SEE D-70914 FOR IDENTIFICATION, CODE, AND SYMBOL REQUIREMENTS

SEE D-70002 FOR STANDARD MANUFACTURING AND INSTALLATION REQUIREMENTS

ASSOCIATED LIST REQUIRED SEE DOCUMENT SAME NUMBER PREFIXED
PL ☐ FOR PARTS LIST
UL ☐ FOR USAGE LIST

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES
TOLERANCES .X ±.1 .XX ±.03 .XXX ±.010
ANGULAR PER D-70010

MATL
PROD. EFF.

CONTRACT NO.

CONVAIR DIVISION OF GENERAL DYNAMICS
SAN DIEGO, CALIFORNIA

F102A INLET DUCT VANE - FWD HORIZONTAL, VERSION II

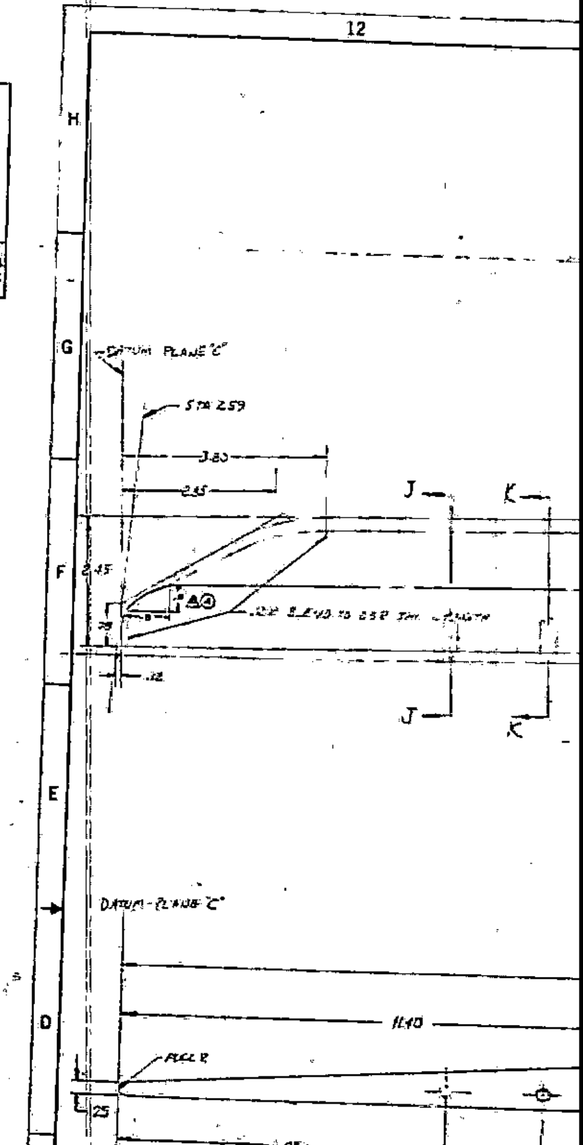
SIZE J | CODE IDENT 14170 | DRAWING NO. FT 11-68325

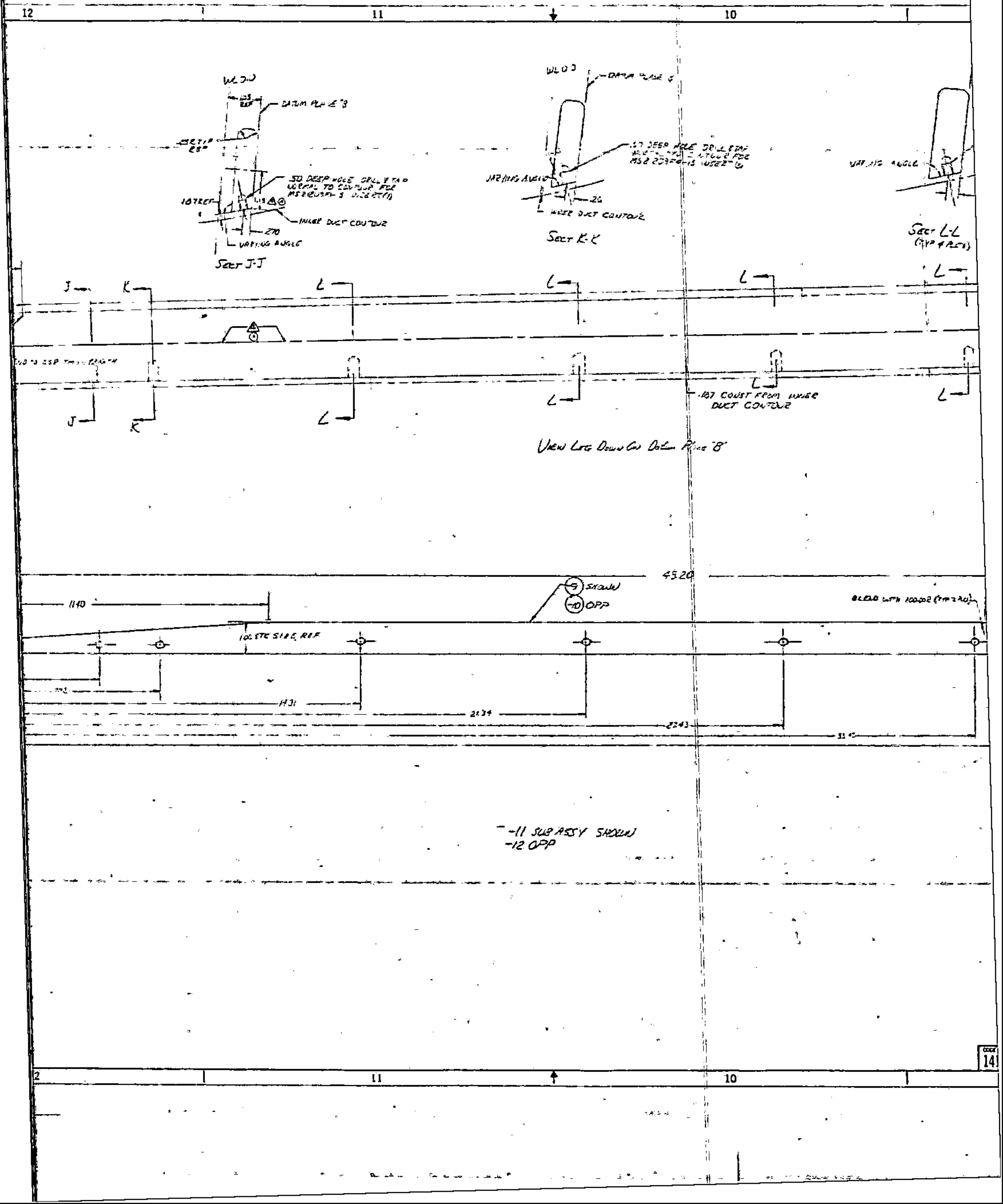
SCALE | RELEASED | SHEET

SECURITY CLASSIFICATION

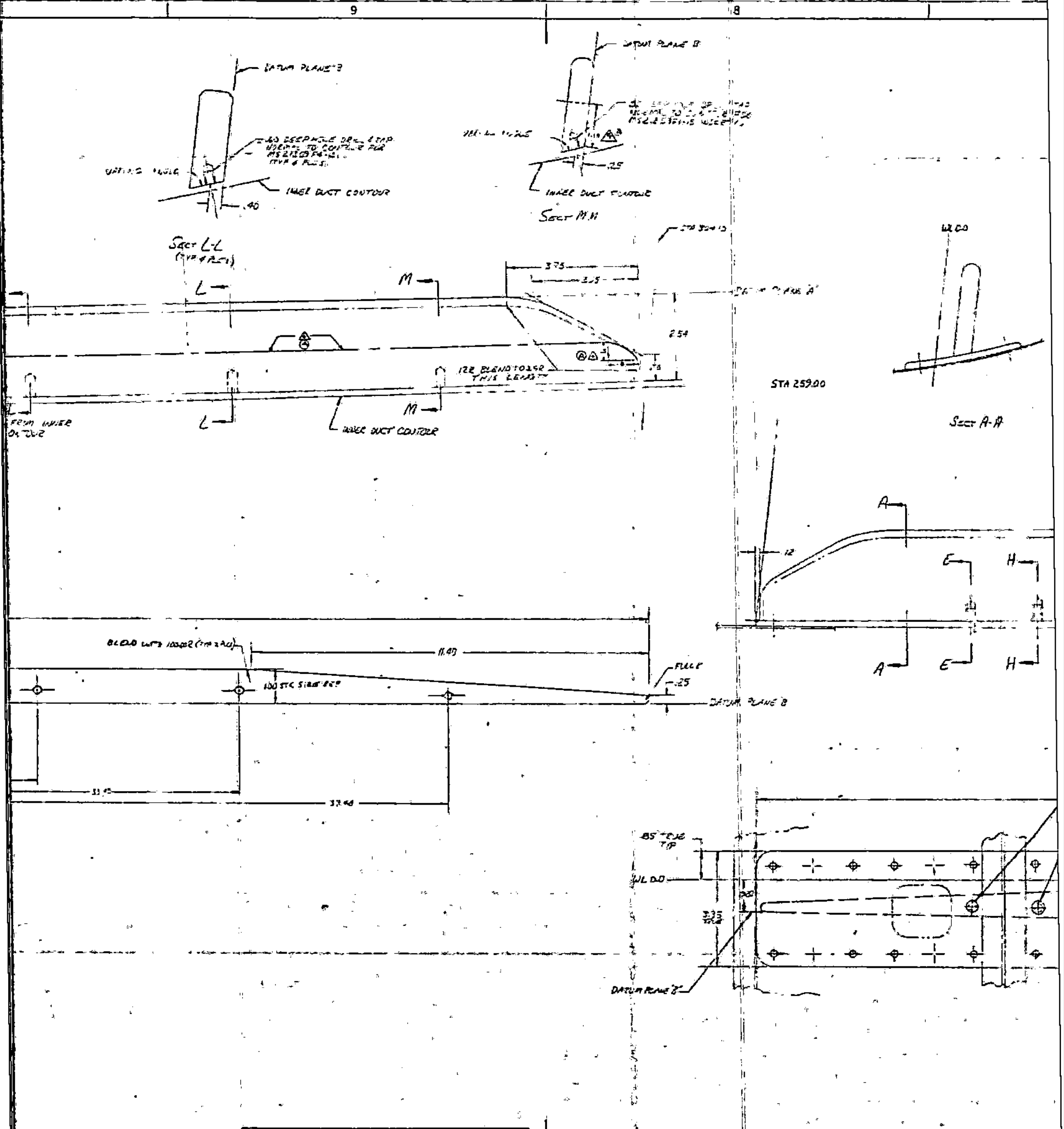
PACKAGE NO.

E111-PR051

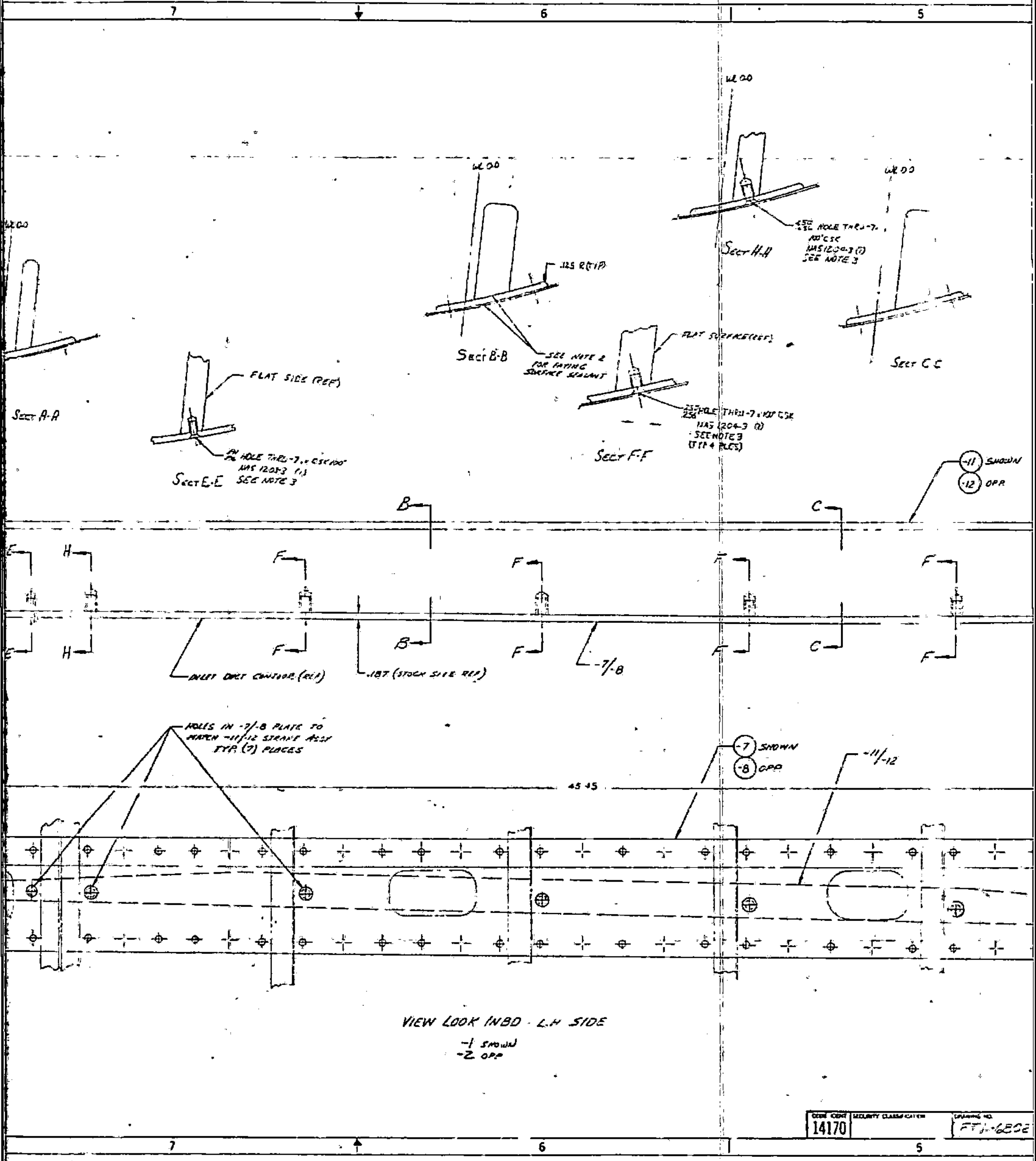

12
H
G
F
E
D
C
B
A
DATUM PLANE "C"
STA 259
3.80
2.85
J
K
2.45
.75
.12
.122 SLEEVE TO 252 IN. LENGTH
J
K
DATUM-PLANE "C"
11.40
FULL R
.25
4.05
12



12
11
10
WL 0.0
.25 REF
DATUM PLANE '8
.50 DEEP HOLE DRILL & TAP NORMAL TO CONTOUR FOR MS21209F1-5 INSERT (1)
.187 REF
INNER DUCT CONTOUR
.270
VARYING ANGLE
SECT J-J
WL 0.0
DATUM PLANE 8
VARYING ANGLE
.26
INNER DUCT CONTOUR
SECT K-K
VARYING ANGLE
SECT L-L
(TYP 4 PLCS)
J
K
L
.187 CONST FROM INNER DUCT CONTOUR
VIEW LOOKING DOWN ON DATUM PLANE '8'
45.20
-9 SHOWN
-10 OPP
11.40
BLEND WITH 100.002 (TYP 2 PLC)
100 STC SIDE REF
14.31
21.34
22.43
-11 SUB ASSY SHOWN
-12 OPP
14



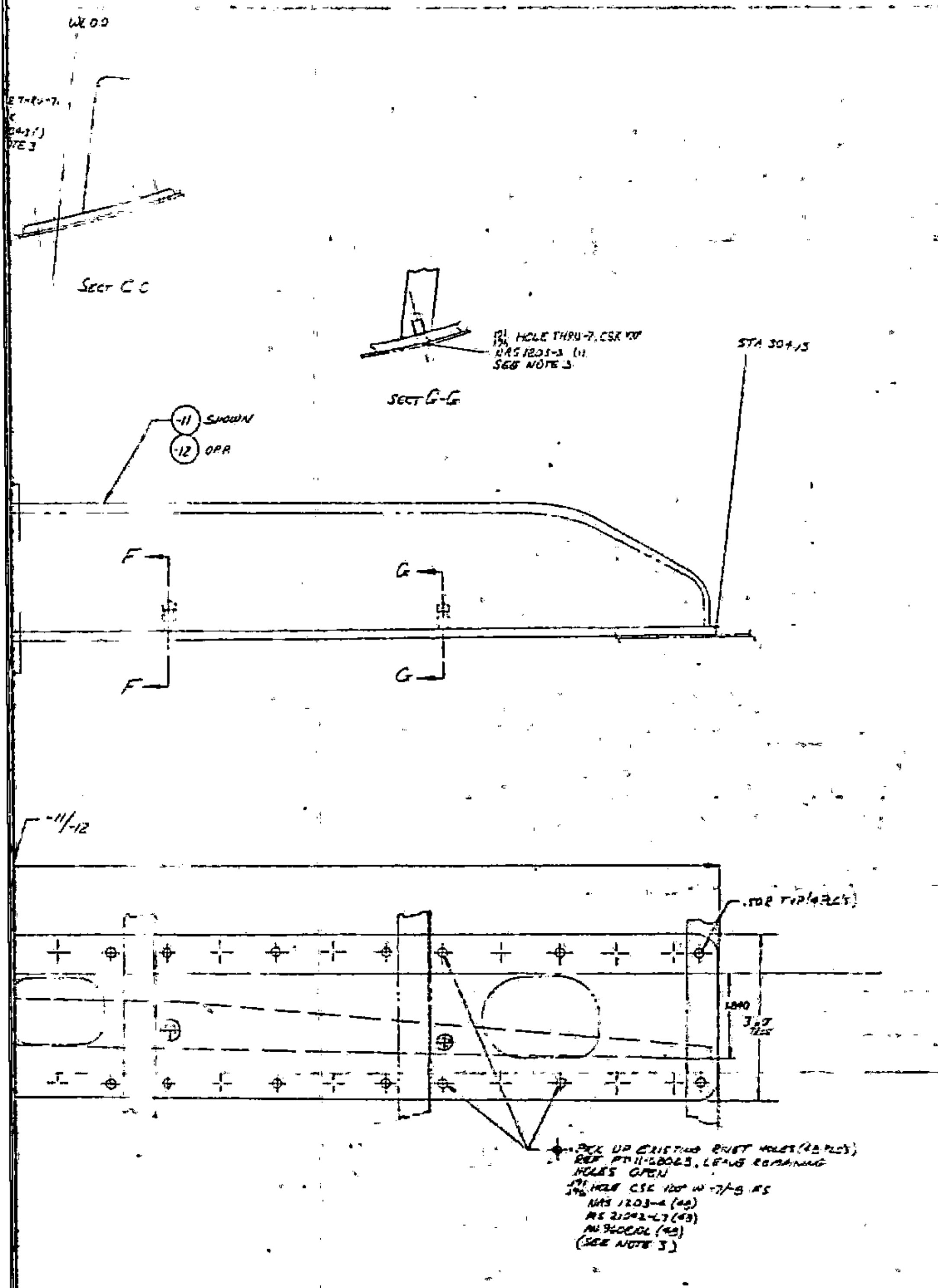
CODE IDENT	SECURITY CLASSIFICATION	DRAWING NO.	SH
14170		FT11-62027	


7
6
5
WL 0.0
WL 0.0
WL 0.0
WL 0.0
.125 R (TYP)
SEE NOTE 2 FOR FAYING SURFACE SEALANT
Sect B-B
.250/.256 HOLE THRU -7, 100° CSK NAS1204-3 (7) SEE NOTE 3
Sect H-H
Sect C-C
Sect A-A
FLAT SIDE (REF)
.250/.256 HOLE THRU -7, CSK 100° NAS 1204-3 (1) SEE NOTE 3
Sect E-E
FLAT SURFACE (REF)
.250/.256 HOLE THRU -7, 100° CSK NAS 1204-3 (1) SEE NOTE 3 (TYP 4 PLCS)
Sect F-F
-11 SHOWN
-12 OPP
B
C
E
H
F
INLET DUCT CONTOUR (REF)
.187 (STOCK SIZE REF)
-7/-8
HOLES IN -7/-8 PLATE TO MATCH -11/-12 STRAKE ASSY TYP (7) PLACES
-7 SHOWN
-8 OPP
-11/-12
45.45
VIEW LOOK INBD - L.H SIDE
-1 SHOWN
-2 OPP
7
6
5

CODE IDENT	SECURITY CLASSIFICATION	DRAWING NO.
14170		FTW-6852

REVISIONS				
ZONE	LTR	DESCRIPTION	DATE	APPROVED

REVISIONS				
ZONE	LTR	DESCRIPTION	DATE	APPROVED



NOTES:

1. ALL PARTS TO BE ALODINE FINISH ONLY (CO

2. APPLY EC1293 ON FAYING SURFACE ON INSTALL ON -11/-12 STRAKE ASSY & -7/-8 PLATE.

3. INSTALL ALL FASTENERS WITH "LOCTITE" GRADE "H" SEALANT.

△4 1½" WIDTH CONFIG. TRIM LINE, MAKE FROM -9 & -10, USE DATUM PLANE "B". BREAK EDGES RADIUS ENDS AS SHOWN & FAIR INTO EXISTING CONTOURS.

SECURITY CLASSIFICATION	DRAWING NO.	SH
	FT11-68027	

67

ZONE	LTR	DESCRIPTION	DATE	APPROVED
	A	INCORP "F" CONFIG. & MISC. CHGS		

-12	-11	2	[illegible]	UNIT OF MEAS	ITEM OR FIND NO.	CODE IDENT NO.	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	NO	STOCK SIZE, MATERIAL, SPECIFICATION
		AR	AR				LOCTITE"	SEALANT			MIL-S-22473 GRADE "H"
		AR	AR				EC 1293	SEALANT			
		40	40				MS 21042 L3	NUT			
		5	5				NAS 1204-?	BOLT			
		2	2				NAS 1203-?	BOLT			
		48	48				NAS 1203-4	BOLT			
		48	48				AN 960 C 10 L	WASHER			
5	5						MS 21209 F4-15	INSERT			
2	2						MS 21209 F1-15	INSERT			
1							-10	STRAKE			.100 x 3.25 x 45.5 [illegible]
	1						-9	STRAKE			.110 x 3.25 x 45.5 [illegible]
		1					-8	PLATE			.072 x 2.90 x 45.6 [illegible]
			1				-7	PLATE			.071 x 2.90 x 45 [illegible]
		1					-12	STRAKE ASSY			
			1				-11	STRAKE ASSY			

QTY REQD PER ASSEMBLY / INSTALLATION

PARTS LIST

WIDTH CONFIG. TRIM LINE. MAKE FROM -10. USE DATUM PLANE "3". BREAK EDGES DIUS ENDS AS SHOWN & FAIR INTO EXISTING NTOURS.

ALL ALL FASTENERS WITH "LOCTITE" DE "H" SEALANT.

LY EC1293 ON FAYING SURFACE ON INSTALLATION -11/-12 STRAKE ASSY & -7/-8 PLATE.

L PARTS TO BE ALODINE FINISH ONLY (OPTIONAL)

ES:

UNLESS OTHERWISE SPECIFIED
ALL MACHINED SURFACES √ 125
FIRST BASIC CODE NUMBER

SEE D-70914 FOR ABBREVIATION, CODE, AND SYMBOL REQUIREMENTS
SEE D-70910 FOR STANDARD MANUFACTURING AND INSTALLATION REQUIREMENTS
ASSOCIATED LIST REQUIRED SEE DOCUMENT SAME NUMBER PREFIXED LL ☐ FOR USAGE LIST

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES
TOLERANCES
.X ±.1 .XX ±.03 .XXX ±.010
ANGULAR PER D-70912

CONTRACT NO.

SECURITY CLASSIFICATION

CONVAIR DIVISION OF GENERAL DYNAMICS
SAN DIEGO, CALIFORNIA

F-102A INLET DUCT
STRAKE VERSION IV

SIZE J | CODE IDENT 14170 | DRAWING NO. FT11-68027

SCALE | SHEET

FT11-68027 SHEET 1 A

APPENDIX I

TEST OPERATIONS

APPENDIX I

TEST OPERATIONS

Flight Test Operations

Convair was assigned prime responsibility for test operations which included program planning, instrumentation, test aircraft configuration control and maintenance, conduct of tests and coordination of several agencies involved in the test program. Program "go ahead" was received on 4 June 1969 and the last data flight completed on 16 October 1969. A milestone chart is presented in Table I-1.

Aircraft Utilization

Total flights during program			35
Weather penetration flights	16		
Non-weather data flights	4		
FCF plus data flights	1		
Ferry flights	13		
FCF flights	1		
Total flight hours during program			49.8
Weather penetration flight hours	27.2	(54.62%)	
Non weather data flight hours	3.2	(6.42%)	
FCF plus data flight hours	1.3	(2.61%)	
Ferry flight hours	17.3	(34.74%)	
FCF flight hours	0.8	(1.61%)	
		(100.00%)	

Compressor stalls were encountered on 8 of 16 weather penetration flights and 3 of 4 non-weather data flights.

A total of 13 separate compressor stall incidents were encountered during the program.

Number of ferry flights required increased rapidly as the seasonal high altitude thunderstorm activity drew to a close. Early data flights were based out of San Diego and storms were encountered in Southern California and Arizona and seldom required ferry flights. The last storm data flight was conducted over Southern Florida and other storms toward the end of the season were encountered in New

Mexico and Texas requiring one or more ferry flights. In most instances, however, the number of ferry flights did not contribute significantly to the cost of the program.

Flights for this program were 49 through 83, inclusive. Flight numbers, duct configurations, maneuvers, and data obtained are shown on Table I-2.

Ground Operations

The test aircraft was based at Lindbergh Field-San Diego, California. All maintenance, modifications and flights utilized GD/Convair personnel. Logistic support was provided by SAAMA, various Air Force Depots and Air National Guard Squadrons.

Maintenance and modifications to the airplane were accomplished with Convair internal documentation. This was in accordance with the Test Program Plan, Reference B. In addition, aircraft records were maintained in accordance with Air Force Technical Order 00-20-5.

Incorporation of mandatory TCTO's in functioning systems of the aircraft were accomplished by GD/Convair. Mandatory TCTO's which affected systems that were not functional were waived in accordance with Technical Order 00-5-1. Most waived TCTO's applied to the MG-10 fire control system.

Duct vane and strake designs were analyzed and approved by stress and dynamics engineers prior to fabrication or flight. There were no inspection rejections or rework required on any part of the duct system throughout the program. An engine run was completed after each duct modification or vane angle of attack adjustment, and baseline data analyzed.

Flight Operations

Flight operations started by reviewing the test program plan and data obtained on preceding flights. Since majority of test points required thunderstorm penetration, it was necessary to hold daily briefings with the United States Weather Bureau Foreaster (located in a building adjacent to Convair Engineering building). Additional weather information was obtained from FAA Flight Service and various FAA flight centers.

Flight Operations (Continued)

Flight planning was accomplished after weather forecasts indicated geographical area, size and course of various thunderstorms. Air Force Bases which could be utilized as refueling stops during the missions were contacted prior to aircraft launch and permission obtained to land. Any "Notices to Airmen" in effect at various bases were also discussed.

The operations engineer then set up each test plan on a series of flight cards in a condensed format, with concurrence of the Principal Engineer and pilot. Both flight plan and cards were distributed to the pilot and instrumentation/analysis engineers at a pre-flight meeting.

Weather flights depended on FAA-Air Route Traffic Control Centers which controlled airspace where storms were located. These centers provided excellent support for the program. Block altitude assignments, vector service, course and altitude changes, and guidance around restricted areas were handled in a most satisfactory manner.

It was standard procedure for GD/Convair to telephone these FAA Centers when flights were estimated to be 50 per cent complete. Information was gained on test progress, fuel remaining, pilot's recovery intentions, and estimated time of arrival.

Operations which required use of the Air Force Flight Test Center range were conducted in the same manner as that used during the Compressor Stall Investigation Program.

Final weather test points required that the aircraft be ferried to Florida to seek suitable thunderstorms. An instrumentation engineer and instrumentation technician were also sent to Florida to pre-flight the instrumentation and to remove any water which may have accumulated in the P_{t2} rakes. Refueling and transient line maintenance were provided by various Air Force Bases.

Instrumentation Reliability

There were no lost data flights due to instrumentation malfunctions. Of 82 individual measurements recorded on each flight, only P_{t2} rake position 4D was a chronic problem. (Note: Later flights had 83 measurements after reactivation of the compressor bleed valve position instrumentation.) However, as stated in the Test Program Plan, only 36 of these 40 P_{t2} rake positions were

Instrumentation Reliability (Continued)

needed for satisfactory data. Probe 4D problems were eventually resolved with discovery of a small spur on a pin adjacent to an electrical connector which had worn through the sleeving and would occasionally "ground" the signal.

Safety Considerations

Icing conditions were not encountered on this program, so the efficacy of ICEX, ice adhesive depressant, was not demonstrated.

A special test ice detector system with pilot readout was used on each weather flight. Test of this unit was a pilot check list item.

The "all points" engine ignition was not held manually "ON" during adverse weather penetration to portray the actual condition wherein an operational pilot would not use it, unless needed, due to the 10-minute limit. This differed from the technique used on compressor stall investigation program. Note: The aircraft was also equipped with "auto-igniter" system.

The program was completed without material loss or injury.

TEST SUPPORT

San Antonio Air Materiel Area

For this program, SAAMA furnished on a bailed basis to Convair: the test aircraft, two J57-P23A engines with modified first stage nozzle turbine area, fuel controls, afterburners, aircraft spare parts, and ground support equipment.

Oklahoma City Air Materiel Area

OCAMA provided engine modification which consisted of re-sizing turbine nozzle area to a dimension within 93 - 94 square inches. This was accomplished by changing the stator blocks in the A5 area and which required re-striking of vanes to a new class size.

Engine calibration runs prior to and after engine modification were also conducted by OCAMA.

TABLE I-1

PROGRAM MILESTONES

DATE	EVENT
4 June 1969	TWX OK to start fix development program
13 June 1969	Request VE-06 Mod AD Test Program Plan "Pre-Print" complete
18 June 1969	Configuration #1 design "off board"
19 June 1969	Configuration #1 drawings to SAAMA for approval
20 June 1969	Engine "C" arrived San Diego from OCAMA
10 July 1969	Completed configuration #1 installation in aircraft
21 July 1969	Flight 49, first program flight
22 July 1969	Started fabrication of configurations #2 and #3
4 August 1969	Stop order on configuration #3 work
13 August 1969	Configuration #2 installation
29 August 1969	Remove configuration #2
3 September 1969	Flight 61, no vanes - support strakes only
3 September 1969	Engine "D" arrived at San Diego (spare)
10 September 1969	Flight 64, plus configuration at 6 degrees to water line plus configuration #2 support strakes. Best indicated results.
11 September 1969	DCAS-Los Angeles safety inspection
18 September 1969	First flight with 3-inch strake
29 September 1969	First flight with $1\frac{1}{2}$ inch strake
13 October 1969	Airplane to Southeast United States for weather flights
17 October 1969	Prelininary report on test results mailed to SAAMA
20 October 1969	Airplane returned to San Diego after accomplishing final compressor stall program data flight in Florida
23 October 1969	Duct vane anti-ice mock-up and final prototype design of duct vane modifications started

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
Configuration #1 aft horizontal vane, at 2 degrees to water line	49	FCF									
	50	Snap-up									
	51	Thunderstorm penet	35-40	0.80 - 0.85	NRT	None	—	Between Gila Bend & Tucson, Arizona	Dissipating	35-40	Light
	52	Thunderstorm penet	41	240	MIL	N_2 stall-1 bang	Ignition	30 miles west of Prescott, Arizona	Mature	50	Light to moderate
Configuration #1 at 4 degrees	53	Thunderstorm penet	42 and 41	250 and 240	MIL	—	—	30 miles west of Prescott, Arizona	Mature with slight anvil overhang	50 and 43	Light to moderate
Configuration #1 at 6 degrees	54	Thunderstorm penet	41	250		—	—	20 miles northeast of Phoenix Arizona	Building cumulus	43	Moderate

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
Configuration #2 forward horizontal vanes, at 0 degrees angle of incidence plus support strakes for Configuration #1	55	FCF									
	56	Thunderstorm penet	34-38	250-280		——	——	50 miles south southeast of Gila Bend, Arizona	Mature	38 with slight anvil overhang to the northeast	
			34.5	230	MIL	N_2 stall - 4 bangs	Ignition, and nose lowered slightly in left turn to clear cloud. Power was not changed, EGT steady at 638 degrees C				
Configuration #2 at 2 degrees plus support strakes for Configuration #1	57	Thunderstorm penet	42	250	MIL	N_2 stall- 2 bangs	Ignition and nose lowered slightly	50 miles south southeast of Gila Bend, Arizona	Mature	42-43	
			40	250	MIL	N_2 stall- 3 bangs	Ignition and nose lowered slightly				
	58	Snap-up	35-45	1.07 - 170	Max A/B	N_2 stall- EGT 650 degrees C 500 degrees C	Throttle to idle, ignition, and nose lowered by rolling 90 degrees left and falling through	——	——	——	——

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
	58 (Cont'd)	Snap-up		<170		Low energy bangs, EGT 688 deg C ↓ 690 deg C with total EGT time above 670 deg C about 5 sec	Wings level Engine shut down Airstart initiated at 38,000 & 270 KIAS. Minimum altitude in recovery 33,000 feet				
		Roller coaster 2.5 to 0.5 g	45	0.9		—	—	—	—	—	—
		High angle of attack	On descent from 38 to 33,000 feet with power at 82%, speed brakes out and gear down, airspeed was reduced to 150 KIAS. Below 150 KIAS power was set at 89%, gear and speed brakes in, and airspeed was reduced to 100 KIAS at 35,000 feet. Angle of attack of 29 degrees was observed. No stalls experienced.								

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
Configurations #1 and #2 support strakes only (no vane installed)	59	Ferry to Edwards AFB									
	60	Snap-up	36-48 Recovery at 39	1.1-160	Max A/B	With fwd stick, N_2 stall. one bang EGT 660 degrees C	Throttle to idle, ignition, & nose lowered by rolling 60 degrees and falling through	—	—	—	—
		High angle of attack	On descent from 38 to 33,000 feet with power at 82%, speed brakes out and gear down, airspeed was reduced to 150 KIAS. Power was changed to 89%, gear and speed brakes were retracted, and airspeed was reduced to 105 KIAS at 35,000 feet. Angle of attack of 28.5 degrees was observed. No stalls experienced.								
	61	Thunderstorm penet	36-40	260-280		—	—	30 miles northwest of Gila Bend, Arizona	36-40	Cirrus & cumulus	
			36	260	MIL	N_2 stall-one bang	Ignition and nose lowered slightly		37	Rapidly building puffy cumulus	Moderate
			36	260		N_2 stall-six bangs EGT 670 degrees C	Throttle to idle		37	Rapidly building puffy cumulus	Moderate

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
Configuration #1 at 6 degrees plus support strakes for Configuration #2	62	Ferry to Luke AFB									
	63	Thunderstorm penet	31-34	0.9 & 0.7	MIL	—	—	Within an area of 100 miles around Tucson, Arizona	Large cumulus	25 with some towers to 35-36 but separated from mother cloud	Light to moderate
	64	Thunderstorm penet	31-35	0.9 & 0.7		—	—	50 miles east of Blythe, California	Mature	36-39	
			33	0.7	MIL	N_2 stall- 6 bangs EGT 667 degrees C	Ignition, nose lowered, Power to idle				Moderate to severe
			29	0.85	MIL	N_2 stall- 4 bangs EGT 657 degrees C	Ignition, nose lowered, throttle chop to idle				

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
	65	Sideslip	25 &40	0.7 & 0.92							
		High angle of attack in pullup	35	100	MIL						
		Supersonic flight	23	1.35	Max A/B						
Configuration #4, support strakes for Configuration #2 only	66	Thunderstorm penet	40	300 decelerating to 260	MIL	—	—	Twentynine Palms, California	Mature	40.5	Light to moderate
			35	250	MIL	N_2 stall-one bang	Ignition, nose lowered slightly				
Clean ducts, support strakes attachment holes filled with screws	67	Ferry to Luke AFB									
	68	Thunderstorm penet	25-28	280 - 310	MIL	—	—	40 miles south of St. Johns, Arizona	Dissipating	30	Light to moderate

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
Configuration #5, 3-inch strakes at 2 degrees (Configuration #1 support strakes removed)	69	Thunderstorm penet	25-28	220-300	MIL	—	—	50 miles south of Albuquerque, New Mexico	Building	29	Light to moderate
	70	Ferry from Kirtland AFB									
	71	Ferry to Kirtland AFB									
	72	Thunderstorm penet	36 & 33	260 & 230	MIL	—	—	70 miles east southeast of Ft. Stockton, Texas	Building	37	Moderate
			32	250		N_2 stall-one bang	Ignition, nose lowered slightly				
	73	Ferry to San Diego									

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
Configuration #6, $1\frac{1}{2}$ inch strakes at 2 degrees (Configuration #1 support strakes removed)	74	Thunderstorm penet	20	300 - 350	82-89%	—	—	Prescott, Arizona	Broken cumulus	21	—
		High angle of attack	35	100	89%	—	—	—	—	—	—
	75	Thunderstorm penet	28 - 30	280 - 340	MIL	—	—	60 miles SW of Bryce Canyon, Utah	Mature cells	33	Light to Moderate
	76	Ferry from Nellis AFB to San Diego									
Clean ducts, support strakes attachment holes filled and smoothed	77	Ferry to Webb AFB									
	78	Ferry to Ellington AFB									
	79	Ferry to Tyndall AFB									
	80	Thunderstorm penet	27-30	220-320	MIL			50 miles west of Miami, Florida	Building	31	Light to Moderate

TABLE I-2. COMPRESSOR STALL FLIGHT AND CONFIGURATION SUMMARY (Continued)

CONFIGURATION	FLT. NO.	FLIGHT PURPOSE	AIRCRAFT CONDITIONS			STALL CONDITIONS		THUNDERSTORM CONDITIONS			
			ALTITUDE ($X10^{-3}$ FEET)	SPEED (KIAS OR MACH)	THRUST SETTING	STALL TYPE	CORRECTIVE ACTION	LOCATION	STAGE	STORM TOPS ($X10^{-3}$ FEET)	TURBULENCE EXPERIENCED
	80 (Cont'd)	Thunderstorm penet	27	290	MIL	N_2 stall- 4 bangs progressed to overtemp 682 deg C	Ignition and nose lowered Throttle to idle	50 miles west of Miami, Florida	Building	31	Light to moderate
	81	Ferry from MacDill AFB to Barksdale AFB									
	82	Ferry to Kirtland AFB									
	83	Ferry to San Diego									

APPENDIX II

RECOMMENDED CHANGES TO FLIGHT MANUAL

TECHNICAL ORDER 1F-102A-1

APPENDIX II

RECOMMENDED CHANGES TO FLIGHT MANUAL

TECHNICAL ORDER 1F-102A-1

The following recommended amendments to the flight manual are based on F-102A aircraft flight test results. It is believed that these amendments are fundamental enough to be applicable to the TF-102A. Therefore, it is recommended that the amendments be applicable to both types.

Page 2-24 AIM-26A Escape Maneuvers

Snap-up paragraph 2. Change dive angle from "15" degrees to "45" degrees. Add "if a compressor stall is experienced during the snap-up phase."

Page 2-25 Figure 2-6. Amend figure to show 45 degree dive angle.

Page 3-16 b(3) "Failure of the fuel control to reduce engine rpm with colder inlet air temperatures can cause compressor stalls at high altitude." Paragraph is not substantiated by test results and should be deleted.

Page 3-17 Compressor Stall Procedures. Delete paragraphs 1 through 6 and warning following paragraph 5. Replace with:

1. THROTTLE - INBOARD;

   IGNITION BUTTON - DEPRESS;

   EGT GAGE - CHECK.

   At first indication of compressor stall, simultaneously shut off afterburner, depress and hold ignition button, and check exhaust gas temperature gage for overtemperature.

CAUTION

Continuous use of the ignition system in excess of ten minutes, or consecutive continuous usages without a ten-minute interim cooling period may result in damage to the ignition system which will render it inoperable for restarts. Energizing the ignition system will not prevent compressor stalls, but may aid in preventing a flameout.

APPENDIX II - RECOMMENDED CHANGES TO FLIGHT MANUAL TECHNICAL ORDER 1F-102A-1 (Continued)

Page 3-17 (Continued)

2. THROTTLE - IDLE (IF EGT APPROACHES LIMITS).
3. THROTTLE - OFF (IF STALL CANNOT BE BROKEN OR EGT INCREASE PERSISTS)

NOTE

Engine rpm below 40% will be insufficient to provide normal ac generator operation. Prior to moving the throttle to OFF, the emergency ac generator should be selected by placing the AC bus switch to RESET, then ON.

Engine windmilling rpm at 220 KIAS will be insufficient to provide normal ac generator operation and may not be sufficient at 250 to 275 KIAS.

With the throttle in the OFF position, duct rumble will be present and should not be interpreted as a compressor stall.

4. MAINTAIN AIRCRAFT CONTROL

WARNING

Primary cause of aircraft accidents associated with compressor stalls has been loss of aircraft control.

NOTE

Reducing altitude and increasing airspeed will improve compressor stall recovery potential or will possibly preclude re-entering a stall.

5. Make an air start

After a stabilized condition is obtained, restart the engine. Refer to ENGINE AIR START for starting procedures.

NOTE

Past experience indicates that an immediate airstart attempt may not be successful because the engine may still be in a stalled condition.

APPENDIX II - RECOMMENDED CHANGES TO FLIGHT MANUAL TECHNICAL ORDER 1F-102A-1 (Continued)

Page 3-17 (Continued)

6. AFTO 781 - Record any compressor stall.

Record any compressor stall and indicate duration and peak temperatures of any overtemperature operation.

Page 9-14 "Turbulence and Thunderstorm"

CAUTION

In reference to probable flameout, change word probable to possible flameout.

Page A9-1 Part 9 Mission Planning

To Part 9, add Figure 38 of this report and action to check temperatures aloft for use of the figure.

APPENDIX III

INSTRUMENTATION

APPENDIX III

INSTRUMENTATION

AIRBORNE MEASUREMENTS

All airborne data were recorded on magnetic tape.

Measurements were selected, Table III-1, for the F-102 Compressor Stall Fix Program to determine flight environment and to provide quantitative data on the performance of the aircraft ducts. Measurements were selected and optimized to provide data on gas flow and other parameters affecting compressor stalls rather than on propulsion system performance in terms of thrust, specific fuel consumption, etc. Figures III-1 and III-2 show approximate locations of the aircraft and engine instrumentation transducers.

A complete description of the instrumentation system; covering design, tape recorder, subcarrier oscillators, and signal conditioning units may be found in Reference A, the final report of Compressor Stall Investigation Program. The instrumentation system is essentially the same as was used in that program, with the following changes:

a) Eleven measurements relating to fuel control/afterburner functions, not pertinent to this portion of the program were not monitored (Reference Figure III-2).

b) An additional P_{t2} rake was added at the scroll top center line. However, this rake was not connected to the basic instrumentation system but was used only to determine ice particle distribution.

RAKE INSTALLATION

Figure III-3 is schematic showing radial locations of the T_{t2} and T_{t3} rakes. Axially the P_{t2} rakes were approximately $3\frac{1}{2}$ inches forward of the low pressure compressor inlet flange. The five ports on each rake were spaced to represent an equal area distribution.

The four T_{t3} probes, each consisting of five thermocouples manifolded together, measured temperature in the gas path at the low pressure compressor exit area, and were located radially at approximately 2, 5, 8, and 10 o'clock positions.

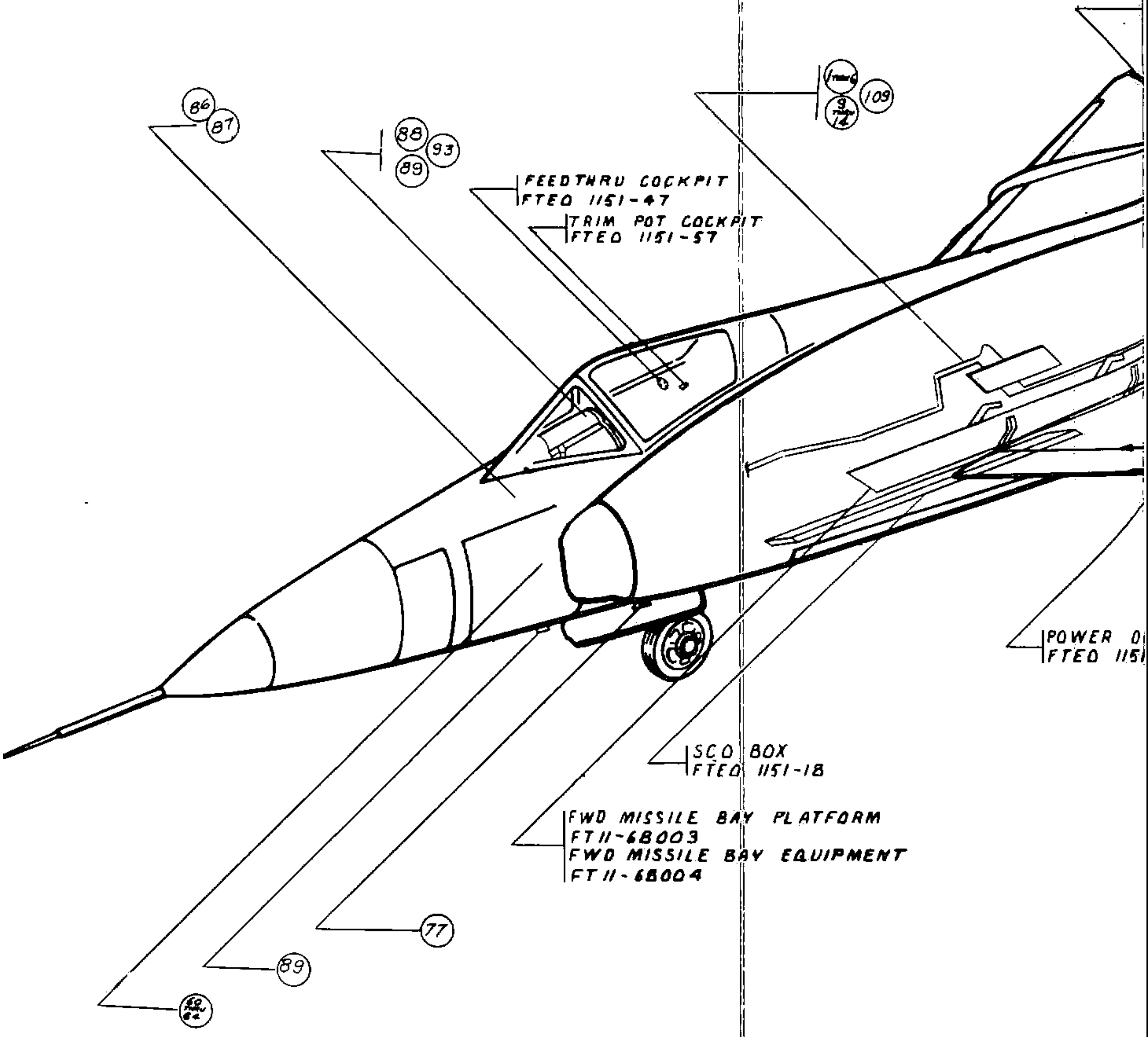
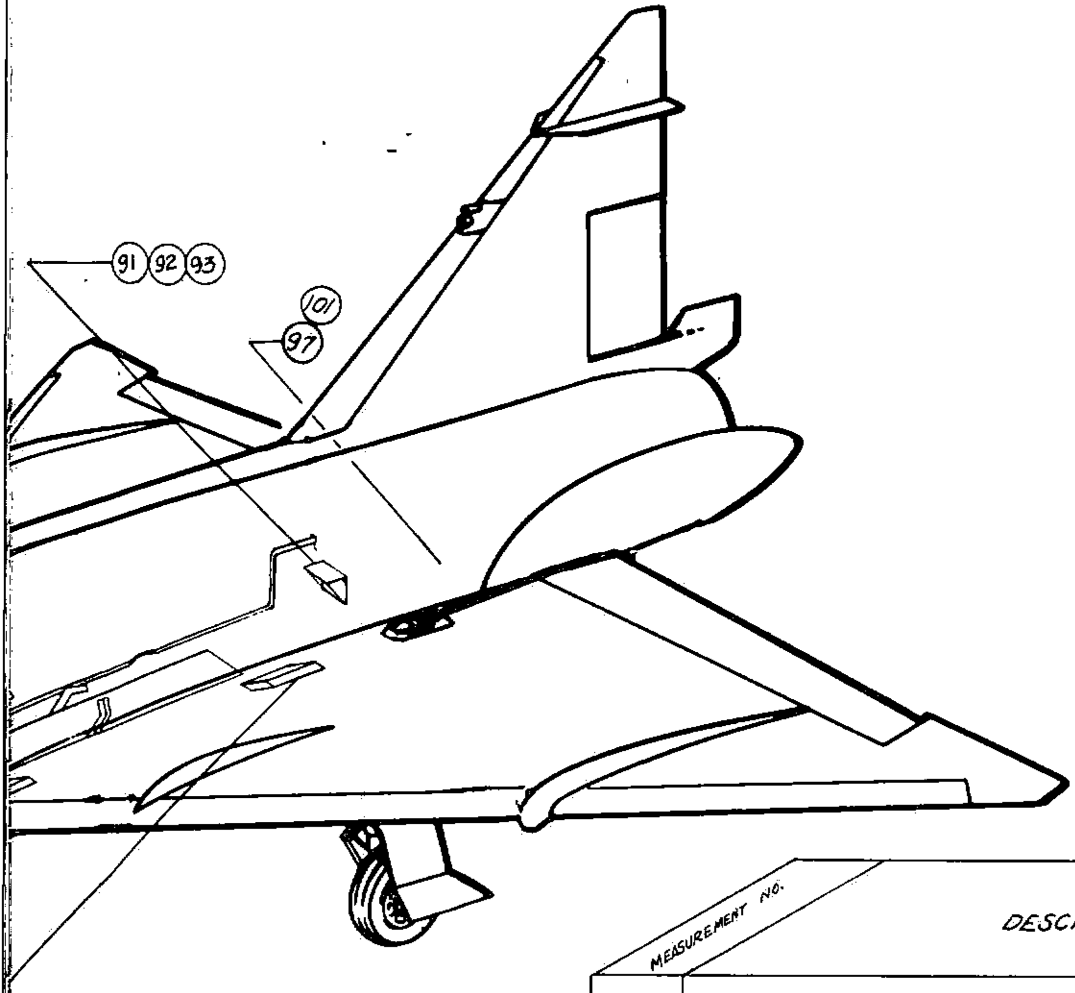
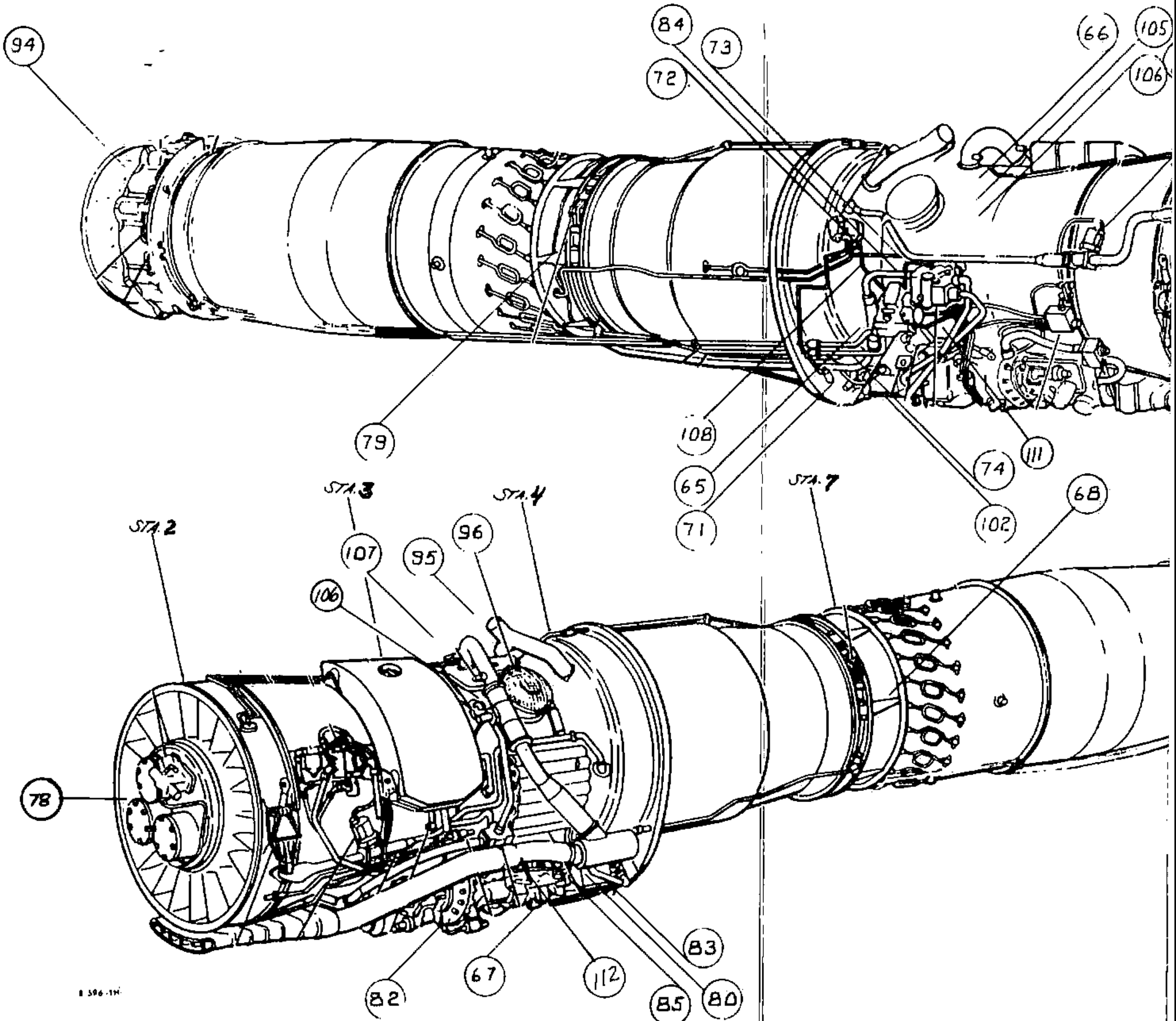

86
87
88
89
93
FEEDTHRU COCKPIT
FTED 1151-47
TRIM POT COCKPIT
FTED 1151-57
1 THRU 6
9 THRU 14
109
POWER D
FTED 115
SCO BOX
FTED 1151-18
FWD MISSILE BAY PLATFORM
FT11-68003
FWD MISSILE BAY EQUIPMENT
FT11-68004
77
89
60 THRU 64

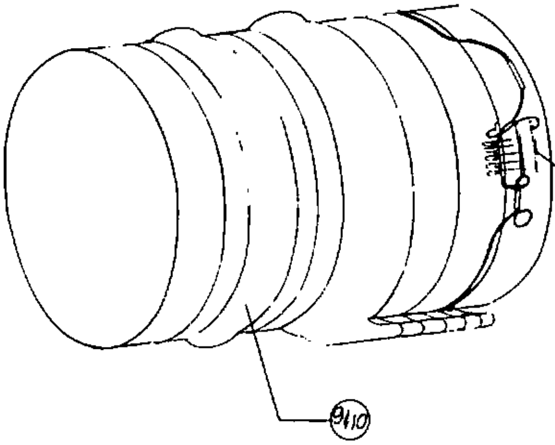
FIGURE III-1
AIRCRAFT INSTRUMENTATION LOCAT



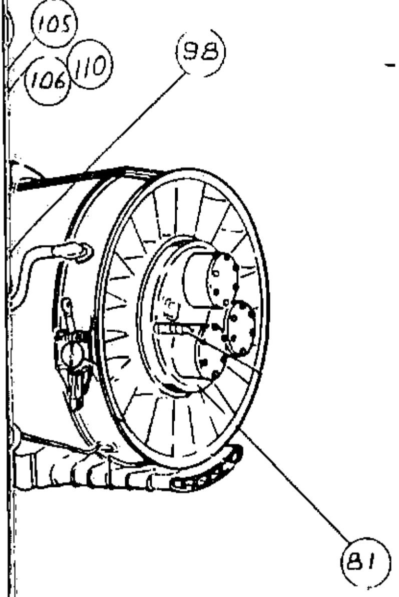
MEASUREMENT NO.	DESCRIPTION
109	DIFFERENTIAL STATIC PRESSURE, LEFT/RIGHT
★ 101	A/B MOTOR ACTUATOR POSITION
★ 97	RAIN CLEARING VALVE POSITION
93	NORMAL ACCELERATION, C.G.
92	ROLL ANGLE, GYRO
91	PITCH ANGLE, GYRO
89	SIDESLIP ANGLE, PRODUCTION PROBE
88	ANGLE OF ATTACK, PRODUCTION PROBE
87	ALTITUDE
86	AIRSPEED
77	RAM AIR TEMPERATURE
60-64	TOTAL PRESSURE, BOUNDRY LAYER RAKE
11-14	STATIC PRESSURE, DUCT FLOOR
1-6	STATIC PRESSURE, SPLITTER DUCT WALL

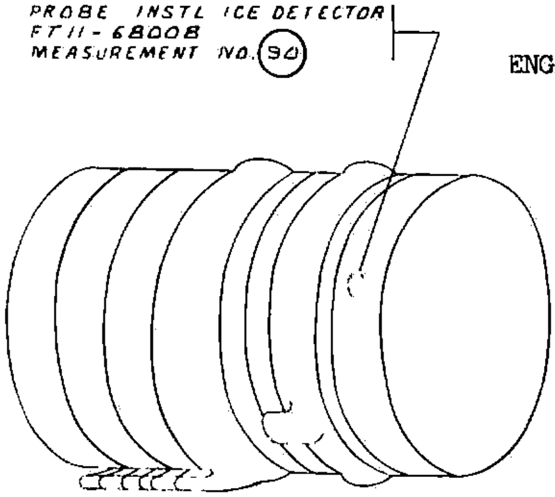
★ NOT MONITORED ON FLIGHTS 49 THRU 83





TRANSDUCER INSTL PT_2 & PS_2
FT 11-68009
MEASUREMENT NO. (56) THRU (59), (69) & (70)





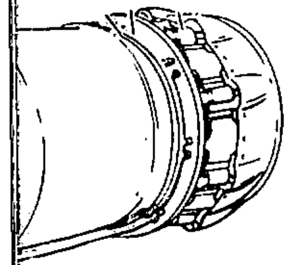


FIGURE III-2
ENGINE INSTRUMENTATION LOCATIONS

	MEASUREMENT NO.	DESCRIPTION
	78	T_{t2} COMPRESSOR INLET TOTAL TEMPERATURE
	112	T_{t}3-3 TEMPERATURE BETWEEN N_1 AND N_2 AT 8:00
	111	T_{t}3-2 TEMPERATURE BETWEEN N_1 AND N_2 AT 5:00
	110	T_{t3}-4 TEMPERATURE BETWEEN N_1 AND N_2 AT 10:00
*	108	ENC TO IGNITER PRESSURE
*	107	GROUND AIR COOLING VALVE POSITION N_1 BLEED SENSE ELEC CONTROL SIGNAL
	106	T_{t}3.1 TEMPERATURE BETWEEN N_1 AND N_2 AT 2:00.
	105	P_{t}3 TWO 5 PROBE RAKES AT APPROX 1:30 & 10:30 POSITIONS ALL MANF. FOR 1 XDCR
	102	IGNITION ON / OFF
*	101	A/B ON MOTOR ACTUATOR POSITION (LOCATED ON A/C VIEW)
*	98	ANTI-ICEING VALVE POSITION SENSE ELEC CONTROL SIGNAL
*	97	RAIN CLEARING VALVE POSITION SENSE ELEC CONTROL SIGNAL (ON A/C VIEW
	96	BLEED VALVE POSITION (ANTI-SURGE)
*	95	GROUND COOLING VALVE POSITION N_2 BLEED SENSE ELEC CONTROL SIGNAL
*	94	A/B NOZZLE POSITION
	85	POWER LEVER ANGLE - AT FUEL CONTROL
	84	FUEL FLOW A/B
	83	FUEL FLOW ENGINE
	82	N_2 RPM
	81	N_1 RPM
	80	FUEL TEMP. AT ENGINE TEST FLOW METER
	79	T_{t7} PARALLEL EGT INDICATOR & HOT SECTION FACTOR COMPUTER
*	74	A/B ACTUATOR PRESSURE
*	73	IGNITER TO EXHAUST CONTROL PRESSURE
*	72	IGNITER SQUIRT PRESSURE
*	71	A/B FUEL PRESSURE
	68	P_{t7} RING MANIFOLD (4 PROBES)
	67	P_S4.5 (BURNER PRESS.) "T" AT FUEL CONTROL
	66	P_S3 ONE BOSS AT APPROX 1 OCLOCK POSITION
	65	P_S4 ONE BOSS AT APPROX 4 OCLOCK POSITION
	56-59	STATIC PRESSURE, COMPRESSOR INLET (P_S2-1,2,3,4)
	69	P_{t}2 COMPRESSOR INLET PRESSURE (REF. TO AIRSPEED TOTAL)
	70	P_S2M MASTER P_S2 REFERENCE PRESSURE (DIFF. P_S0)
	90	ICE DETECTOR
	9 & 10	P_S2 ENGINE SCROLL, STUB DUCT

* NOT MONITORED ON FLIGHTS 49 THRU 83

FIGURE III - 3

COMPRESSOR INLET TOTAL PRESSURE (P_{t_2}) AND TEMPERATURE (T_{t_2}) PROBE LOCATIONS

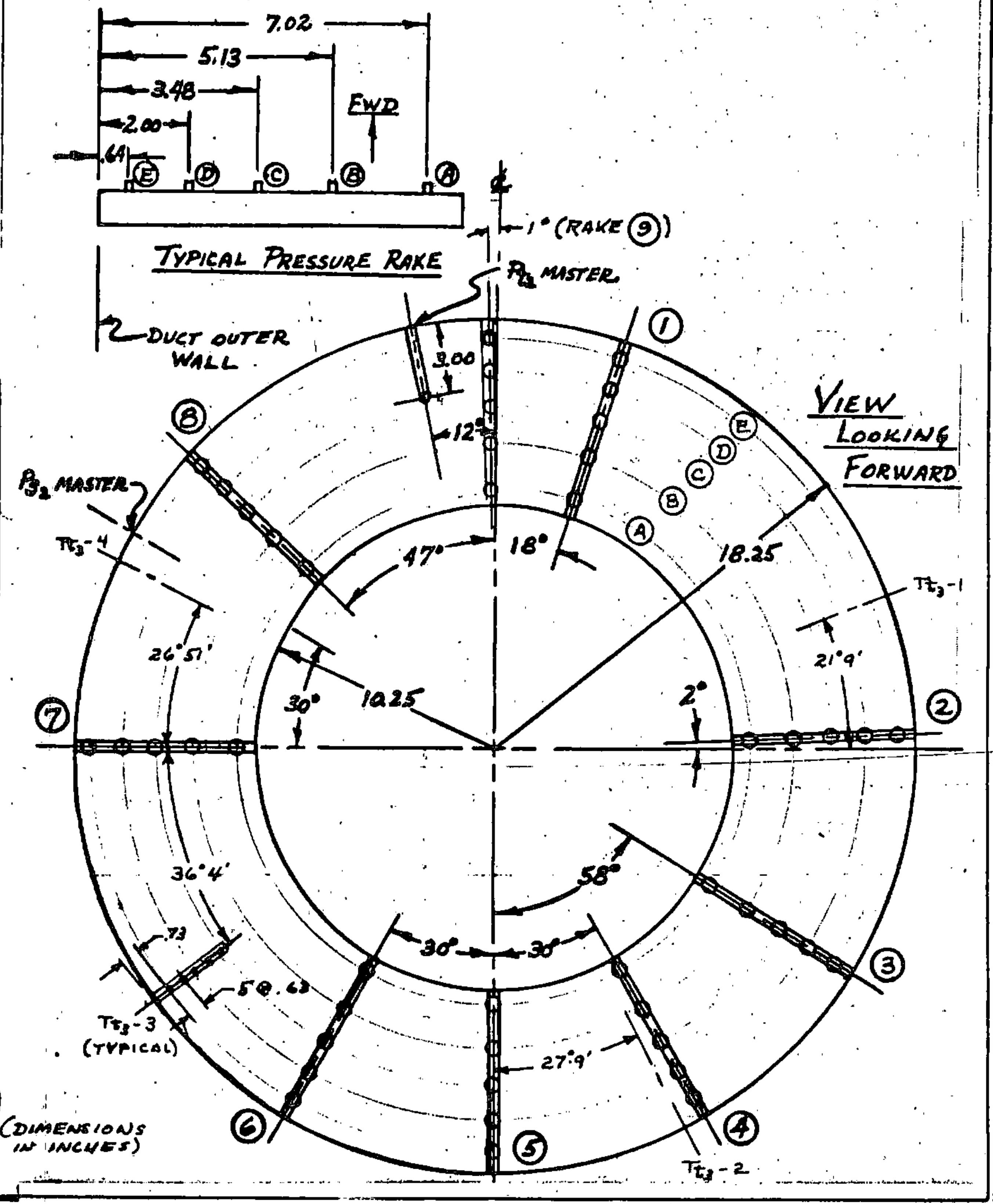


TABLE III FLIGHT TEST MEASUREMENTS

Tape Location		I.D. No.	Measurement
Track	Freq		
1		103	IRIG Time, Format B
2	1.3	106	T_{t3} #1, R/H 12:00
	1.7	77	Ram Air Temperature
	2.3	78	T_{t2}
	3.0	79	T_{t7}
	3.9	80	Fuel temp
	5.4	110	T_{t3} #4, L/H 10:00
	7.35	69	Master P_{t2}
	10.5	70	Master P_{s2}
	22.0	104	Voice
3	3.9	16	Rake 1A
	5.4	17	Rake 1B
	7.35	18	Rake 1C
	10.5	19	Rake 1D
	14.5	20	Rake 1E
	22.0	103	IRIG Time, Format B
4	1.3	111	T_{t3} #2, R/H 5:00
	1.7	112	T_{t3} #3, L/H 8:00
	2.3	86	Airspeed
	3.0	87	Altitude
	3.9	21	Rake 2A
	5.4	22	Rake 2B
	7.35	23	Rake 2C
	10.5	24	Rake 2D
	14.5	25	Rake 2E
	22.0		Compensation
5	2.3	88	Angle of Attack
	3.0	89	Sideslip
	3.9	26	Rake 3A
	5.4	27	Rake 3B
	7.35	28	Rake 3C
	10.5	29	Rake 3D
	14.5	30	Rake 3E

TABLE III FLIGHT TEST MEASUREMENTS (Continued)

Tape Location		I.D.	Measurement
Track	Freq	No.	
6	1.3	91	Pitch Attitude
	1.7	92	Roll Attitude
	2.3	93	Normal Acceleration
	3.9	31	Rake 4A
	5.4	32	Rake 4B
	7.35	33	Rake 4C
	10.5	34	Rake 4D
	14.5	35	Rake 4E
7	3.9	36	Rake 5A
	5.4	37	Rake 5B
	7.35	38	Rake 5C
	10.5	39	Rake 5D
	14.5	40	Rake 5E
8	3.9	41	Rake 6A
	5.4	42	Rake 6B
	7.35	43	Rake 6C
	10.5	44	Rake 6D
	14.5	45	Rake 6E
9	2.3	59	P_{s2} - 4
	3.0	58	P_{s2} - 3
	3.9	46	Rake 7A
	5.4	47	Rake 7B
	7.35	48	Rake 7C
	10.5	49	Rake 7D
	14.5	50	Rake 7E
10	2.3	56	P_{s2} -1
	3.0	57	P_{s2} -2
	3.9	51	Rake 8A
	5.4	52	Rake 8B
	7.35	53	Rake 8C
	10.5	54	Rake 8D
	14.5	55	Rake 8E

TABLE III FLIGHT TEST MEASUREMENTS (Continued)

Tape Location		I.D.	Measurement
Track	Freq	No.	
11	3.9	3	P_s Splitter, Station 203
	5.4	4	P_s Splitter, Station 231
	7.35	5	P_s Splitter, Station 273
	10.5	6	P_s Splitter, Station 316
	14.5	109	ΔP Inlet Duct, Station 266
12	1.3	65	P_{s4}
	1.7	66	P_{s3}
	2.3	67	$P_{s4.5}$
	3.0	9	P_{s2}, Scroll
	3.9	10	P_{s2}, Scroll
	5.4	108	Enc. to Igniter
	7.35	68	P_{t7}
	10.5	71	A/B Fuel Pressure
	14.5	72	Ignition Squirt Pressure
13	1.3	85	Power Lever
	3.0	102	Ignition
	3.9	105	P_{t3}
	10.5	73	Ignition Control Pressure
	14.5	74	A/B Actuator Pressure
14	1.3	90	Ice Detector
	2.3	96	Bleed Valve Position
	5.4	83	Engine Fuel Flow
	7.35	84	A/B Fuel Flow
	10.5	82	N_2
	14.5	81	N_1

APPENDIX IV

DATA REDUCTION METHODS

APPENDIX IV

DATA REDUCTION METHODS

Data acquisition and reduction techniques followed basically the same methods employed in the Compressor Stall Investigation Program (Reference A). Briefly summarizing, data were obtained essentially from airborne magnetic tape recordings of measurements listed in Table III-1 and pilot comments recorded on tape and later transcribed into flight logs.

DATA CONVERSION

Following a quick-look analysis and check for instrumentation discrepancies, the flight tape was processed in analog form on Brush Recorders. Table IV-1 of this appendix outlines how measurements were grouped for Brush playbacks of the test data.

Playbacks were then reviewed in detail, and specific data areas were selected for further processing and evaluation. This involved reading analog traces at the selected time-slice intervals with the Benson-Lehner OSCAR J oscillograph reader which transfers data to punched cards for IBM 1130 computer processing. Required computations were performed and generated the flight history, system and engine parameters for intervals being investigated.

Compressor inlet pressure distribution maps were obtained for flight operations with each duct configuration to examine any effects of vanes on the pressure recovery patterns.

Information in this appendix includes the following:

(a) A summary of selected points of each data flight. (Table IV-2)

(b) Supplementary information; i.e., explanation of symbols used in the computer output, Table IV-3, and maneuver codes, Table IV-4.

(c) Because of the voluminous quantities of data obtained during the program, only one set of "typical" data is included in this appendix and is representative of the information which is contained in Volume II.

1. Tabulated output of selected data points of Flight 58. (Table IV-5)
2. Analog traces of snap-up maneuver (and compressor stall) of Flight 58. (Figure IV-1)

APPENDIX IV - DATA REDUCTION METHODS (Continued)

3. Compressor Inlet Pressure Recovery Distribution Map, prior to compressor stall of Flight 58. (Figure IV-2)

4. Plot of the deviation from average pressure recovery at each P_{t2} probe prior to compressor stall of Flight 58. (Figure IV-3)

5. Analog traces showing the drop of four T_{t3} measurements upon cloud entry, Flight 64, Figure IV-4, and Flight 72, Figure IV-5.

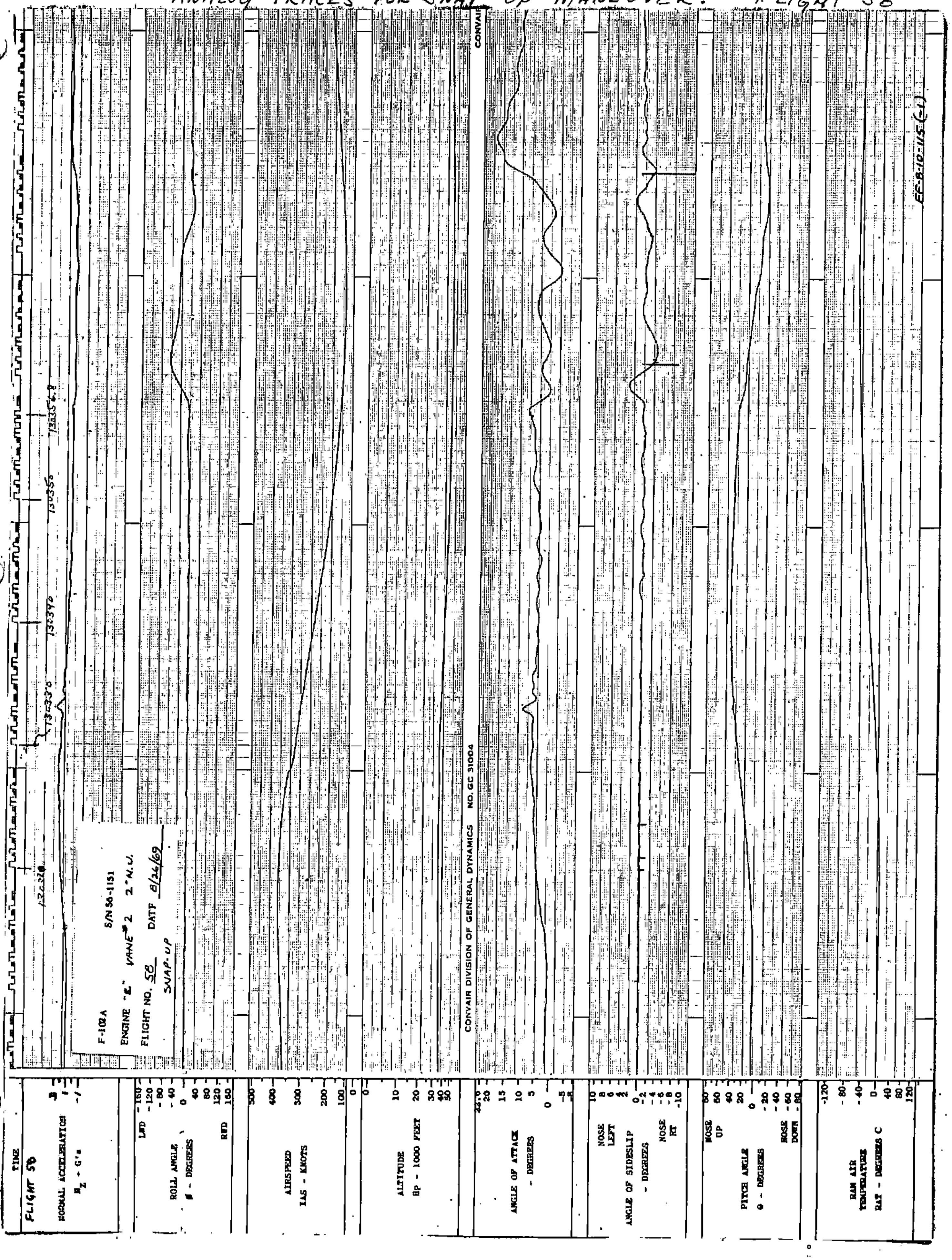
6. Schematic of P_{t2} probes, indicating sequence and time of probe filling (as indicated by analog trace frequency attenuation) during each cloud penetration of Flight 64, Figure IV-6, and Flight 72, Figure IV-7.

Volume II of this report has been compiled to present tabulations of aircraft and engine parameters of all data points selected for evaluation, together with associated analog trace cutouts, compressor face pressure distribution maps, probe damping schematics and other data pertinent to this program.

Volume II also contains the engine calibration and trim data developed during the program. It should be noted that the trim curve generated is unique to the engine and P_{t7} system.

FIGURE IV - 1

ANALOG TRACES FOR SNAP-UP MANEUVER: FLIGHT 58



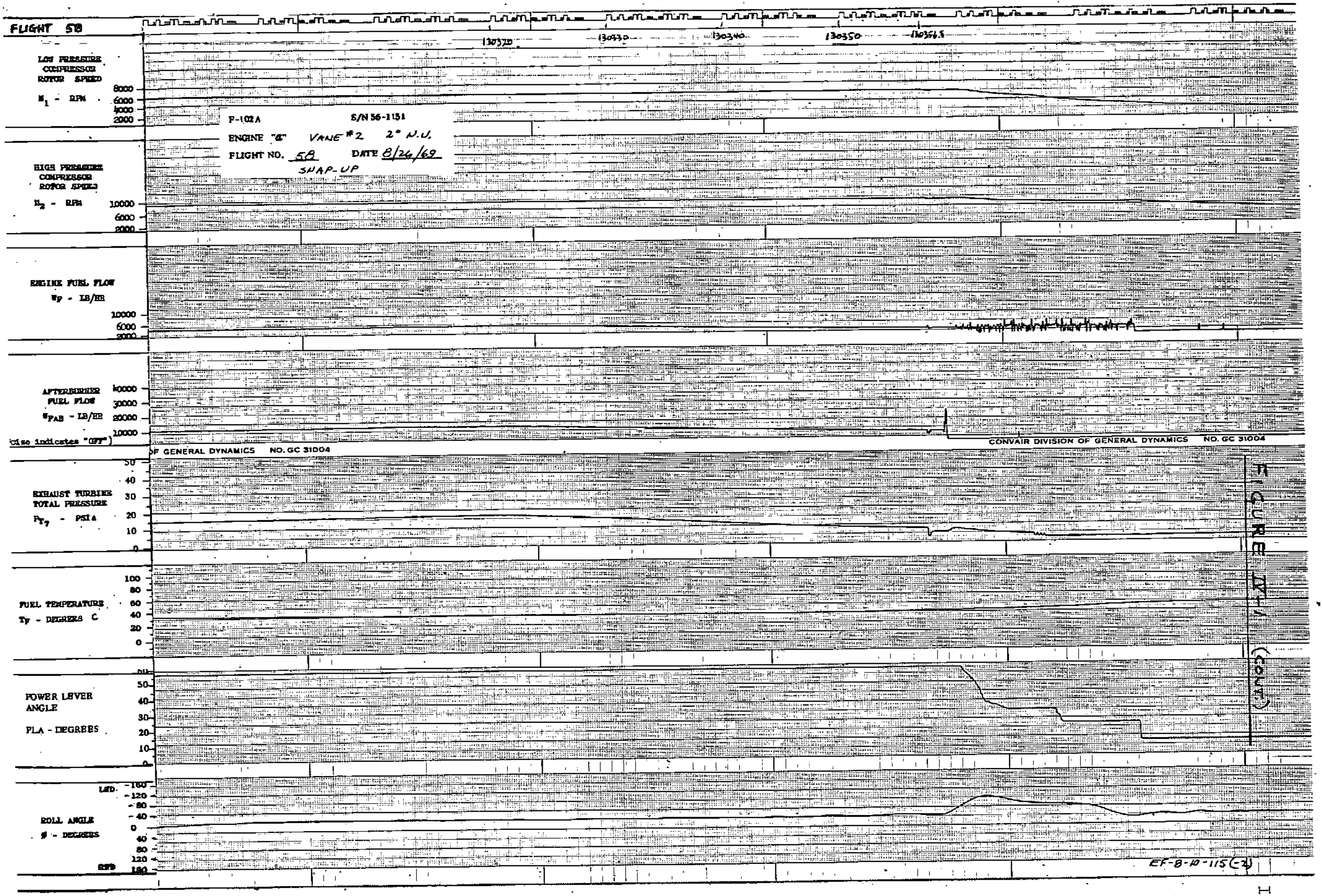


FIGURE IV-1 (CONT)

FIGURE IV-1 (CONT.)

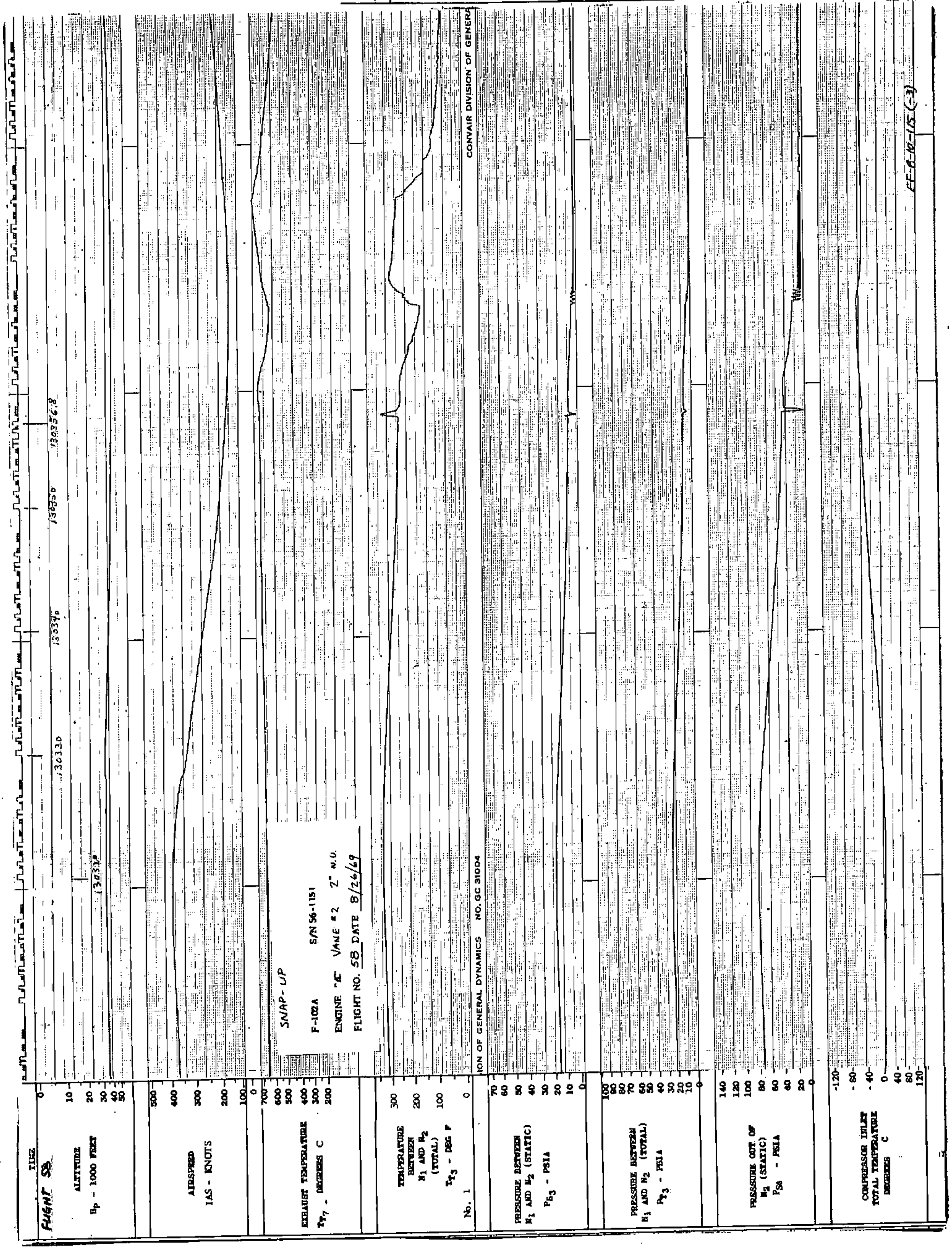
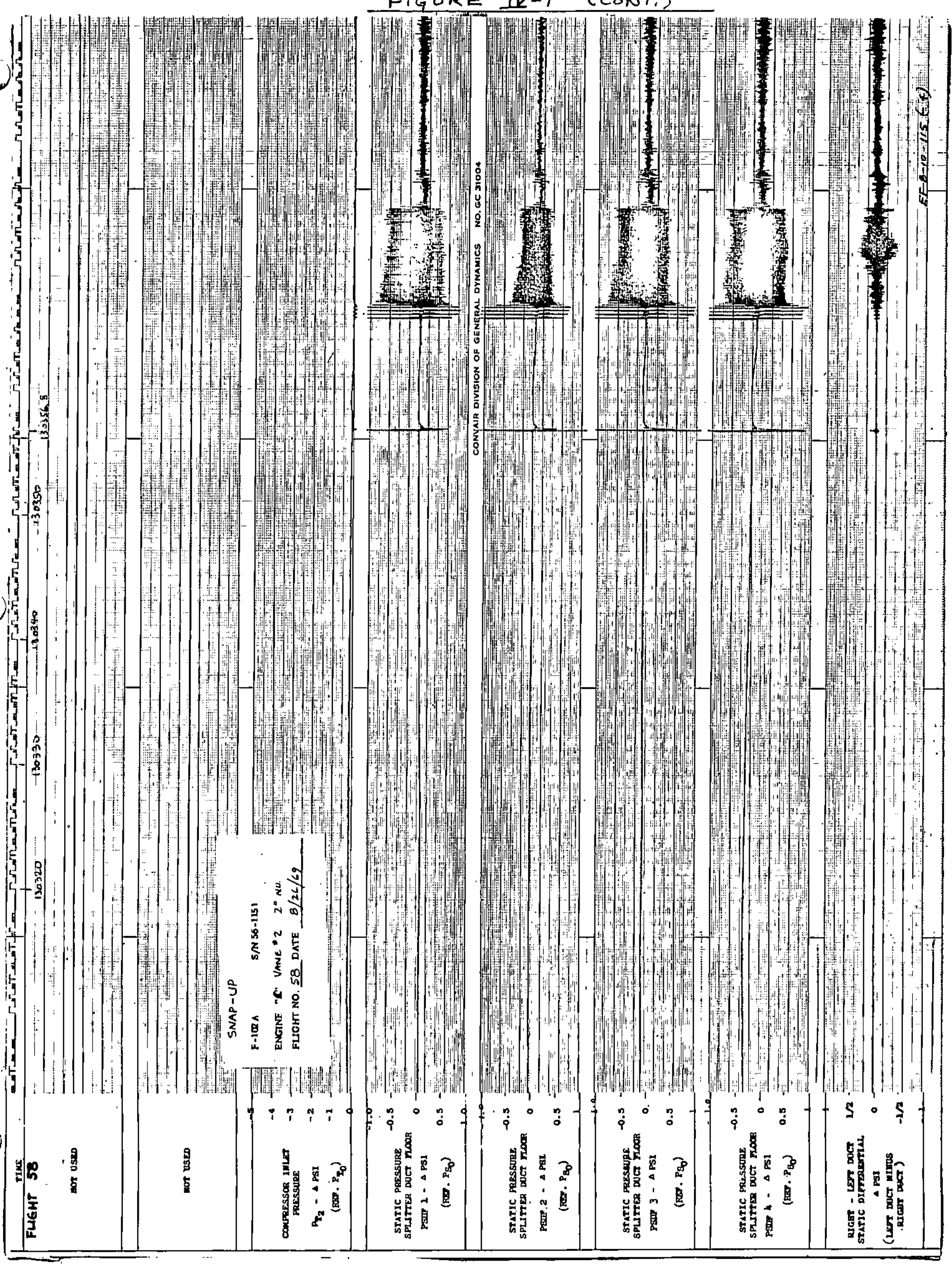
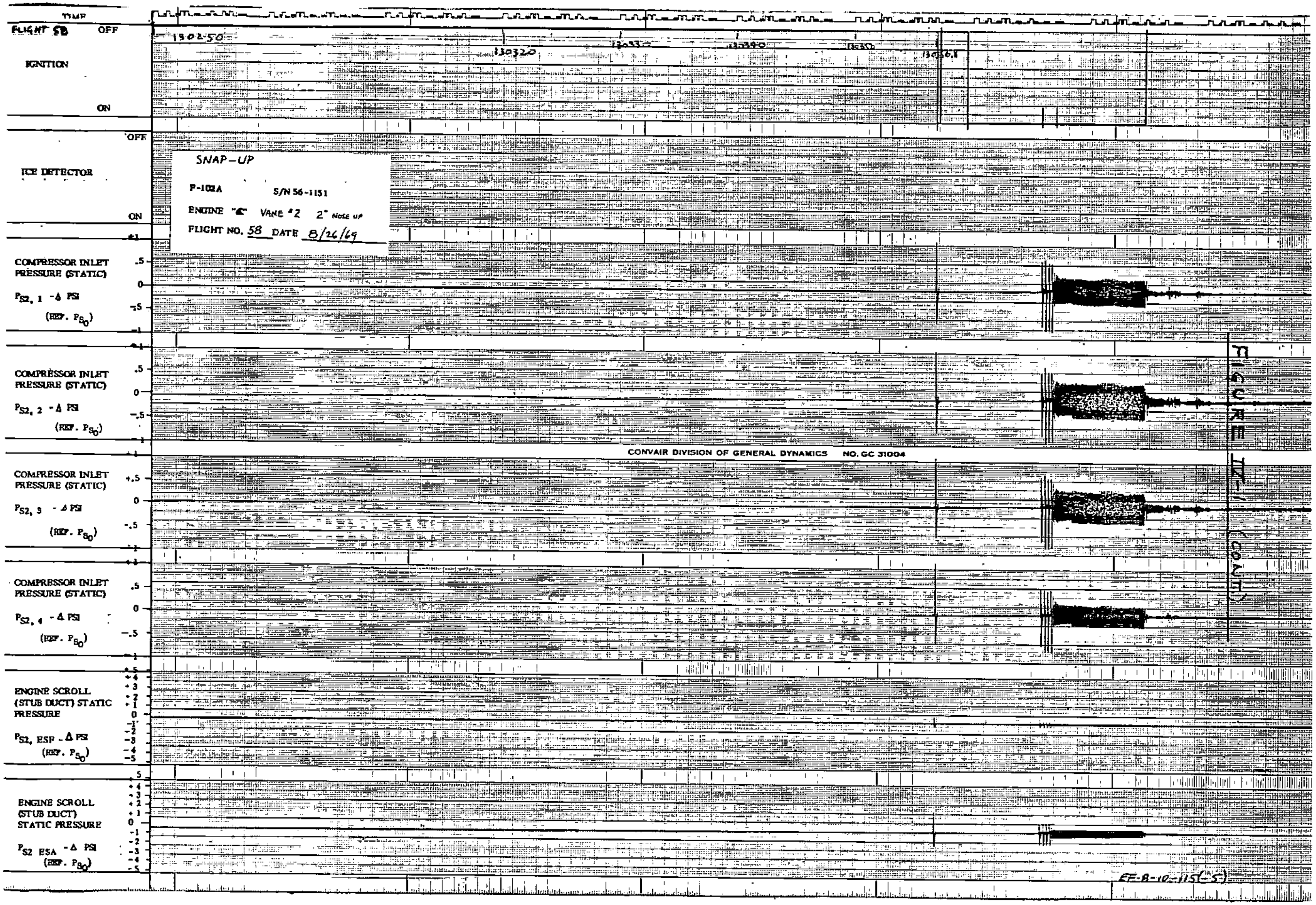


FIGURE IV-1 (CONT.)





SNAP-UP

F-102A S/N 56-1151

ENGINE "C" VANE #2 2° NOSE UP

FLIGHT NO. 58 DATE 8/26/69

FIGURE IV-1 (CONT.)

FIGURE IV-1 (CONCL.)

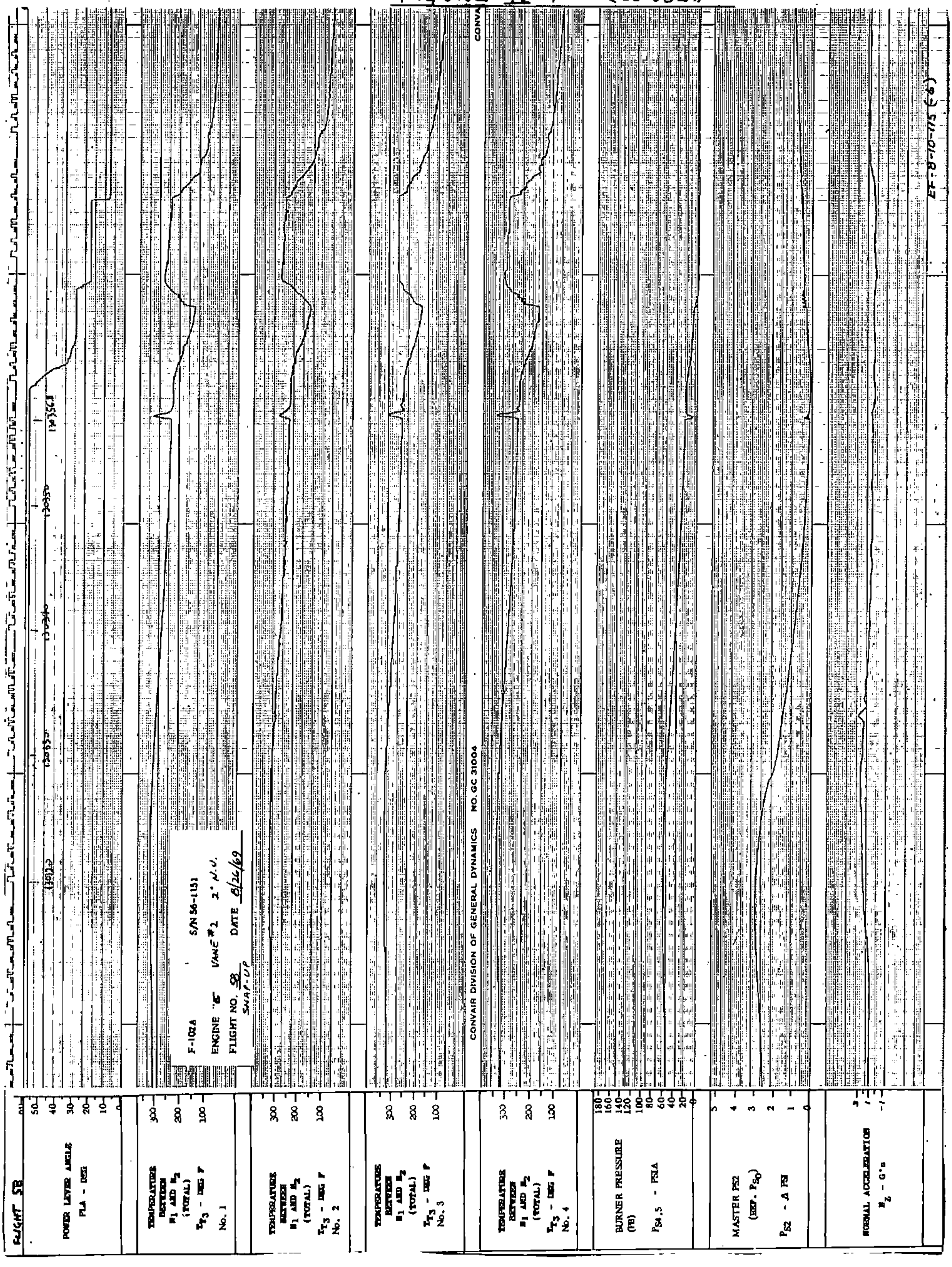


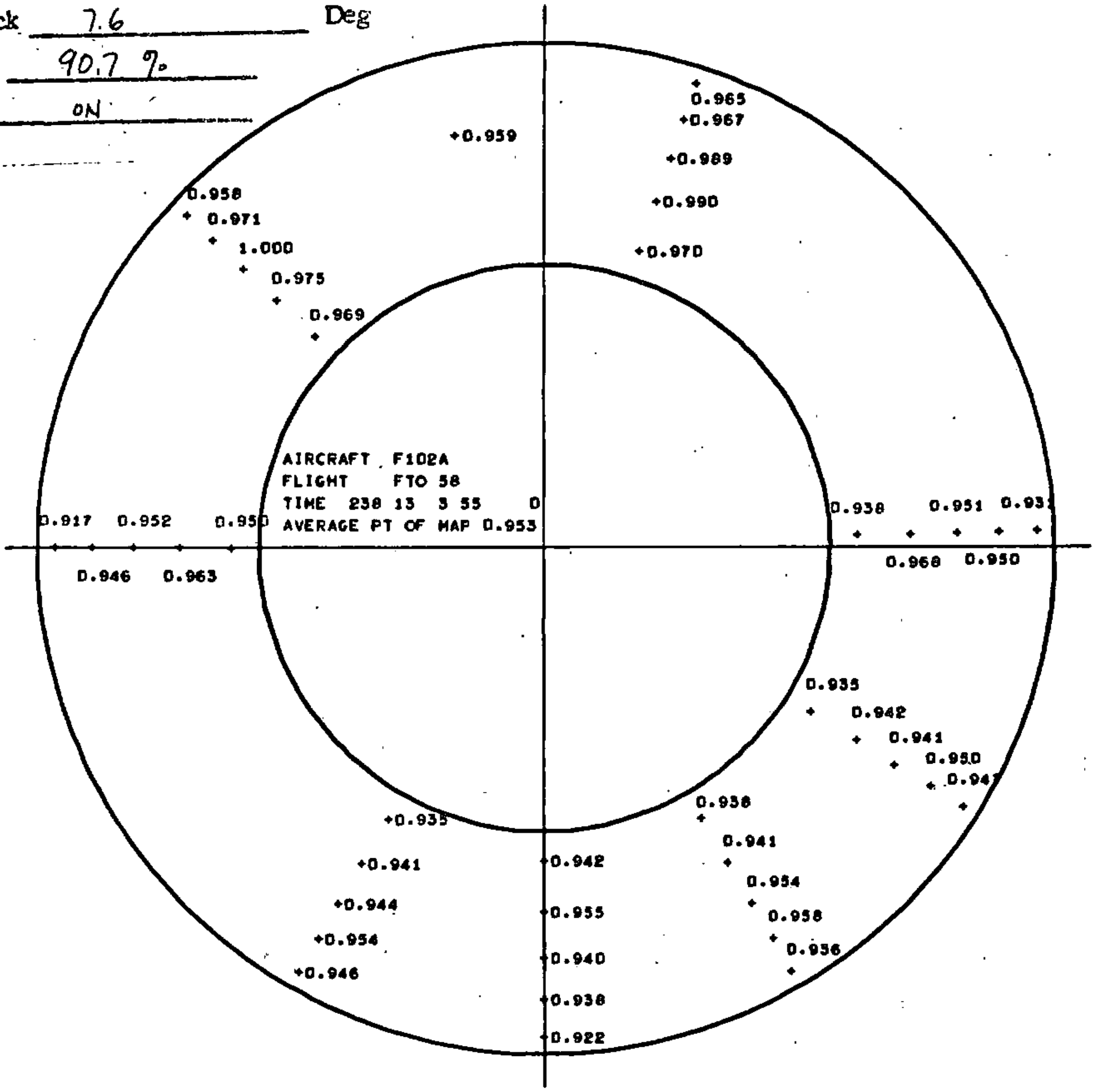
FIGURE IV-2

COMPRESSOR INLET TOTAL PRESSURE RATIO
DISTRIBUTION (PT2/PT0)

FLIGHT 58
PRIOR TO STALL

Pr. Altitude 45733 Ft.
Mach 0.613 CAS 158.2 Kt
Ambient Temp -62.0 Deg C
Angle of Attack 7.6 Deg
Power Setting 90.7 %
Afterburner ON

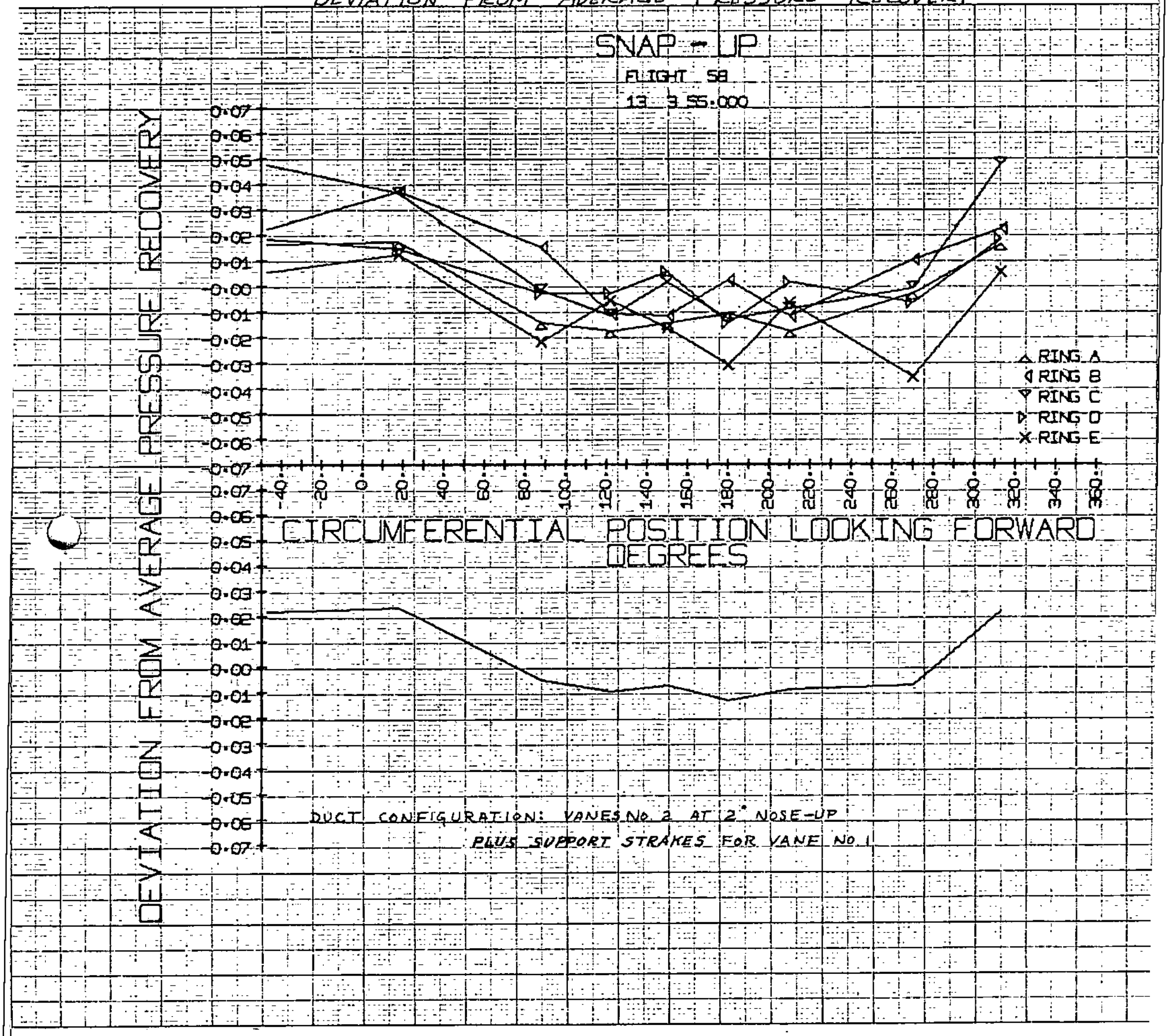
VANES # 2 AT 2° NOSE-UP AND
SUPPORT STRAKES FOR VANE #1)



EF-8-10-114(-5)

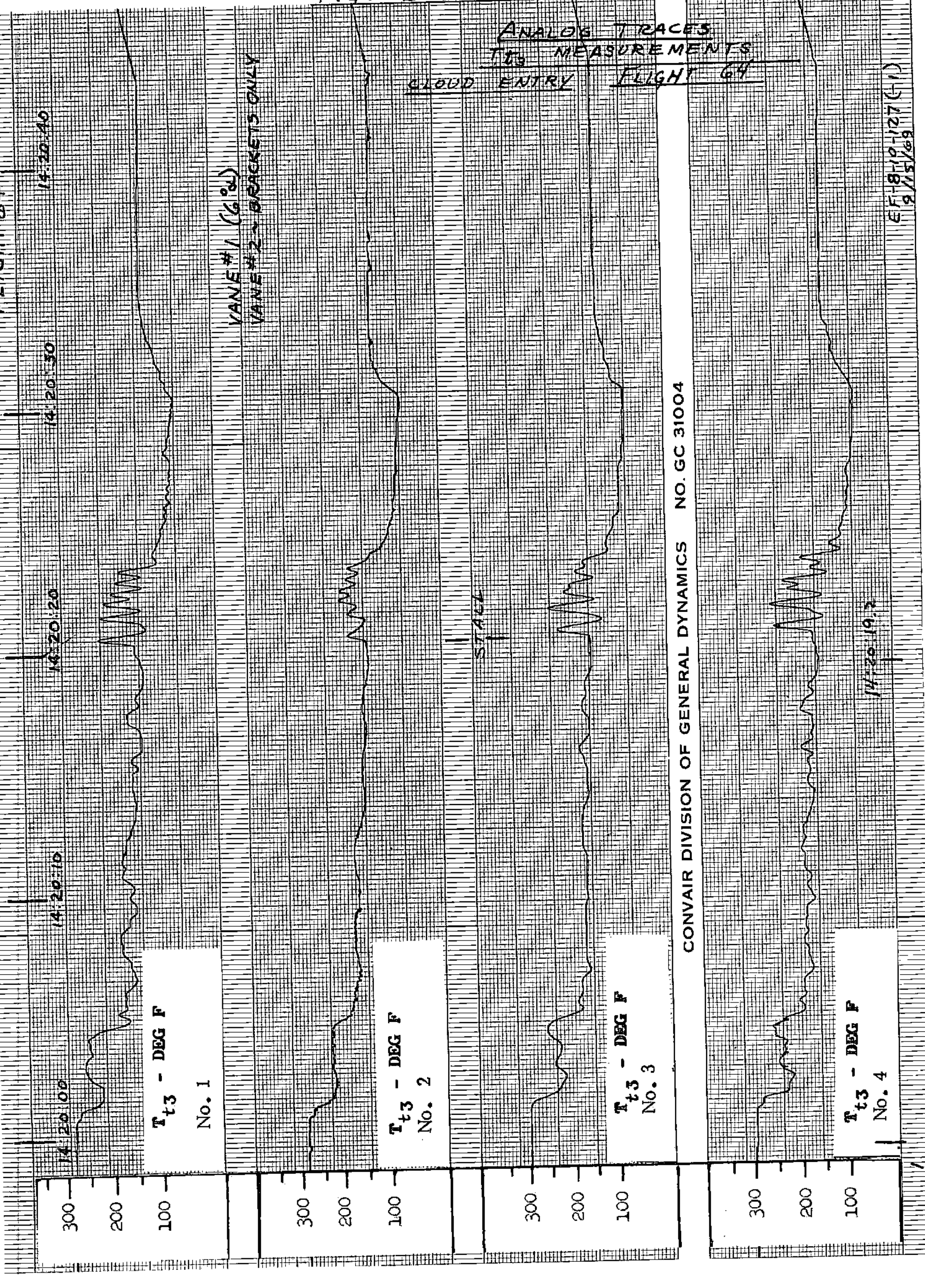
FIGURE IV - 3

DEVIATION FROM AVERAGE PRESSURE RECOVERY



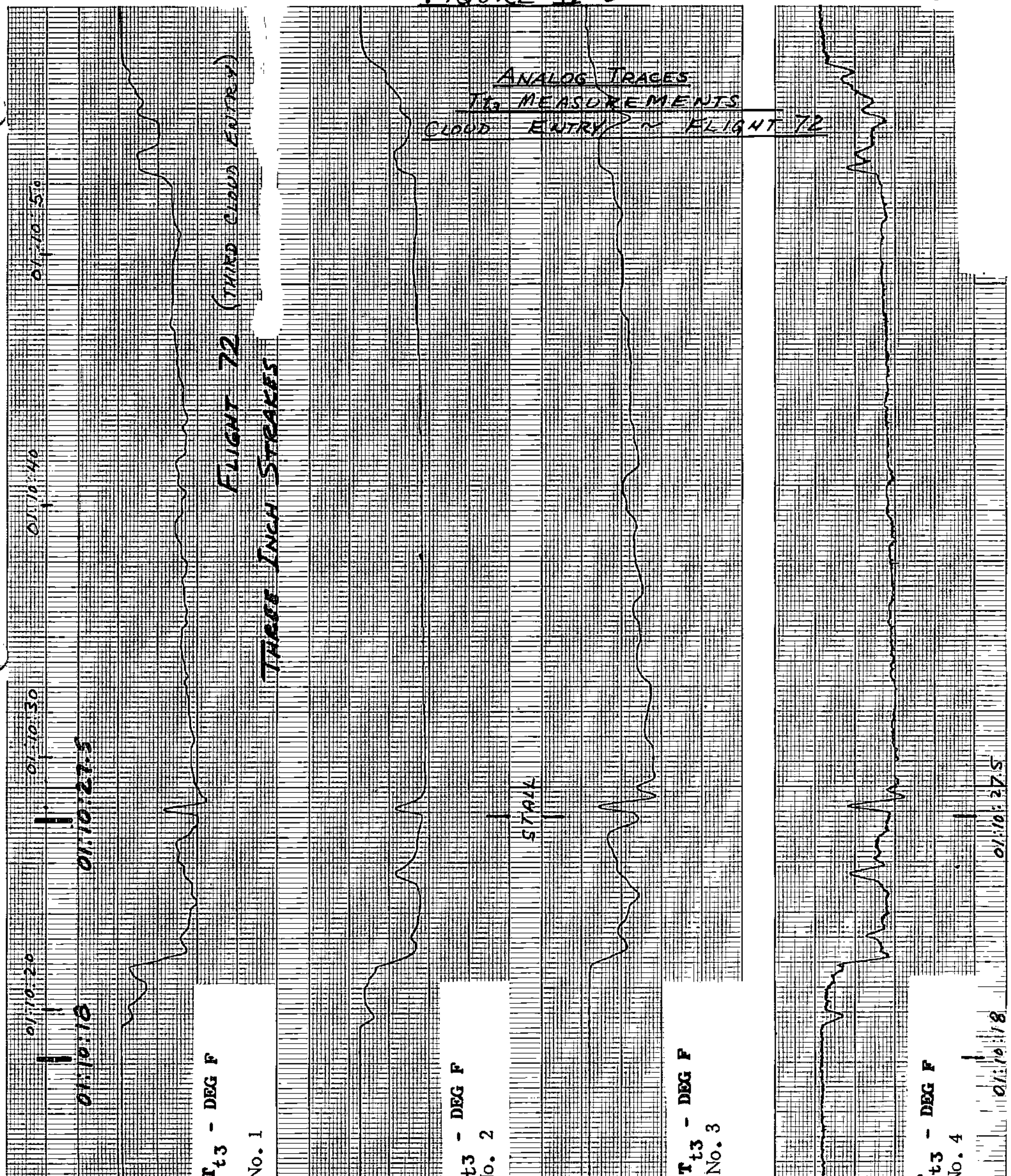
DUCT CONFIGURATION: VANES No. 2 AT 2° NOSE-UP
PLUS SUPPORT STRAKES FOR VANE No. 1

FIGURE IV-4



ANALOG TRACES
Tt3 MEASUREMENTS
CLOUD ENTRY FLIGHT 64

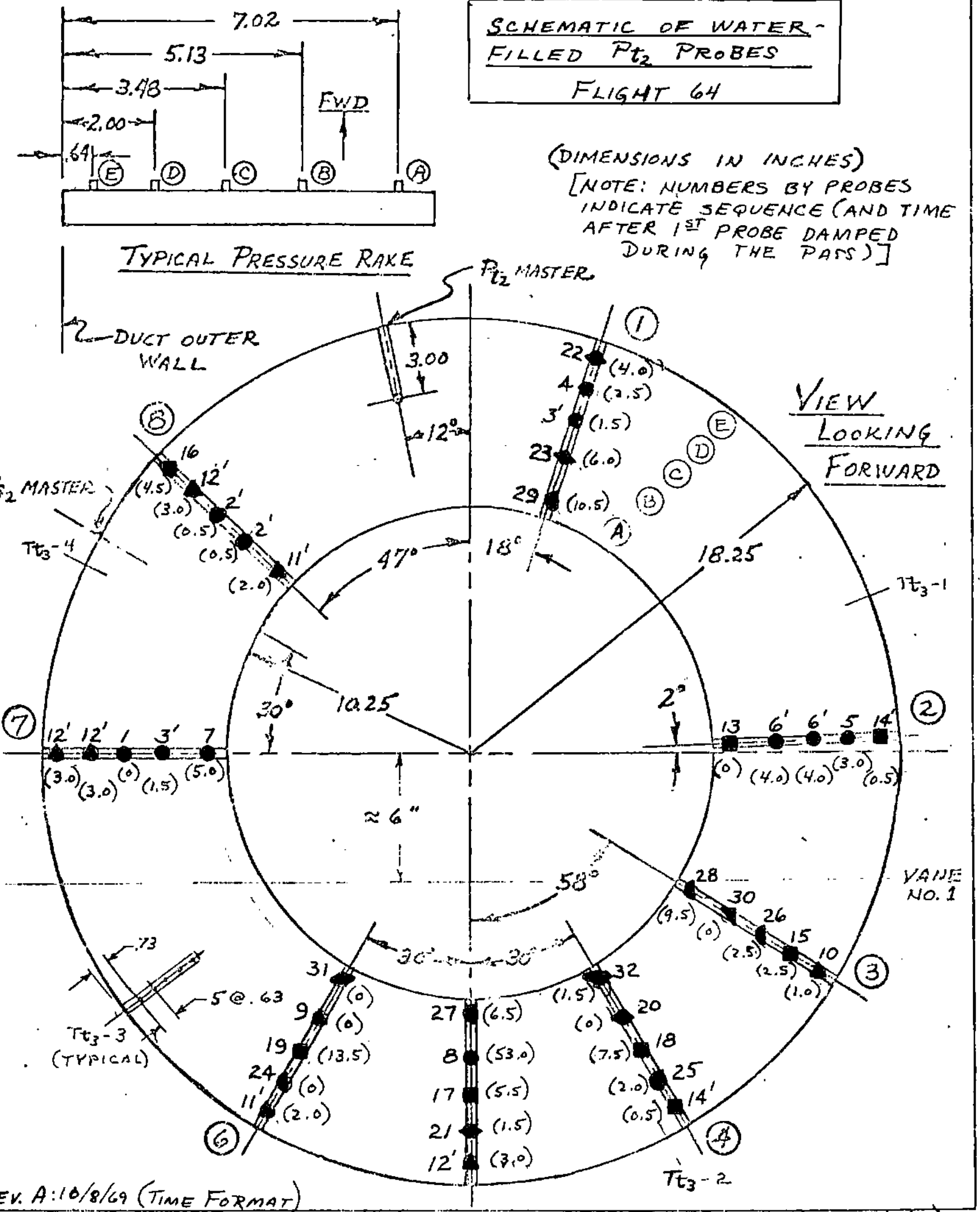
FIGURE IV-5


ANALOG TRACES
T_{t_3} MEASUREMENTS
CLOUD ENTRY ~ FLIGHT 72

EF-8-10-144(-3)

SYM	PENETRATION (& NUMBER OF PROBES DAMPED)		1ST SIGNIFICANT T_{t_3} DROP	TIME IN CLOUD	TIME 1ST PROBE DAMPED	Δ TIME FROM ENTRY
●	1ST	(11)	14:04:48.6	15 SEC.	14:04:55.5	6.9 SEC.
▲	2ND	(8)	14:05:53.4	5 "	14:05:54	0.6 "
■	3RD	(8)	14:08:41.6	16 "	14:08:43	1.4 "
◆	4TH	(4)	14:13:37.6	26 "	14:13:53	15.4 "
◖	5TH	(6)	14:14:28.4	24 "	14:14:31.5	3.1 "
◥	6TH	(1) STALL	14:20:00.8	≈ 30 "	14:20:27	26.2 "
▬	7TH	(2) STALL	14:26:58.8	≈ 20 "	14:27:04.5	5.7 "

FIGURE IV-6 RECOMMENDED DUCT MODIFICATION

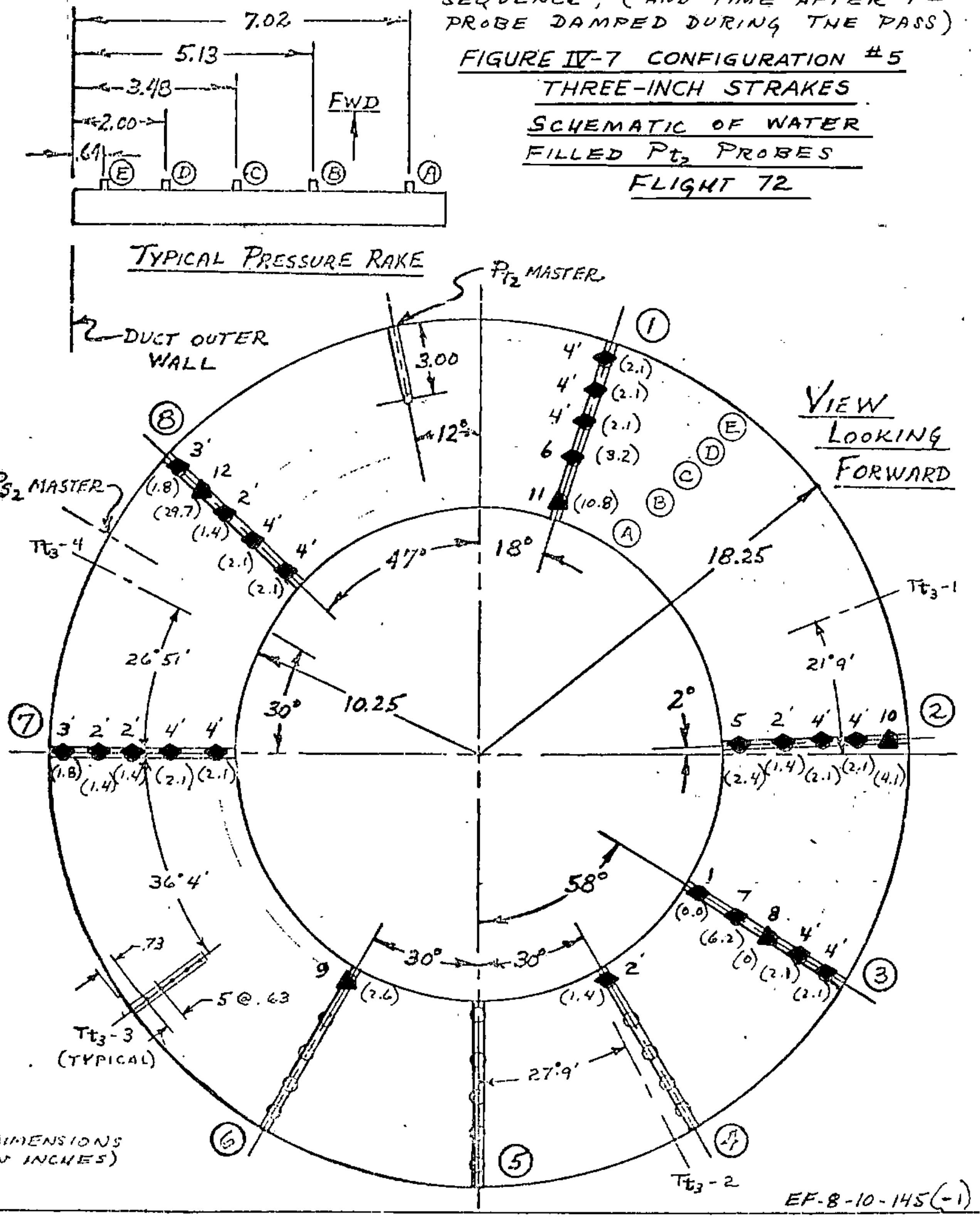


REV. A: 10/8/69 (TIME FORMAT)

SYM	PENETRATION (& No. OF PROBES DAMPED)	TIME IN CLOUD	1ST SIGNIFICANT Tt_3 DROP	TIME 1ST PROBE DAMPED	Δ TIME FROM ENTRY
■	1ST (0)	16 SEC.	01:03:51.5	——	——
◆	2ND (22)	28 SEC	01:06:22.6	01:06:43.6	21 SEC.
▲	3RD (5)	40 SEC.	01:10:19.2	01:10:22.2	3 SEC.

NOTE: PROBE 4D INOPERATIVE ~ NUMBERS BY PROBES INDICATE SEQUENCE, (AND TIME AFTER 1ST PROBE DAMPED DURING THE PASS)

FIGURE IV-7 CONFIGURATION #5
THREE-INCH STRAKES
SCHEMATIC OF WATER FILLED Pt_2 PROBES
FLIGHT 72



COMPRESSOR INLET TOTAL PRESSURE (Pt_2) PROBE LOCATIONS

10/8/69

(87)

TABLE IV-1

F-102 COMPRESSOR STALL PROGRAM

GROUPINGS FOR BRUSH RECORD PLAYBACKS OF TAPE DATA

EC O.	REC POS	TRK	CH	SCO	MEASUREMENT VALID FLIGHTS 49 THRU :	SYM	MEAS NO.	UNITS (AS MEAS.)
1	1	6	7	2.3	NORMAL ACCELERATION (REF ONLY)	NZ	93	G.S
1	2	6	6	1.7	ROLL ANGLE (REF ONLY)	ϕ	92	DEG
1	3	4	7	2.3	AIRSPEED	IAS	86	KNOTS
1	4	4	8	3.0	ALTITUDE	HP	87	PSIA
1	5	5	7	2.3	ANGLE OF ATTACK	α	88	DEG
1	6	5	8	3.0	ANGLE OF SIDESLIP	β	89	DEG.
1	7	6	5	1.3	PITCH ANGLE	θ	91	DEG
1	8	2	6	1.7	RAM AIR TEMPERATURE	RAT	77	DEG C
2	1	14	13	14.5	LOW PRESSURE COMPRESSOR ROTOR SPEED	N1	81	RPM
2	2	14	12	10.5	HIGH PRESSURE COMPRESSOR ROTOR SPEED	N2	82	RPM
2	3	14	10	5.4	ENGINE FUEL FLOW	WF	83	GPM
2	4	14	11	7.35	AFTERBURNER FUEL FLOW	WFAB	84	GPM
2	5	12	11	7.35	EXHAUST TURBINE TOTAL PRESSURE	PT7	68	PSIA
2	6	2	9	3.9	FUEL TEMPERATURE	TF	80	DEG C
2	7	13	5	1.3	POWER LEVER ANGLE	PLA	85	DEG
2	8	6	6	1.7	ROLL ANGLE	ϕ	92	DEG
3	1	4	8	3.0	ALTITUDE (REF ONLY)	HP	87	PSIA
3	2	4	7	2.3	AIRSPEED (REF ONLY)	IAS	86	KNOTS
3	3	2	8	3.0	EXHAUST TEMPERATURE	TT7	79	DEG C
3	4	2	5	1.3	TEMP. BETWEEN N1 AND N2 (2:00)	TT3-1	106	DEG F
3	5	12	6	1.7	PRESS. BETWEEN N1 AND N2 (STATIC)	PS3	66	PSIA
3	6	13	9	3.9	PRESS. BETWEEN N1 AND N2 (TOTAL)	PT3	105	PSIA
3	7	12	5	1.3	PRESS. OUT OF N2 (STATIC)	PS4	65	PSIA
3	8	2	7	2.3	COMPR. INLET TOTAL TEMPERATURE	TT2	78	DEG C
4	1				(NOT USED)			
4	2				(NOT USED)			
4	3	2	11	7.35	COMPR. INLET PRESS. (REF. TO A/S TOTAL)	PT2	69	ΔPSI
4	4	11	9	3.9				
4	5	11	10	5.4				
4	6	11	11	7.35	(SELECT 5 OF THE FOLLOWING 16) *			
4	7	11	12	10.5				
4	8	11	13	14.5				
					STATIC PRESSURE SPLITTER DUCT WALL (6)	PSSW	1-6	ΔPSI
					STATIC PRESSURE DUCT FLOOR (4)	PSDF	11-14	ΔPSI
					TOTAL PRESSURE BOUNDARY LAYER RAKE (5)	PTBL	60-64	ΔPSI
					DIFF. STATIC PRESS., LT-RT (1)	ΔPSI	109	ΔPSI
					(TOTAL OF 16)			

* FLIGHTS 49 THRU : MEASUREMENT NO.S 3 THRU 6, AND 109

EF-8-10-87 (-1)
6/16/69

TABLE IV-1 (CONT.)

F-102 COMPRESSOR STALL PROGRAM

GROUPINGS FOR BRUSH RECORD PLAYBACKS OF TAPE DATA (CONTINUED)

REC O.	REC POS	TRK	CH	SCO	MEASUREMENT	SYM	MEAS NO.	UNITS (AS MEAS.)
5	1	13	8	3.0	IGNITION	--	102	ON/OFF
5	2	14	5	1.3	ICE DETECTOR	--	90	ON/OFF
5	3	10	7	2.3	COMPRESSOR INLET PRESSURE (STATIC)	PS2 1	56	ΔPSI
5	4	10	8	3.0	COMPRESSOR INLET PRESSURE (STATIC)	PS2 2	57	ΔPSI
5	5	9	8	3.0	COMPRESSOR INLET PRESSURE (STATIC)	PS2 3	58	ΔPSI
5	6	9	7	2.3	COMPRESSOR INLET PRESSURE (STATIC)	PS2 4	59	ΔPSI
5	7	12	8	3.0	ENGINE SCROLL(STUB DUCT)STATIC PRESSURE	PS2 ESF	9	ΔPSI
5	8	12	9	3.9	ENGINE SCROLL(STUB DUCT)STATIC PRESSURE	PS2 ESA	10	ΔPSI
6	1	13	5	1.3	POWER LEVER ANGLE (REF ONLY)	PLA	85	DEG
6	2	2	5	1.3	TEMP. BETWEEN N1 AND N2 (2:00)(REF)	TT3-1	106	DEG F
6	3	4	5	1.3	TEMP. BETWEEN N1 AND N2 (5:00)	TT3-2	111	DEG F
6	4	4	6	1.7	TEMP. BETWEEN N1 AND N2 (8:00)	TT3-3	112	DEG F
6	5	2	10	5.4	TEMP. BETWEEN N1 AND N2 (10:00)	TT3-4	110	DEG F
6	6	12	7	2.3	BURNER PRESSURE	PS4.5	67	PSIA
6	7	2	12	10.5	MASTER PS2 REFERENCE (DIFF. PSO)	PS2M	70	ΔPSI
6	8	6	7	2.3	NORMAL ACCELERATION	NZ	93	G.S

THE 40 INDIVIDUAL RAKE PRESSURES,(ΔP REF. TO PT2 MASTER), PLAYED BACK AS FOLLOWS.

ECORD NO. 7	RAKES 1 THRU 4, RINGS A AND B	(REF. FIG.)
ECORD NO. 8	RAKES 5 THRU 8, RINGS A AND B	
RECORD NO. 9	RAKES 1 THRU 4, RINGS C AND D	
RECORD NO. 10	RAKES 5 THRU 8, RINGS C AND D	
RECORD NO. 11	RAKES 5 THRU 8, RING E	
RECORD NO. 12	RAKES 1 THRU 4, RING E	

EF-8-10-87 (-2)
6/16/69

TABLE IV-2

F-102 COMPRESSOR STALL PROGRAM
SUMMARY OF DATA POINTS (FLIGHTS)

VANE NO. 1 - TWO DEGREES NOSE UP

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANEUVER
49	13 39 25.000	-8.	0.000	79.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
49	14 0 10.000	37642.	0.924	-57.4	7	CRUISE WITHOUT A/B
49	14 5 30.000	43593.	0.980	-89.8	8	CRUISE WITH A/B
49	14 6 30.000	46840.	0.963	-93.7	8	CRUISE WITH A/B
49	14 8 0.000	44074.	0.617	-85.0	82	HIGH ANGLE OF ATTACK-HIGH POWER
49	14 8 20.000	44484.	0.525	-83.0	82	HIGH ANGLE OF ATTACK-HIGH POWER
49	14 8 40.000	43611.	0.441	-78.5	82	HIGH ANGLE OF ATTACK-HIGH POWER
49	14 9 0.000	42793.	0.389	-78.9	82	HIGH ANGLE OF ATTACK-HIGH POWER
50	13 2 50.000	-54.	0.000	73.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
50	13 5 0.000	4635.	0.533	67.1	3	CLIMB WITHOUT A/B
50	13 18 30.000	41112.	1.007	-77.3	32	SNAP-UP CLIMB WITH A/B
50	13 19 0.000	37746.	1.126	-55.2	32	SNAP-UP CLIMB WITH A/B
50	13 19 20.000	35816.	1.122	-51.0	32	SNAP-UP CLIMB WITH A/B
50	13 19 30.000	37638.	1.026	-55.5	32	SNAP-UP CLIMB WITH A/B
50	13 19 40.000	42146.	0.818	-70.0	32	SNAP-UP CLIMB WITH A/B
50	13 19 45.800	46950.	0.727	-79.3	9	PRIOR TO STALL
51	13 21 10.000	1.	0.000	76.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
51	13 58 30.000	38229.	0.842	-60.0	20	PRIOR TO CLOUD ENTRY
51	13 59 20.000	37849.	0.851	-56.9	21	CLOUD ENTRY/TURBULENCE
51	14 4 20.000	37348.	0.843	-57.2	20	PRIOR TO CLOUD ENTRY
51	14 4 47.500	37361.	0.850	-57.2	21	CLOUD ENTRY/TURBULENCE
51	14 6 20.000	37363.	0.851	-57.8	20	PRIOR TO CLOUD ENTRY
51	14 6 35.500	37057.	0.840	-58.8	21	CLOUD ENTRY/TURBULENCE
51	14 10 30.000	36809.	0.859	-55.9	20	PRIOR TO CLOUD ENTRY
51	14 10 34.600	36895.	0.855	-54.5	21	CLOUD ENTRY/TURBULENCE
51	14 18 40.000	38322.	0.838	-66.1	20	PRIOR TO CLOUD ENTRY
51	14 19 7.800	38624.	0.837	-66.4	21	CLOUD ENTRY/TURBULENCE
51	14 20 20.000	37944.	0.849	-64.6	20	PRIOR TO CLOUD ENTRY
51	14 20 45.800	38292.	0.877	-65.1	21	CLOUD ENTRY/TURBULENCE
51	14 25 40.000	37607.	0.879	-62.1	20	PRIOR TO CLOUD ENTRY
51	14 25 56.000	37635.	0.894	-64.3	21	CLOUD ENTRY/TURBULENCE
52	13 45 50.000	-26.	0.000	75.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
52	14 31 0.000	48242.	0.899	-105.9	19	CRUISE IN ICE CRYSTAL WEATHER
52	14 34 30.000	41548.	0.834	-85.2	20	PRIOR TO CLOUD ENTRY
52	14 36 11.400	41277.	0.841	-81.4	21	CLOUD ENTRY/TURBULENCE
52	14 39 45.000	38683.	0.717	-71.2	20	PRIOR TO CLOUD ENTRY
52	14 39 58.000	39545.	0.731	-75.7	21	CLOUD ENTRY/TURBULENCE
52	14 40 19.000	39313.	0.718	-69.6	20	PRIOR TO CLOUD ENTRY
52	14 40 33.000	39313.	0.718	-72.6	9	PRIOR TO STALL

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PAGE 1 OF 9

UPDATED 10/22/69

TABLE IV-2 (CONT.)

F-102 COMPRESSOR STALL PROGRAM
SUMMARY OF DATA POINTS (FLIGHTS)

VANE NO. 1 - FOUR DEGREES NOSE UP

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANEUVER
53	13 17 20.000	-8.	0.000	75.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
53	14 5 30.000	43859.	0.886	-71.3	20	PRIOR TO CLOUD ENTRY
53	14 6 22.600	44659.	0.879	-68.0	21	CLOUD ENTRY/TURBULENCE
53	14 10 0.000	41487.	0.831	-79.2	20	PRIOR TO CLOUD ENTRY
53	14 10 17.000	41864.	0.796	-77.6	21	CLOUD ENTRY/TURBULENCE
53	14 13 0.000	40218.	0.610	-68.6	20	PRIOR TO CLOUD ENTRY
53	14 13 37.000	40437.	0.694	-69.7	21	CLOUD ENTRY/TURBULENCE
53	14 14 55.000	39801.	0.721	-68.9	20	PRIOR TO CLOUD ENTRY
53	14 15 0.400	39596.	0.731	-67.7	21	CLOUD ENTRY/TURBULENCE
53	14 16 50.000	39947.	0.746	-71.5	20	PRIOR TO CLOUD ENTRY
53	14 18 19.000	39398.	0.743	-69.8	21	CLOUD ENTRY/TURBULENCE

VANE NO. 1 - SIX DEGREES NOSE UP

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANEUVER
54	13 16 40.000	1.	0.000	79.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
54	14 1 30.000	42329.	0.866	-82.4	7	CRUISE WITHOUT A/B
54	14 2 0.000	43171.	0.835	-82.6	7	CRUISE WITHOUT A/B
54	14 5 10.000	40957.	0.812	-77.7	20	PRIOR TO CLOUD ENTRY
54	14 5 22.800	40988.	0.830	-76.4	21	CLOUD ENTRY/TURBULENCE
54	14 5 30.000	40694.	0.819	-78.8	21	CLOUD ENTRY/TURBULENCE
54	14 5 35.800	40328.	0.808	-73.5	21	CLOUD ENTRY/TURBULENCE
54	14 17 19.000	39808.	0.822	-72.1	20	PRIOR TO CLOUD ENTRY
54	14 17 37.000	39987.	0.800	-74.7	21	CLOUD ENTRY/TURBULENCE
54	14 20 0.000	39360.	0.808	-73.9	20	PRIOR TO CLOUD ENTRY
54	14 20 23.000	39783.	0.807	-73.0	21	CLOUD ENTRY/TURBULENCE
54	14 21 43.000	39788.	0.810	-72.0	20	PRIOR TO CLOUD ENTRY
54	14 21 54.000	39270.	0.817	-75.3	21	CLOUD ENTRY/TURBULENCE
54	14 47 0.000	35536.	0.391	-46.2	81	HIGH ANGLE OF ATTACK-LOW POWER
54	14 47 30.000	34400.	0.341	-46.0	81	HIGH ANGLE OF ATTACK-LOW POWER
54	14 48 10.000	30177.	0.497	-33.0	81	HIGH ANGLE OF ATTACK-LOW POWER
54	14 49 40.000	29132.	0.229	-26.4	81	HIGH ANGLE OF ATTACK-LOW POWER

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TABLE IV-2 (CONT.)

F-102 COMPRESSOR STALL PROGRAM
SUMMARY OF DATA POINTS (FLIGHTS)

VANE NO. 2 - ZERO DEGREES NOSE UP
(PLUS BRACKETS FOR VANE NO. 1)

55	13	54	50.000	121.	0.000	77.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
55	14	13	0.000	37365.	0.964	-65.9	8	CRUISE WITH A/B
55	14	14	0.000	37783.	0.952	-64.3	7	CRUISE WITHOUT A/B
55	14	17	30.000	36273.	0.849	-58.8	7	CRUISE WITHOUT A/B
55	14	26	30.000	38043.	0.628	-62.0	81	HIGH ANGLE OF ATTACK-LOW POWER
55	14	27	0.000	37720.	0.486	-56.1	81	HIGH ANGLE OF ATTACK-LOW POWER
55	14	27	30.000	38445.	0.356	-53.7	81	HIGH ANGLE OF ATTACK-LOW POWER
55	14	28	0.000	34868.	0.307	-47.3	81	HIGH ANGLE OF ATTACK-LOW POWER
56	15	12	40.000	121.	0.000	77.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
56	15	48	0.000	40416.	0.908	-76.5	7	CRUISE WITHOUT A/B
56	15	55	40.000	37137.	0.855	-68.0	20	PRIOR TO CLOUD ENTRY
56	15	55	47.200	37457.	0.818	-65.7	21	CLOUD ENTRY/TURBULENCE
56	16	2	52.000	36130.	0.863	-59.4	20	PRIOR TO CLOUD ENTRY
56	16	2	57.500	36380.	0.850	-65.0	21	CLOUD ENTRY/TURBULENCE
56	16	5	55.000	34581.	0.719	-55.8	20	PRIOR TO CLOUD ENTRY
56	16	5	56.800	35129.	0.724	-58.1	21	CLOUD ENTRY/TURBULENCE
56	16	6	27.000	35666.	0.745	-59.0	20	PRIOR TO CLOUD ENTRY
56	16	6	32.000	35372.	0.740	-59.8	21	CLOUD ENTRY/TURBULENCE
56	16	14	0.000	34794.	0.852	-55.8	20	PRIOR TO CLOUD ENTRY
56	16	14	14.500	35086.	0.834	-58.9	21	CLOUD ENTRY/TURBULENCE
56	16	17	5.000	35157.	0.663	-58.9	20	PRIOR TO CLOUD ENTRY
56	16	17	13.000	35375.	0.662	-59.7	21	CLOUD ENTRY/TURBULENCE
56	16	17	14.600	35370.	0.658	-60.2	9	PRIOR TO STALL

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TABLE IV-2 (CONT.)

F-102 COMPRESSOR STALL PROGRAM
SUMMARY OF DATA POINTS (FLIGHTS)

VANE NO. 2 - TWO DEGREES NOSE UP
(PLUS BRACKETS FOR VANE NO. 1)

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANEUVER
57	12 55 5.000	317.	0.000	75.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
57	13 32 10.000	37963.	0.803	-57.0	7	CRUISE WITHOUT A/B
57	13 38 10.000	42646.	0.820	-80.5	7	CRUISE WITHOUT A/B
57	13 39 58.000	41973.	0.858	-78.7	20	PRIOR TO CLOUD ENTRY
57	13 40 2.700	41958.	0.850	-78.1	21	CLOUD ENTRY/TURBULENCE
57	13 40 3.700	42258.	0.854	-78.1	21	CLOUD ENTRY/TURBULENCE
57	13 40 21.600	41896.	0.814	-75.8	9	PRIOR TO STALL
57	13 43 16.000	39646.	0.822	-69.4	20	PRIOR TO CLOUD ENTRY
57	13 43 22.000	40315.	0.831	-73.3	21	CLOUD ENTRY/TURBULENCE
57	13 43 23.800	39896.	0.842	-73.1	21	CLOUD ENTRY/TURBULENCE
57	13 43 26.000	40100.	0.834	-73.1	9	PRIOR TO STALL
57	13 43 30.000	40540.	0.832	-67.1	10	AFTER STALL
58	12 36 10.000	56.	0.000	76.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
58	13 3 20.000	35345.	1.099	-48.2	32	SNAP-UP CLIMB WITH A/B
58	13 3 30.000	36732.	1.008	-49.6	32	SNAP-UP CLIMB WITH A/B
58	13 3 40.000	40209.	0.861	-63.2	32	SNAP-UP CLIMB WITH A/B
58	13 3 50.000	44447.	0.716	-73.5	32	SNAP-UP CLIMB WITH A/B
58	13 3 56.800	45733.	0.613	-79.6	9	PRIOR TO STALL
58	13 9 25.000	47226.	0.960	-93.2	7	CRUISE WITHOUT A/B
58	13 9 29.800	45966.	0.893	-87.8	85	ROLLER COASTER
58	13 9 34.000	46108.	0.870	-89.8	85	ROLLER COASTER
58	13 9 49.600	46536.	0.870	-87.7	85	ROLLER COASTER
58	13 9 52.000	46960.	0.863	-87.0	85	ROLLER COASTER
58	13 10 9.000	45257.	0.893	-89.1	85	ROLLER COASTER
58	13 12 27.000	48422.	0.779	-93.1	75	CONSTANT ALTITUDE TURN
58	13 12 57.000	44354.	0.644	-84.1	75	CONSTANT ALTITUDE TURN
58	13 25 0.000	39068.	0.698	-69.9	81	HIGH ANGLE OF ATTACK-LOW POWER
58	13 26 0.000	37453.	0.423	-57.0	81	HIGH ANGLE OF ATTACK-LOW POWER
58	13 26 40.000	35930.	0.336	-51.3	81	HIGH ANGLE OF ATTACK-LOW POWER
58	13 27 0.000	35801.	0.274	-46.9	81	HIGH ANGLE OF ATTACK-LOW POWER

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TABLE IV-2 (CONT.)

F-102 COMPRESSOR STALL PROGRAM
SUMMARY OF DATA POINTS (FLIGHTS)

NO VANE INSTALLED
VANE BRACKETS ONLY (VANES 1 AND 2)

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANEUVER
60	0 17 20.000	2316.	0.000	100.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
60	0 27 0.000	39863.	0.942	-67.1	7	CRUISE WITHOUT A/B
60	0 28 20.000	40459.	1.001	-78.2	32	SNAP-UP CLIMB WITH A/B
60	0 28 40.000	37851.	1.069	-66.9	32	SNAP-UP CLIMB WITH A/B
60	0 29 0.000	35388.	1.149	-54.5	32	SNAP-UP CLIMB WITH A/B
60	0 29 20.000	38704.	0.936	-61.2	32	SNAP-UP CLIMB WITH A/B
60	0 29 40.000	46364.	0.710	-79.1	32	SNAP-UP CLIMB WITH A/B
60	0 29 43.700	48302.	0.686	-80.2	9	PRIOR TO STALL
60	0 41 10.000	37141.	0.566	-60.4	81	HIGH ANGLE OF ATTACK-LOW POWER
60	0 41 40.000	36498.	0.465	-50.2	81	HIGH ANGLE OF ATTACK-LOW POWER
60	0 42 10.000	35337.	0.359	-45.6	81	HIGH ANGLE OF ATTACK-LOW POWER
60	0 42 26.000	34696.	0.295	-48.0	81	HIGH ANGLE OF ATTACK-LOW POWER
61	15 33 15.000	1.	0.000	79.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
61	16 10 50.000	38695.	0.866	-67.2	20	PRIOR TO CLOUD ENTRY
61	16 10 55.300	38825.	0.880	-69.3	21	CLOUD ENTRY/TURBULENCE
61	16 17 20.000	35955.	0.839	-63.6	20	PRIOR TO CLOUD ENTRY
61	16 17 33.000	36328.	0.844	-64.2	21	CLOUD ENTRY/TURBULENCE
61	16 22 0.000	35962.	0.729	-58.7	20	PRIOR TO CLOUD ENTRY
61	16 22 14.000	36175.	0.750	-61.9	21	CLOUD ENTRY/TURBULENCE
61	16 25 49.000	35580.	0.714	-61.1	20	PRIOR TO CLOUD ENTRY
61	16 25 55.300	35958.	0.726	-60.9	21	CLOUD ENTRY/TURBULENCE
61	16 26 59.000	35849.	0.774	-60.2	20	PRIOR TO CLOUD ENTRY
61	16 27 2.300	36004.	0.757	-59.2	21	CLOUD ENTRY/TURBULENCE
61	16 27 13.300	35671.	0.831	-60.7	9	PRIOR TO STALL
61	16 27 17.800	35881.	0.795	-57.1	9	PRIOR TO STALL

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TABLE IV-2 (CONT.)

F-102 COMPRESSOR STALL PROGRAM
SUMMARY OF DATA POINTS (FLIGHTS)

VANE NO. 1 AT SIX DEGREES NOSE UP
(PLUS BRACKETS FOR VANE NO. 2)

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANEUVER
63	0 15 50.000	1100.	0.000	111.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
63	0 37 0.000	33603.	0.797	-44.0	20	PRIOR TO CLOUD ENTRY
63	0 37 16.200	34483.	0.781	-48.6	21	CLOUD ENTRY/TURBULENCE
63	0 37 22.800	34185.	0.785	-46.1	21	CLOUD ENTRY/TURBULENCE
63	1 12 20.000	32324.	0.887	-50.6	20	PRIOR TO CLOUD ENTRY
63	1 12 27.500	33101.	0.896	-47.9	21	CLOUD ENTRY/TURBULENCE
63	1 12 28.200	33096.	0.894	-49.5	21	CLOUD ENTRY/TURBULENCE
63	1 18 10.000	33665.	0.886	-47.6	20	PRIOR TO CLOUD ENTRY
63	1 18 20.200	33777.	0.893	-52.7	21	CLOUD ENTRY/TURBULENCE
63	1 26 20.000	31631.	0.909	-41.9	20	PRIOR TO CLOUD ENTRY
63	1 26 23.000	31529.	0.904	-50.3	21	CLOUD ENTRY/TURBULENCE
64	13 30 50.000	1.	0.000	77.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
64	14 4 40.000	36251.	0.933	-54.7	20	PRIOR TO CLOUD ENTRY
64	14 4 59.400	35576.	0.923	-62.1	21	CLOUD ENTRY/TURBULENCE
64	14 5 50.000	34772.	0.889	-47.3	20	PRIOR TO CLOUD ENTRY
64	14 5 56.900	34577.	0.894	-54.4	21	CLOUD ENTRY/TURBULENCE
64	14 8 40.000	35106.	0.716	-48.0	20	PRIOR TO CLOUD ENTRY
64	14 8 49.400	35807.	0.684	-57.0	21	CLOUD ENTRY/TURBULENCE
64	14 14 20.000	32135.	0.882	-45.9	20	PRIOR TO CLOUD ENTRY
64	14 14 40.000	32595.	0.879	-50.6	21	CLOUD ENTRY/TURBULENCE
64	14 19 50.000	33219.	0.739	-49.5	20	PRIOR TO CLOUD ENTRY
64	14 20 19.200	33389.	0.722	-53.0	9	PRIOR TO STALL
64	14 26 50.000	29839.	0.837	-39.8	20	PRIOR TO CLOUD ENTRY
64	14 27 7.200	29628.	0.859	-45.8	9	PRIOR TO STALL
65	12 50 20.000	1.	0.000	72.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
65	13 7 48.000	39923.	0.734	-64.6	7	CRUISE WITHOUT A/B
65	13 8 8.000	40065.	0.740	-63.4	62	SIDESLIP
65	13 8 38.000	39412.	0.742	-60.1	62	SIDESLIP
65	13 10 25.000	40426.	0.932	-64.2	7	CRUISE WITHOUT A/B
65	13 10 45.000	40431.	0.933	-64.8	62	SIDESLIP
65	13 10 47.800	40802.	0.926	-65.2	62	SIDESLIP
65	13 13 20.000	43232.	1.017	-78.6	6	DESCENT WITH A/B
65	13 13 50.000	36497.	1.192	-56.2	6	DESCENT WITH A/B
65	13 14 10.000	30132.	1.296	-36.0	6	DESCENT WITH A/B
65	13 14 22.000	23652.	1.199	9.5	6	DESCENT WITH A/B
65	13 16 10.000	34041.	0.543	-40.3	83	HIGH ANGLE OF ATTACK-MIL POWER
65	13 16 40.000	34465.	0.407	-41.9	83	HIGH ANGLE OF ATTACK-MIL POWER
65	13 17 5.000	34808.	0.273	-40.6	83	HIGH ANGLE OF ATTACK-MIL POWER
65	13 19 55.000	23213.	0.914	-0.8	7	CRUISE WITHOUT A/B
65	13 20 15.000	22480.	0.911	3.0	62	SIDESLIP
65	13 20 35.000	23137.	0.910	-0.3	62	SIDESLIP
65	13 21 56.000	24347.	0.709	-9.4	7	CRUISE WITHOUT A/B
65	13 22 6.000	24286.	0.715	-7.4	62	SIDESLIP
65	13 22 11.000	24426.	0.714	-5.1	62	SIDESLIP

TABLE IV-2 (CONT.)

F-102 COMPRESSOR STALL PROGRAM
SUMMARY OF DATA POINTS (FLIGHTS)

VANE NO. 2 BRACKETS ONLY

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANFUVER
66	14 46 50.000	-63.	0.000	73.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
66	15 7 50.000	40336.	0.967	-74.5	20	PRIOR TO CLOUD ENTRY
66	15 7 58.200	40513.	0.915	-77.7	21	CLOUD ENTRY/TURBULENCE
66	15 8 3.900	41362.	0.880	-80.2	21	CLOUD ENTRY/TURBULENCE
66	15 9 22.000	42004.	0.679	-68.7	76	WIND-UP TURN
66	15 9 30.000	40526.	0.684	-66.5	76	WIND-UP TURN
66	15 10 30.000	36397.	0.782	-61.0	20	PRIOR TO CLOUD ENTRY
66	15 10 45.900	36397.	0.782	-67.8	9	PRIOR TO STALL
66	15 16 28.000	37210.	0.501	-60.0	83	HIGH ANGLE OF ATTACK-MIL POWER
66	15 17 30.000	35603.	0.273	-53.2	83	HIGH ANGLE OF ATTACK-MIL POWER

FLIGHTS 67 AND 68
CLEAN DUCTS (ALL BRACKETS REMOVED)

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANFUVER
67	FERRY FLIGHT (NO DATA)					
68	0 12 40.000	1100.	0.000	93.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
68	0 38 40.000	28229.	0.772	-28.7	7	CRUISE WITHOUT A/B
68	0 38 50.000	28085.	0.782	-29.0	21	CLOUD ENTRY/TURBULENCE
68	0 41 52.000	26434.	0.776	-21.4	20	PRIOR TO CLOUD ENTRY
68	0 42 1.100	26779.	0.799	-24.7	21	CLOUD ENTRY/TURBULENCE
68	0 42 27.000	26589.	0.821	-20.5	20	PRIOR TO CLOUD ENTRY
68	0 42 31.800	26513.	0.821	-20.0	21	CLOUD ENTRY/TURBULENCE
68	0 43 40.000	26108.	0.762	-18.7	20	PRIOR TO CLOUD ENTRY
68	0 43 50.500	26638.	0.762	-21.6	21	CLOUD ENTRY/TURBULENCE
68	0 45 10.000	25570.	0.704	-18.4	20	PRIOR TO CLOUD ENTRY
68	0 45 19.600	25655.	0.711	-22.7	21	CLOUD ENTRY/TURBULENCE

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TABLE IV-2 (CONT.)

F-102 COMPRESSOR STALL PROGRAM
SUMMARY OF DATA POINTS (FLIGHTS)

3 INCH STRAKES (VANE 1 BRACKETS REMOVED)

FLT. NO.	TIME HR.MIN.SEC.	PR. ALT. FT.	MACH NO.	AMB. TEMP. DEG.F	MAN. NO.	TYPE OF MANEUVER
69	14 14 40.000	10.	0.000	71.0	51	ADVANCE THROTTLE
69	14 15 0.000	10.	0.000	71.0	53	RETARD THROTTLE
69	14 15 20.000	10.	0.000	71.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
69	15 26 15.000	29263.	0.703	-30.0	20	PRIOR TO CLOUD ENTRY
69	15 26 20.000	29408.	0.685	-33.1	21	CLOUD ENTRY/TURBULENCE
69	15 31 25.000	25832.	0.583	-16.2	20	PRIOR TO CLOUD ENTRY
69	15 31 28.500	25912.	0.588	-16.2	21	CLOUD ENTRY/TURBULENCE
69	15 33 43.000	25918.	0.708	-17.7	20	PRIOR TO CLOUD ENTRY
69	15 33 53.000	26521.	0.705	-24.3	21	CLOUD ENTRY/TURBULENCE
69	15 35 25.000	25834.	0.649	-17.7	20	PRIOR TO CLOUD ENTRY
69	15 35 29.600	26210.	0.646	-23.5	21	CLOUD ENTRY/TURBULENCE
69	15 37 5.000	25684.	0.586	-15.4	20	PRIOR TO CLOUD ENTRY
69	15 37 16.100	25769.	0.596	-19.0	21	CLOUD ENTRY/TURBULENCE
69	15 39 34.000	28318.	0.539	-26.0	20	PRIOR TO CLOUD ENTRY
69	15 39 37.800	28087.	0.556	-27.0	21	CLOUD ENTRY/TURBULENCE
69	15 41 16.000	26609.	0.596	-18.9	20	PRIOR TO CLOUD ENTRY
69	15 41 28.000	26237.	0.607	-19.0	21	CLOUD ENTRY/TURBULENCE
69	15 41 42.000	26338.	0.627	-19.5	20	PRIOR TO CLOUD ENTRY
69	15 41 54.000	27006.	0.663	-22.3	21	CLOUD ENTRY/TURBULENCE
69	15 42 17.000	26844.	0.714	-22.9	21	CLOUD ENTRY/TURBULENCE
70	FERRY FLIGHT (NO DATA)					
71	FERRY FLIGHT (NO DATA)					
72	0 14 50.000	5810.	0.000	76.0	1	PRIOR TO BRAKE RELEASE/ENGINE RUNS
72	1 3 50.000	35854.	0.767	-53.9	20	PRIOR TO CLOUD ENTRY
72	1 4 0.300	36393.	0.766	-59.7	21	CLOUD ENTRY/TURBULENCE
72	1 6 20.000	33022.	0.626	-41.2	20	PRIOR TO CLOUD ENTRY
72	1 6 24.600	33252.	0.655	-42.6	21	CLOUD ENTRY/TURBULENCE
72	1 6 47.100	32894.	0.677	-43.8	21	CLOUD ENTRY/TURBULENCE
72	1 10 18.000	32037.	0.672	-39.8	20	PRIOR TO CLOUD ENTRY
72	1 10 27.500	32629.	0.693	-41.0	9	PRIOR TO STALL
72	1 50 35.000	35419.	0.460	-55.1	81	HIGH ANGLE OF ATTACK-LOW POWER
72	1 51 0.000	34319.	0.376	-51.9	83	HIGH ANGLE OF ATTACK-MIL POWER
72	1 51 20.000	33959.	0.284	-51.8	83	HIGH ANGLE OF ATTACK-MIL POWER
73	FERRY FLIGHT (NO DATA)					

TABLE IV-2 (CONCL.)

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          F-102 COMPRESSOR STALL PROGRAM
                       SUMMARY OF DATA POINTS (FLIGHTS)

FLT.     TIME          PR.      MACH    AMB. MAN.      TYPE OF MANEUVER
NO.  HR.MIN.SEC.       ALT.      NO.    TEMP. NO.
                       FT.              DEG.F


                    1 1/2 INCH STRAKES (VANE 1 BRACKETS REMOVED)


74    13 23 20.000      84.    0.000    78.0    1 PRIOR TO BRAKE RELEASE/ENGINE RUNS
74    14 39 25.000   33149.    0.353   -47.7   83 HIGH ANGLE OF ATTACK-MIL POWER
74    14 39 40.000   32803.    0.280   -47.9   83 HIGH ANGLE OF ATTACK-MIL POWER

75    13 21  0.000      75.    0.000    76.0    1 PRIOR TO BRAKE RELEASE/ENGINE RUNS
75    14 10 17.000   29784.    0.825   -42.2   20 PRIOR TO CLOUD ENTRY
75    14 10 18.400   29613.    0.825   -39.9   20 PRIOR TO CLOUD ENTRY
75    14 10 21.800   29679.    0.814   -43.8   21 CLOUD ENTRY/TURBULENCE
75    14 32  8.700   29775.    0.820   -43.1   20 PRIOR TO CLOUD ENTRY
75    14 32 34.400   30516.    0.843   -46.4   21 CLOUD ENTRY/TURBULENCE
75    14 35  6.800   29287.    0.732   -48.6   20 PRIOR TO CLOUD ENTRY
75    14 35 23.700   29828.    0.748   -42.5   21 CLOUD ENTRY/TURBULENCE
75    14 35 25.800   29387.    0.794   -47.7   21 CLOUD ENTRY/TURBULENCE
75    14 36 52.800   27868.    0.866   -33.9   20 PRIOR TO CLOUD ENTRY
75    14 37  1.600   28259.    0.859   -44.0   21 CLOUD ENTRY/TURBULENCE
75    14 39 19.400   28984.    0.805   -39.9   20 PRIOR TO CLOUD ENTRY
75    14 39 25.300   29337.    0.814   -44.3   21 CLOUD ENTRY/TURBULENCE

76                 FERRY FLIGHT (NO DATA)


                   CLEAN INLET DUCTS (NO VANES, NO BRACKETS)

77, 78, 79         FERRY FLIGHTS (NO DATA)

80     0 15 10.000      46.    0.000    78.2    1 PRIOR TO BRAKE RELEASE/ENGINE RUNS
80     1 11 33.000   29109.    0.795   -22.5   20 PRIOR TO CLOUD ENTRY
80     1 11 38.900   29469.    0.809   -24.8   21 CLOUD ENTRY/TURBULENCE
80     1 13 12.000   29210.    0.636   -26.2   20 PRIOR TO CLOUD ENTRY
80     1 13 18.700   29911.    0.650   -26.2   21 CLOUD ENTRY/TURBULENCE
80     1 15 40.000   29452.    0.627   -26.3   20 PRIOR TO CLOUD ENTRY
80     1 15 53.700   30002.    0.654   -26.1   21 CLOUD ENTRY/TURBULENCE
80     1 18 20.000   29290.    0.696   -22.6   20 PRIOR TO CLOUD ENTRY
80     1 18 36.400   29073.    0.720   -24.8   21 CLOUD ENTRY/TURBULENCE
80     1 25 50.000   27472.    0.733   -17.0   20 PRIOR TO CLOUD ENTRY
80     1 25 55.800   28022.    0.728   -20.9   21 CLOUD ENTRY/TURBULENCE
80     1 28  9.000   27033.    0.751   -13.9   20 PRIOR TO CLOUD ENTRY
80     1 28 17.700   27116.    0.754   -22.0   21 CLOUD ENTRY/TURBULENCE
80     1 28 52.000   28075.    0.762   -16.3   20 PRIOR TO CLOUD ENTRY
80     1 29  0.300   28104.    0.779   -19.5   21 CLOUD ENTRY/TURBULENCE
80     1 30 33.000   28040.    0.740   -14.2   20 PRIOR TO CLOUD ENTRY
80     1 30 39.800   28186.    0.731   -18.2    9 PRIOR TO STALL

81, 82, 83,        FERRY FLIGHTS (NO DATA)
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TABLE IV-3

EXPLANATION OF SYMBOLS LISTED IN COMPUTER OUTPUT

CAS	CALIBRATED AIRSPEED (KNOTS)
TAS	TRUE AIRSPEED (KNOTS)
ALT	ALTITUDE (FEET)
MACH	MACH NUMBER
TA	AMBIENT TEMPERATURE (DEG C)
PSO	PRESSURE ALTITUDE (PSIA)
PTO	FREE STREAM TOTAL PRESSURE (PSIA)
RNI	REYNOLDS NUMBER INDEX = $(\delta_{T_2} / \theta_{T_2}^{1.24})$
PLA	POWER LEVER ANGLE (DEG)
OV BL	OVERBOARD BLEED VALVE (OPEN/CLOSED)
A/B NOZ	AFTERBURNER NOZZLE POSITION (OPEN/CLOSED)
T FUEL	FUEL TEMPERATURE (DEG F)
RAT	RAM AIR TEMPERATURE (DEG C)
TT2	LOW COMPRESSOR INLET TEMPERATURE (DEG C AND F)
TT7	TURBINE OUTLET TEMPERATURE (DEG C AND F)
N1	LOW PRESSURE COMPRESSOR SPEED (RPM)
N2	HIGH PRESSURE COMPRESSOR SPEED (RPM)
TT3-(1,2 3,4)	LOW COMPRESSOR DISCHARGE TEMPERATURE (DEG F) (4 POSITIONS, NO. 1 AT 2:00, NO. 2 AT 5:00, NO. 3 AT 8:00, NO. 4 AT 10:00)(LOOKING FORWARD)

TT2/TSL =TR1 = θ_{T_2} =COMPRESSOR INLET TEMP/STD SEA LEVEL TEMP
TH2 .5 =TR2 = $\sqrt{\theta_{T_2}}$
TH2 .68 =TR3 = $\theta_{T_2}^{.68}$
TH2 1.24 =TR4 = $\theta_{T_2}^{1.24}$
TT3/TSL.5=TR5 = $\sqrt{\theta_{T_3}}$ = AVG. LOW COMPR.DISCHARGE TEMP/STD SEA LEVEL TEMP

PT2 LOW COMPRESSOR INLET TOTAL PRESSURE (PSIA)
δT2 =PT2/PSL
TT3/TR1 = T_{T_3}/θ_{T_2} (AVERAGE TT3)
TT7/TR1 = T_{T_7}/θ_{T_2}
N1/TR2 = $N_1/\sqrt{\theta_{T_2}}$
N2/TR2 = $N_2/\sqrt{\theta_{T_2}}$
N2/TR5 = $N_2/\sqrt{\theta_{T_3}}$

WF = W_f =ENGINE FUEL FLOW (LB/HR)
WF A/B = $W_{f(A/B)}$ =AFTERBURNER FUEL FLOW (LB/HR)
WFC.5 = $W_f/(\delta_{T_2} \cdot \sqrt{\theta_{T_2}})$
WFC.68 = $W_f/(\delta_{T_2} \cdot \theta_{T_2}^{.68})$
WFABC.68 = $W_{f(A/B)}/(\delta_{T_2} \cdot \theta_{T_2}^{.68})$
WF/PS4 = W_f/P_{S4}
WF4/TR3 = $W_f/(P_{S4} \cdot \theta_{T_2}^{.68})$
WF/PS4.5 = $W_f/P_{S4.5}$
WF/4.5TR3= $W_f/(P_{S4.5} \cdot \theta_{T_2}^{.68})$

PS3	STATIC PRESSURE AT LOW COMPRESSOR DISCHARGE (PSIA)
PT3	TOTAL PRESSURE AT LOW COMPRESSOR DISCHARGE (PSIA)
PS4	STATIC PRESSURE AT HIGH COMPRESSOR DISCHARGE (PSIA)
PS4.5	STATIC PRESSURE (BURNER)
PT7	TOTAL PRESSURE AT TURBINE OUTLET (PSIA)
PT7-ALT	=PT7-(ALTITUDE IN PSIA)

PRESSURE RATIOS ARE SELF EXPLANATORY

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TABLE IV-3 (CONT.)

	EXPLANATION OF SYMBOLS LISTED IN COMPUTER OUTPUT (PART II)		
CAS	CALIBRATED AIRSPEED (KNOTS)		
TAS	TRUE AIRSPEED (KNOTS)		
ALT	ALTITUDE (FEET)		
MACH	MACH NUMBER		
TA	AMBIENT TEMPERATURE (DEG F)		
PS0	PRESSURE ALTITUDE (PSIA)		
PT0	FREE STREAM TOTAL PRESSURE (PSIA)		
TT2	LOW COMPRESSOR INLET TEMPERATURE (DEG K)		
PS2M	(MASTER PS2 REFERENCE)STATIC PRESSURE AT LOW COMP. INLET(STA 460)300°		
		LOOKING FORWARD	
PS2 1	STATIC PRESSURE AT COMPRESSOR INLET	45 DEG CW	(PSIA) (OBTAINED FROM DIFF. PRESSURE MEASUREMENTS.REF TO MASTER PS2)
PS2 2	STATIC PRESSURE AT COMPRESSOR INLET	135 DEG CW	
PS2 3	STATIC PRESSURE AT COMPRESSOR INLET	225 DEG CW	
PS2 4	STATIC PRESSURE AT COMPRESSOR INLET	317 DEG CW	
PS2 ESF	STATIC PRESSURE AT ENGINE SCROLL	STA 439 (FWD)	
PS2 ESA	STATIC PRESSURE AT ENGINE SCROLL	STA 442 (AFT)	
PSSW1	STATIC PRESSURE AT SPLITTER DUCT WALL	STA 128	
PSSW2	STATIC PRESSURE AT SPLITTER DUCT WALL	STA 170	
PSSW3	STATIC PRESSURE AT SPLITTER DUCT WALL	STA 203	
PSSW4	STATIC PRESSURE AT SPLITTER DUCT WALL	STA 231	
PSSW5	STATIC PRESSURE AT SPLITTER DUCT WALL	STA 273	
PSSW6	STATIC PRESSURE AT SPLITTER DUCT WALL	STA 316	
PSDF1	STATIC PRESSURE AT DUCT FLOOR	STA 195	
PSDF2	STATIC PRESSURE AT DUCT FLOOR	STA 260	
PSDF3	STATIC PRESSURE AT DUCT FLOOR	STA 292	
PSDF4	STATIC PRESSURE AT DUCT FLOOR	STA 349	
PTBL1	BOUNDARY LAYER RAKE PRESSURE	(STA 107) 0.8 INCHES	(PSIA) (OBTAINED FROM DIFF. PRESSURE MEASUREMENTS.REF TO PT0)
PTBL2	BOUNDARY LAYER RAKE PRESSURE	1.6 INCHES	
PTBL3	BOUNDARY LAYER RAKE PRESSURE	2.4 INCHES	
PTBL4	BOUNDARY LAYER RAKE PRESSURE	3.2 INCHES	
PTBL5	BOUNDARY LAYER RAKE PRESSURE	4.0 INCHES	
		(1.65 IN. FROM FUSELAGE SKIN TO CENTER LINE OF SPLITTER DUCT LIP)	
DUCT DP	DIFFERENTIAL STATIC PRESSURE, RIGHT-LEFT SIDE OF DUCT (STA 266)		
ALPHA	ANGLE OF ATTACK (DEG)		
SIDESLIP	ANGLE OF SIDESLIP (DEG)		
PITCH	PITCH ATTITUDE (DEG)		
ROLL	ROLL ATTITUDE (DEG)		
V ACCEL	NORMAL ACCELERATION (G S)		
N1BLV	GROUND COOLING VALVE POSITION, N1 BLEED (OPEN/CLOSED)		
N2BLV	GROUND COOLING VALVE POSITION, N2 BLEED (OPEN/CLOSED)		
ICE DET	SPECIAL ICE DETECTOR SIGNAL LIGHT, COMP. INLET (ON/OFF)		
ENC TO IG	EXHAUST NOZZLE CONTROL TO IGNITER, PRESSURE (PSIA)		
IG SQ PR	IGNITER SQUIRTER PRESSURE (PSIA)		
IG TO EXH	IGNITER TO EXHAUST CONTROL, PRESSURE (PSIA)		
AB FU PR	AFTERBURNER FUEL PRESSURE (PSIA)		
AB AC PR	AFTERBURNER ACTUATOR PRESSURE (PSIA)		
AB MOTOR	AFTERBURNER MOTOR ACTUATOR (ON/OFF)		
RAIN CL	RAIN CLEARING VALVE POSITION (OPEN/CLOSED)		
ANTI ICE	ANTI-ICING VALVE POSITION (OPEN/CLOSED)		
IGNITION	IGNITION (ON/OFF)		

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TABLE IV-4

MANEUVER CODES F-102A COMPRESSOR STALL PROGRAM	
CODE	MANEUVER
	GENERAL
0/01	PRIOR TO BRAKE RELEASE
01	PRIOR TO BRAKE RELEASE
02	TAKE-OFF
03	CLIMB WITHOUT A/B
04	CLIMB WITH A/B
05	DESCENT WITHOUT A/B
06	DESCENT WITH A/B
07	CRUISE WITHOUT A/B
08	CRUISE WITH A/B
09	PRIOR TO STALL
10	AFTER STALL
11	PRIOR TO A/B BLOWOUT
13	ANTI ICING SELECTOR
15	ELECTRONIC COOLING JET PUMP SYSTEM
17	OPEN GROUND COOLING SYSTEM VALVE
18	RAIN CLEARING
19	CRUISE IN ICE CRYSTAL WEATHER
20	PRIOR TO CLOUD ENTRY
21	CLOUD ENTRY/TURBULENCE
31	SNAP-UP CLIMB
32	SNAP-UP CLIMB WITH A/B
33	SNAP-UP AND PUSHOVER
34	SNAP-UP AND PUSHOVER WITH A/B
36	SNAP-UP AND PULLUP WITH A/B
40	TANKER FLIGHT (SETTING UP)
41	TANKER FLIGHT (WATER ON)
	THROTTLE VARIATIONS
51	ADVANCE THROTTLE
52	ADVANCE THROTTLE WITH A/B
53	RETARD THROTTLE
56	MAX A/B TO MIN A/B
58	MIN A/B TO MAX A/B
59	MIL POWER TO A/B
60	A/B TO MIL POWER
	MISCELANEOUS MANEUVERS
61	SIDESLIP, MILITARY POWER
62	SIDESLIP
67	ADVANCE THROTTLE WITH SIDESLIP
69	RETARD THROTTLE WITH SIDESLIP
71	RUDDER MOVEMENT
75	CONSTANT ALTITUDE TURN
76	WIND-UP TURN
81	HIGH ANGLE OF ATTACK-LOW POWER
82	HIGH ANGLE OF ATTACK-HIGH POWER
83	HIGH ANGLE OF ATTACK-MILITARY POWER
85	ROLLER COASTER
89	ROLLS
99	

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TABLE IV-5

F-102A COMPRESSOR STALL DATA

A/C S/N 56-1151 FLIGHT NO. 58

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1   CAS KT    TAS KT    PR ALT FT     MACH       TA DEG C  PSO PSIA   PTO PSIA   RNI
2   PLA       OV BL     A/B NOZ       T FUEL C   RAT C     TT2 C      TT2 F      TT7 C
3    N1        N2       TT2/TSL       TT3-1 F    TT3-2 F   TT3-3 F    TT3-4 F     TT7 F
4   N1/TR2    N2/TR2    TH2 .5        TH2 .68    TH2 1.24  PT2        TT3/TR1    TT7/TR1
5    WF       N2/TR5     WF A/B       PT2/PSL     PS3      PT3        PS4.5      PT7
6   WFC.68    WFC.5     WFABC.68      PS4/PT2     PS4      PT3/PT2    PS4/PT7    PT7/PT2
7   WF/PS4    WF4/TR3   WF/PS4.5      WF/4.5TR3   PS4/PS3  PS4/PT3    PS3/PT2    PT7-ALT


      TIME OF DAY  12 HR  36 MIN  10.000 SEC        TEST MANEUVER   1
1     0.0       0.0         56.       0.000       24.4     14.665    14.665     0.901
2    53.1      CLSD                  46.8         25.5     22.0      71.7     629.6
3  5884.     9306.      1.024       352.9        363.4     387.7     380.7    1165.4
4  5813.     9194.      1.012         1.016        1.030   13.660    362.3    1137.4
5  8049.     7352.         0.         0.929       39.928   48.427   133.294    30.606
6  8518.     8555.         0.        10.370      141.662    3.545     4.628     2.240
7    56.82     55.89      60.38      59.39         3.547    2.925     2.923    15.940

      TIME OF DAY  13 HR   3 MIN  20.000 SEC        TFST MANEUVER  32
1   387.7     647.6      35345.       1.099      -44.5      3.401     7.249     0.492
2    52.6      CLSD                  32.8         12.7     10.3      50.6     646.5
3  5884.     9230.      0.983       322.5        330.3     338.1     347.8    1195.8
4  5931.     9305.      0.991         0.989        0.980    7.097    340.1    1215.3
5  4470.     7458.      11540.        0.482       20.858   24.401    63.169    16.478
6  9359.     9332.      24160.       11.002       78.088    3.438     4.738     2.321
7    57.25     57.88      70.77      71.55         3.743    3.200     2.938    13.076

      TIME OF DAY  13 HR   3 MIN  30.000 SEC        TEST MANEUVER  32
1   340.0     593.2      36732.       1.008      -45.3      3.182     6.085     0.431
2    52.5      CLSD                  32.3         -4.2      0.7      33.3     651.0
3  5926.     9236.      0.950       308.6        318.7     331.9     334.4    1203.8
4  6079.     9474.      0.974         0.966        0.938    5.957    340.2    1266.5
5  4026.     7517.      10113.        0.405       17.466   20.340    53.035    14.619
6 10281.    10187.      25823.       11.466       68.315    3.414     4.672     2.453
7    58.94     61.01      75.92      78.59         3.911    3.358     2.931    11.437

      TIMF OF DAY  13 HR   3 MIN  40.000 SEC        TEST MANEUVER  32
1   261.8     497.9      40209.       0.861      -52.9      2.692     4.368     0.334
2    52.6      CLSD                  32.3        -19.4    -20.4      -4.7     662.6
3  5950.     9161.      0.877       277.3        284.0     296.2     299.6    1224.7
4  6353.     9782.      0.936         0.914        0.849    4.172    329.8    1396.4
5  3287.     7623.       8601.        0.283       12.398   14.724    38.075    11.787
6 12661.    12366.      33123.       12.647       52.765    3.529     4.476     2.825
7    62.31     68.12      86.35      94.40         4.255    3.583   . 2.971     9.094

      TIMF OF DAY  13 HR   3 MIN  50.000 SEC        TEST MANEUVER  32
1   193.1     408.8      44447.       0.716      -58.6      2.196     3.091     0.256
2    52.7      CLSD                  33.2        -38.0    -36.7     -34.0     665.3
3  6032.     9086.      0.820       247.0        263.3     271.2     264.1    1229.5
4  6659.    10031.      0.905         0.874        0.782    2.946    318.6    1498.4
5  2643.     7706.       7082.        0.200        9.033    9.534    26.568     8.815
6 15083.    14556.      40408.       13.444       39.615    3.235     4.493     2.991
7    66.73     76.34      99.51     113.83         4.385    4.154     3.065     6.619
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TABLE IV-5 (CONT.)

F-102A COMPRESSOR STALL DATA
A/C S/N 56-1151 FLIGHT NO. 58

1	CAS KT	TAS KT	PR ALT FT	MACH	TA DEG C	PSO PSIA	PTO PSIA	RNI
2	PLA	OV BL	A/B NOZ	T FUEL C	RAT C	TT2 C	TT2 F	TT7 C
3	N1	N2	TT2/TSL	TT3-1 F	TT3-2 F	TT3-3 F	TT3-4 F	TT7 F
4	N1/TR2	N2/TR2	TH2 .5	TH2 .68	TH2 1.24	PT2	TT3/TR1	TT7/TR1
5	WF	N2/TR5	WF A/B	PT2/PSL	PS3	PT3	PS4.5	PT7
6	WFC.68	WFC.5	WFABC.68	PS4/PT2	PS4	PT3/PT2	PS4/PT7	PT7/PT2
7	WF/PS4	WF4/TR3	WF/PS4.5	WF/4.5TR3	PS4/PS3	PS4/PT3	PS3/PT2	PT7-ALT

TIME OF DAY 13 HR 3 MIN 56.800 SEC TEST MANEUVER 9

1	158.2	347.4	45733.	0.613	-62.0	2.064	2.662	0.232
2	52.6	CLSD		33.9	-44.5	-46.2	-51.2	663.5
3	6044.	9038.	0.787	234.9	250.1	262.3	256.4	1226.3
4	6810.	10184.	0.887	0.850	0.743	2.537	318.6	1557.2
5	2359.	7721.	6863.	0.172	7.616	7.603	20.814	7.870
6	16073.	15396.	46765.	13.426	34.067	2.996	4.328	3.101
7	69.24	81.45	113.33	133.32	4.472	4.480	3.001	5.805

TIME OF DAY 13 HR 9 MIN 25.000 SEC TEST MANEUVER 7

1	252.5	534.0	47226.	0.960	-69.5	1.921	3.476	0.286
2	48.6	CLSD		31.9	-31.1	-32.2	-25.9	626.1
3	5794.	8870.	0.836	237.7	250.1	258.3	256.4	1159.0
4	6336.	9700.	0.914	0.885	0.801	3.375	299.7	1386.1
5	2617.	7579.	0.	0.229	10.804	11.828	29.329	8.993
6	12868.	12460.	0.	12.506	42.216	3.503	4.694	2.664
7	61.99	70.01	89.23	100.77	3.907	3.569	3.200	7.071

TIME OF DAY 13 HR 9 MIN 29.800 SEC TEST MANEUVER 85

1	238.9	500.3	45966.	0.893	-66.6	2.041	3.428	0.282
2	48.6	CLSD		32.4	-32.2	-33.7	-28.8	623.4
3	5794.	8834.	0.830	231.2	249.5	257.8	258.3	1154.2
4	6357.	9693.	0.911	0.881	0.794	3.295	300.0	1389.5
5	2527.	7556.	0.	0.224	9.653	11.466	28.409	8.756
6	12790.	12370.	0.	12.338	40.655	3.479	4.642	2.657
7	62.17	70.53	88.98	100.94	4.211	3.545	2.929	6.714

TIME OF DAY 13 HR 9 MIN 34.000 SEC TEST MANEUVER 85

1	231.0	486.0	46108.	0.870	-67.6	2.027	3.322	0.279
2	48.1	CLSD		33.9	-35.2	-36.7	-34.0	621.6
3	5794.	8834.	0.820	229.8	239.3	255.3	250.6	1151.0
4	6396.	9752.	0.905	0.874	0.782	3.209	297.1	1402.7
5	2491.	7585.	0.	0.218	9.476	10.379	26.568	8.815
6	13050.	12593.	0.	12.236	39.268	3.234	4.454	2.747
7	63.43	72.56	93.76	107.26	4.143	3.783	2.952	6.787

TIME OF DAY 13 HR 9 MIN 49.600 SEC TEST MANEUVER 85

1	228.7	487.3	46536.	0.870	-66.5	1.986	3.254	0.270
2	48.3	CLSD		37.0	-32.8	-35.3	-31.6	615.4
3	5770.	8834.	0.825	228.9	242.2	249.8	251.6	1139.8
4	6352.	9725.	0.908	0.877	0.787	3.137	294.6	1381.3
5	2494.	7589.	0.	0.213	9.564	10.983	27.948	8.756
6	13317.	12864.	0.	12.849	40.309	3.501	4.603	2.791
7	61.88	70.52	89.25	101.71	4.214	3.670	3.049	6.770

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TABLE IV-5 (CONT.)

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                    F-102A COMPRESSOR STALL DATA
          A/C S/N 56-1151            FLIGHT NO.  58

 1  CAS KT    TAS KT    PR ALT FT    MACH        TA DEG C  PSO PSIA  PTO PSIA  RNI
 2  PLA       OV BL     A/B NOZ      T FUEL C    RAT C     TT2 C     TT2 F     TT7 C
 3   N1        N2       TT2/TSL      TT3-1 F     TT3-2 F   TT3-3 F   TT3-4 F    TT7 F
 4  N1/TR2    N2/TR2    TH2 .5       TH2 .68     TH2 1.24  PT2       TT3/TR1   TT7/TR1
 5   WF       N2/TR5     WF A/B      PT2/PSL      PS3      PT3       PS4.5     PT7
 6  WFC.68    WFC.5     WFABC.68     PS4/PT2      PS4      PT3/PT2   PS4/PT7   PT7/PT2
 7  WF/PS4    WF4/TR3   WF/PS4.5     WF/4.5TR3   PS4/PS3   PS4/PT3   PS3/PT2   PT7-ALT


      TIME OF DAY  13 HR   9 MIN  52.000 SEC       TEST MANEUVER  85
 1   224.5     484.2      46960.       0.863      -66.1      1.946      3.166     0.264
 2    48.6     CLSD                    38.1       -34.1     -35.3      -31.6     616.3
 3   5770.     8834.      0.825       224.8       233.1     249.3      247.8    1141.4
 4   6352.     9725.      0.908        0.877       0.787     3.065     289.3    1383.2
 5   2481.     7613.         0.        0.208       9.653    10.621     27.258     8.697
 6 13558.     13097.         0.       12.866      39.442     3.464      4.534     2.837
 7   62.92     71.70      91.04      103.75        4.085     3.713      3.149     6.751

      TIME OF DAY  13 HR  10 MIN   9.000 SEC       TEST MANEUVER  85
 1   243.1     499.7      45257.       0.893      -67.3      2.112      3.550     0.296
 2    48.3     CLSD                    39.3       -32.0     -34.5      -30.2     604.7
 3   5747.     8833.      0.827       227.1       241.0     247.5      251.6    1120.5
 4   6316.     9707.      0.909        0.879       0.791     3.445     292.1    1353.5
 5   2640.     7595.         0.        0.234      10.804    12.069     30.020     9.288
 6 12803.     12375.         0.       12.102      41.696     3.503      4.488     2.695
 7   63.31     71.99      87.94       99.99        3.859     3.454      3.136     7.176

      TIME OF DAY  13 HR  12 MIN  27.000 SEC       TEST MANEUVER  75
 1   193.1     433.3      48422.       0.779      -69.5      1.814      2.710     0.236
 2    52.5     CLSD                    32.1       -44.3     -44.9      -48.8     644.8
 3   5926.     8907.      0.792       221.0       239.3     251.3      248.3    1192.6
 4   6659.    10008.      0.890        0.853       0.749     2.607     303.0    1505.6
 5   2456.     7669.      6253.        0.177       8.502     9.655     24.036     8.165
 6 16225.     15558.      41302.      14.065      36.667     3.703      4.490     3.132
 7   66.99     78.49     102.19      119.74        4.312     3.797      3.261     6.351

      TIME OF DAY  13 HR  12 MIN  57.000 SEC       TEST MANEUVER  75
 1   172.3     362.5      44354.       0.644      -64.5      2.206      2.916     0.257
 2    52.6     CLSD                    31.6       -45.6     -47.2      -53.1     642.1
 3   5902.     8907.      0.783       216.9       235.4     243.0      243.6    1187.8
 4   6667.    10060.      0.885        0.847       0.739     2.802     299.4    1515.3
 5   2500.     7698.         0.        0.190       9.033    10.500     24.266     8.402
 6 15473.     14809.         0.       13.640      38.228     3.746      4.549     2.998
 7   65.41     77.19     103.04      121.60        4.231     3.640      3.223     6.195

      TIME OF DAY  13 HR  25 MIN   0.000 SEC       TEST MANEUVER  81
 1   212.9     400.2      39068.       0.698      -56.6      2.844      3.938     0.334
 2    31.4     CLSD                    38.5       -33.3     -35.6      -32.1     473.1
 3   4938.     8110.      0.824       178.7       181.1     208.8      204.1     883.7
 4   5439.     8933.      0.907        0.876       0.786     3.863     234.4    1072.1
 5   1543.     7229.         0.        0.262       9.919    11.586     19.893     7.338
 6  6694.      6465.         0.        8.144      31.466     2.999      4.287     1.899
 7   49.03     55.92      77.56       88.45        3.172     2.715      2.567     4.493
```

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TABLE IV-5 (CONT.)

```
                     F-102A COMPRESSOR STALL DATA
               A/C S/N 56-1151            FLIGHT NO.  58

 1  CAS KT   TAS KT   PR ALT FT    MACH     TA DEG C PSO PSIA  PTO PSIA  RNI
 2  PLA      OV BL     A/B NOZ   T FUEL C  RAT C    TT2 C     TT2 F     TT7 C
 3   N1       N2      TT2/TSL    TT3-1 F   TT3-2 F  TT3-3 F   TT3-4 F    TT7 F
 4  N1/TR2   N2/TR2   TH2 .5     TH2 .68   TH2 1.24 PT2       TT3/TR1   TT7/TR1
 5   WF      N2/TR5    WF A/B    PT2/PSL    PS3     PT3       PS4.5     PT7
 6  WFC.68   WFC.5    WFABC.68   PS4/PT2    PS4     PT3/PT2   PS4/PT7   PT7/PT2
 7  WF/PS4   WF4/TR3  WF/PS4.5   WF/4.5TR3  PS4/PS3 PS4/PT3   PS3/PT2   PT7-ALT


      TIME OF DAY  13 HR  26 MIN   0.000 SEC        TEST MANEUVER  81
 1   130.3    246.8    37453.      0.423     -49.4     3.074     3.477     0.301
 2    46.2    CLSD                 40.5      -41.3     -41.4     -42.6     545.4
 3   5567.     8615.    0.804     196.7      210.0     228.4     228.3    1013.8
 4   6208.     9607.    0.896       0.862      0.763     3.380   268.4    1260.9
 5   2403.     7548.        0.      0.230      9.919    12.069    28.869     8.934
 6 12120.    11654.         0.     11.001     37.188     3.570     4.162     2.643
 7   64.63     74.96     83.25     96.56       3.749     3.081     2.934     5.859

      TIME OF DAY  13 HR  26 MIN  40.000 SEC        TEST MANEUVER  81
 1   106.8    197.7    35930.      0.336     -46.3     3.307     3.577     0.304
 2    49.6    CLSD                 44.7      -39.9     -41.1     -42.1     604.7
 3   5789.     8833.    0.804     222.4      239.9     256.3     260.2    1120.5
 4   6453.     9845.    0.897       0.862      0.764     3.424   304.0    1392.0
 5   2706.     7579.        0.      0.232     11.513    13.397    33.472     9.466
 6 13462.    12946.         0.     12.379     42.389     3.912     4.478     2.764
 7   63.84     73.99     80.85     93.70       3.681     3.164     3.362     6.158

      TIME OF DAY  13 HR  27 MIN   0.000 SEC        TEST MANEUVER  81
 1    87.0    161.9    35801.      0.274     -43.8     3.328     3.507     0.296
 2    49.9    CLSD                 44.9      -38.2     -40.4     -40.7     614.5
 3   5817.     8870.    0.807     224.8      247.3     267.7     261.7    1138.2
 4   6473.     9869.    0.898       0.864      0.767     3.338   310.0    1409.1
 5   2773.     7581.        0.      0.227     11.159    13.759    34.393     9.761
 6 14116.    13584.         0.     12.696     42.389     4.121     4.342     2.923
 7   65.43     75.65     80.64     93.25       3.798     3.080     3.342     6.432
```

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TABLE IV-5 (CONT.)

```
              F-102A COMPRESSOR STALL DATA  (PART II)
        A/C S/N 56-1151          FLIGHT NO.  58


 8  CAS KT   TAS KT   PR ALT FT    MACH    TA DEG F   PSO PSIA   PTO PSIA    TT2 K
 9  PS2M      PS2 1     PS2 2      PS2 3     PS2 4     PS2 ESF    PS2 ESA    N1 BLV
10  ALPHA    SIDESLIP   PITCH      ROLL     V ACCEL   ICE DET   ENC TO IG    N2 BLV
11  PSSW1     PSSW2     PSSW3      PSSW4     PSSW5      PSSW6    IG SQ PR   RAIN CL
12  PTBL1     PTBL2     PTBL3      PTBL4     PTBL5    AB FU PR  IG TO EX   ANTI ICE
13  PSDF1     PSDF2     PSDF3      PSDF4    DUCT DP   AB MOTOR  AB AC PR   IGNITION



     TIME OF DAY  12 HR  36 MIN  10.000 SEC       TEST MANEUVER   1
 8   0.0        0.0         56.     0.000      76.0      14.665     14.665      295.2
 9 14.343    14.355     14.315   14.340     14.285      14.331     14.016
10 -3.3     -16.7       -1.1      2.8        1.03        OFF*********
11***************     14.856   14.984     14.840     15.237********
12***************************************************************
13**********************************        -0.070          *********        OFF


     TIME OF DAY  13 HR   3 MIN  20.000 SEC       TEST MANEUVER  32
 8 387.7      647.6      35345.    1.099      -48.2       3.401      7.249      283.5
 9   6.124     6.126      6.097    6.161       6.129      6.253      6.066
10   4.6      -0.6        8.6      1.2         1.97        OFF*********
11***************      6.058    6.159       6.205      6.464********
12***************************************************************
13**********************************         0.009          *********        OFF


     TIME OF DAY  13 HR   3 MIN  30.000 SEC       TEST MANEUVER  32
 8 340.0      593.2      36732.    1.008      -49.6       3.182      6.085      273.8
 9   4.855     4.857      4.795    4.859       4.843      4.972      4.785
10   6.3      -0.4       31.8      3.2         2.17        OFF*********
11***************      4.807    4.885       4.929      5.167********
12***************************************************************
13**********************************         0.004          *********        OFF


     TIME OF DAY  13 HR   3 MIN  40.000 SEC       TEST MANEUVER  32
 8 261.8      497.9      40209.    0.861      -63.2       2.692      4.368      252.7
 9   3.490     3.486      3.452    3.488       3.486      3.596      3.409
10   4.9      -0.6       44.0      3.2         1.16        OFF*********
11***************      3.449    3.507       3.567      3.746********
12***************************************************************
13**********************************         0.002          *********        OFF


     TIME OF DAY  13 HR   3 MIN  50.000 SEC       TEST MANEUVER  32
 8 193.1      408.8      44447.    0.716      -73.5       2.196      3.091      236.4
 9   2.437     2.432      2.395    2.458       2.441      2.554      2.413
10   5.2      -0.7       40.5      0.8         0.67        OFF*********
11***************      2.437    2.453       2.520      2.667********
12***************************************************************
13**********************************         0.016          *********        OFF
```

TABLE IV-5 (CONT.)

```
                    F-102A COMPRESSOR STALL DATA     (PART II)
               A/C S/N 56-1151          FLIGHT NO.  58

 8  CAS KT  TAS KT  PR ALT FT   MACH   TA DEG F  PSO PSIA  PTO PSIA   TT2 K
 9  PS2M     PS2 1     PS2 2    PS2 3    PS2 4    PS2 ESF   PS2 ESA  N1 BLV
10  ALPHA  SIDFSLIP    PITCH    ROLL    V ACCEL  ICE DET  ENC TO IG  N2 BLV
11  PSSW1    PSSW2     PSSW3    PSSW4    PSSW5     PSSW6   IG SQ PR  RAIN CL
12  PTBL1    PTBL2     PTBL3    PTBL4    PTBL5   AB FU PR  IG TO EX ANTI ICE
13  PSDF1    PSDF2     PSDF3    PSDF4   DUCT DP  AB MOTOR  AB AC PR IGNITION


     TIMF OF DAY  13 HR   3 MIN  56.800 SEC       TEST MANEUVER   9
 8 158.2    347.4    45733.    0.613   -79.6       2.064     2.662    226.9
 9  2.129    2.131     2.090    2.152     2.148     2.258     2.141
10  7.6     -0.5     34.1      1.2       0.63      OFF*********
11***************      2.131    2.129     2.215     2.333********
12***************************************************************
13***********************************    -0.002          *********       OFF

     TIME OF DAY  13 HR   9 MIN  25.000 SEC       TEST MANEUVER   7
 8 252.5    534.0    47226.    0.960   -93.2       1.921     3.476    240.9
 9  2.796    2.791     2.768    2.810     2.807     2.878     2.807
10  4.7     -0.6     3.9       2.8       0.94      OFF*********
11***************      2.762    2.803     2.875     3.035********
12***************************************************************
13***********************************     0.004          *********       OFF

     TIMF OF DAY  13 HR   9 MIN  29.800 SEC       TEST MANEUVER  85
 8 238.9    500.3    45966.    0.893   -87.8       2.041     3.428    239.3
 9  2.740    2.728     2.687    2.740     2.751     2.798     2.740
10 23.0     -0.9     21.6      5.6       3.17      OFF*********
11***************      2.712    2.749     2.798     2.967********
12***************************************************************
13***********************************     0.004          *********       OFF

     TIMF OF DAY  13 HR   9 MIN  34.000 SEC       TEST MANEUVER  85
 8 231.0    486.0    46108.    0.870   -89.8       2.027     3.322    236.4
 9  2.650    2.643     2.629    2.666     2.638     2.931     2.638
10 -1.7     -0.3      5.9      1.2       0.03      OFF*********
11***************      2.629    2.659     2.731     2.880********
12***************************************************************
13***********************************    -0.002          *********       OFF

     TIME OF DAY  13 HR   9 MIN  49.600 SEC       TEST MANEUVER  85
 8 228.7    487.3    46536.    0.870   -87.7       1.986     3.254    237.7
 9  2.608    2.595     2.553    2.611     2.618     2.667     2.562
10 19.5     -1.2     18.5      6.8       2.57      OFF*********
11***************      2.574    2.613     2.675     2.841********
12***************************************************************
13***********************************     0.007          *********       OFF
```

TABLE IV-5 (CONT.)

```
                    F-102A COMPRESSOR STALL DATA  (PART II)
          A/C S/N 56-1151            FLIGHT NO.  58

 8  CAS KT   TAS KT   PR ALT FT   MACH   TA DEG F   PSO PSIA   PTO PSIA    TT2 K
 9  PS2M     PS2 1     PS2 2      PS2 3    PS2 4    PS2 ESF    PS2 ESA   N1 BLV
10  ALPHA   SIDESLIP   PITCH      ROLL    V ACCEL   ICE DET   ENC TO IG   N2 BLV
11  PSSW1    PSSW2     PSSW3      PSSW4    PSSW5     PSSW6    IG SQ PR   RAIN CL
12  PTBL1    PTBL2     PTBL3      PTBL4    PTBL5    AB FU PR  IG TO EX  ANTI ICE
13  PSDF1    PSDF2     PSDF3      PSDF4   DUCT DP   AB MOTOR  AB AC PR  IGNITION


     TIME OF DAY  13 HR   9 MIN  52.000 SEC        TEST MANEUVER  85
 8 224.5     484.2     46960.     0.863   -87.0       1.946      3.166      237.7
 9  2.527     2.509     2.495     2.550     2.527     2.551      2.504
10  1.2      -0.3       7.6       3.6       0.39       OFF*********
11***************       2.513     2.543     2.617     2.769********
12****************************************************************
13**********************************        0.011           *********        OFF


     TIME OF DAY  13 HR  10 MIN   9.000 SEC        TEST MANEUVER  85
 8 243.1     499.7     45257.     0.893   -89.1       2.112      3.550      238.5
 9  2.881     2.863     2.826     2.895     2.888     2.916      2.834
10 11.2      -1.3       4.3       0.0       1.66       OFF*********
11***************       2.863     2.893     2.969     3.132********
12****************************************************************
13**********************************        0.009           *********        OFF


     TIME OF DAY  13 HR  12 MIN  27.000 SEC        TEST MANEUVER  75
 8 193.1     433.3     48422.     0.779   -93.1       1.814      2.710      228.2
 9  2.143     2.122     2.076     2.143     2.154     2.190      2.084
10 25.1      -2.5       0.8     -55.2       2.30       OFF*********
11***************       2.129     2.166     2.219     2.377********
12****************************************************************
13**********************************        0.016           *********        OFF


     TIME OF DAY  13 HR  12 MIN  57.000 SEC        TEST MANEUVER  75
 8 172.3     362.5     44354.     0.644   -84.1       2.206      2.916      225.8
 9  2.218     2.199     2.151     2.218     2.229     2.288      2.183
10 25.3      -2.7      -2.3     -51.8       1.90       OFF*********
11***************       2.215     2.245     2.303     2.462********
12****************************************************************
13**********************************        0.016           *********        OFF


     TIME OF DAY  13 HR  25 MIN   0.000 SEC        TEST MANEUVER  81
 8 212.9     400.2     39068.     0.698   -69.9       2.844      3.938      237.5
 9  3.501     3.490     3.446     3.518     3.513     3.501      3.501
10  8.2      -0.7       5.9       2.0       1.21       OFF*********
11***************       3.458     3.469     3.545     3.677********
12****************************************************************
13**********************************        0.009           *********        OFF
```

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TABLE IV-5 (CONC'D)

```
              F-102A COMPRESSOR STALL DATA  (PART II)
      A/C S/N 56-1151            FLIGHT NO.  58

 8  CAS KT   TAS KT   PR ALT FT   MACH    TA DEG F  PSO PSIA  PTO PSIA   TT2 K
 9  PS2M      PS2 1     PS2 2     PS2 3    PS2 4    PS2 ESF   PS2 ESA   N1 BLV
10  ALPHA   SIDESLIP    PITCH     ROLL    V ACCEL   ICE DET  ENC TO IG  N2 BLV
11  PSSW1     PSSW2     PSSW3     PSSW4    PSSW5     PSSW6   IG SQ PR  RAIN CL
12  PTBL1     PTBL2     PTBL3     PTBL4    PTBL5    AB FU PR  IG TO EX ANTI ICE
13  PSDF1     PSDF2     PSDF3     PSDF4   DUCT DP   AB MOTOR  AB AC PR IGNITION


      TIME OF DAY  13 HR  26 MIN   0.000 SEC       TEST MANEUVER  81
 8 130.3     246.8     37453.     0.423   -57.0      3.074     3.477     231.6
 9  2.950     2.930     2.877     2.953     2.948     2.892     2.974
10 20.2     -1.0       4.1       2.0       1.16      OFF*********
11****************      2.962     2.992     3.073     3.237********
12*************************************************************
13*********************************        0.021           *********        OFF


      TIME OF DAY  13 HR  26 MIN  40.000 SEC       TEST MANEUVER  81
 8 106.8     197.7     35930.     0.336   -51.3      3.307     3.577     231.9
 9  3.020     3.022     2.939     3.017     3.043     3.031     3.054
10 25.3     -1.4      11.7       2.0       1.28      OFF*********
11****************      3.066     3.108     3.152     3.334********
12*************************************************************
13*********************************        0.007           *********        OFF


      TIME OF DAY  13 HR  27 MIN   0.000 SEC       TEST MANEUVER  81
 8  87.0     161.9     35801.     0.274   -46.9      3.328     3.507     232.7
 9  3.023     3.013     2.945     3.016     3.032     3.034     2.906
10 25.3     -1.4       8.0       2.0       1.12      OFF*********
11****************      3.082     3.122     3.161     3.349********
12*************************************************************
13*********************************        0.016           *********         ON
```

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AFFTC Technical Note

1119

F-102A ROLLING CHARACTERISTICS EVALUATION

AFFTC-TN-55-24

NOVEMBER 1955

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Reclassified by: Dept.
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#89

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AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE

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AFFTC-TN-55-24

F-102A ROLLING CHARACTERISTICS EVALUATION

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AFFTC-TN-55-24

PUBLICATION REVIEW

This Technical Note has been reviewed and is approved.

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[redacted]

ABSTRACT

Evaluation of the rolling characteristics of the F-102A have been made to verify the necessity for roll restrictions and to determine the effect of these restrictions on the capability of the weapon to accomplish the basic Air Defense Command intercept problem. With yaw damper and turn coordinator operating, the roll rate is unrestricted for changes in bank angle up to 180 degrees. Full 360 rolls may be made safely if the roll rate is limited to 80 degrees per second. These restrictions do not interfere with normal operational use of the airplane. The limits may be easily exceeded during non-operational type maneuvers so that damage to the airplane is possible. Tests were also made with the artificial stability devices inoperative. Rolling limitations were not changed appreciably. Directional oscillations will seriously impair the kill probability when firing rockets without the yaw damper or turn coordinator operating.

Improved rolling characteristics were obtained from the increased directional stability provided by a sideslip feed-back "black box" which was installed on the test airplane. Limit roll rates of at least 120 degrees are possible with this device and further improvements may be realized. Improved directional stability should be provided as soon as possible on all production aircraft. Increased inherent directional stability is desirable for it will be 100% reliable and should also provide improved tracking capability if the yaw damper or turn coordinator fail to function properly. Tests of the F-102A with an enlarged vertical tail should be expedited to determine if the required improvements can be obtained in this way without too great a penalty in weight and drag.

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INTRODUCTION

This Technical Note presents the results of an evaluation of the rolling characteristics of the F-102A airplane in the region free of inertial coupling effects. These flight tests were conducted under the authority of the Commander, Air Force Flight Test Center, at the request of the Air Research and Development Command. The tests were accomplished at Edwards Air Force Base from 4 October to 11 October 1955. A total of thirteen flights in 12:25 hours were required in the YF-102A airplane USAF No. 53-1787 to complete the evaluation. Evaluation Pilots were: J. S. Holtoner, Brig. General, USAF; H. A. Hanes, Colonel, USAF; R. M. Baker, Colonel, USAF; F. K. Everest, Lt. Col., USAF; R. L. Stephens, Major, USAF; S. R. Childs, Major, USAF.

The YF-102A airplane USAF No. 53-1787 is a prototype of the F-102A airplane with the same aerodynamic contours and approximately the same moment of inertia. The lateral and directional control systems are the most fully developed of any in the present F-102A airplanes and have a yaw damper, a pitch damper, and a turn coordinator. A sideslip feed-back is also incorporated to augment the directional stability for test purposes. Telemetering facilities are provided for ground monitoring during the flight to assist in exploratory tests.

The test airplane does not have a production "yoke" stick with the result that twenty degrees of total aileron are available instead of fourteen degrees of the production airplane. The "glove fit" nose of the prototype airplane restricts the test airplane to a 3.5 g limit load factor.

Limitations in roll were described by the contractor prior to the evaluation. These limitations are as follows:

1. Unrestricted in the top two quadrants (no inverted flight attitudes).

2. A limit rate of roll of 80 degrees per second with either the basic airplane or with the yaw damper and turn coordinator operating for rolls through 360 degrees of bank angle, with up to one degree of critical elevator input. (With fixed elevator this roll limitation is 150 degrees per second).

3. A limit roll rate of 120 degrees per second for roll up to 360 degrees of bank angle with the sideslip feed-back operating and one degree of critical elevator input (150 degrees per second with fixed elevator).

TEST RESULTS

INVESTIGATION OF LIMITS

1. Rolls Restricted to the Top Two Quadrants:

The airplane is unrestricted in rolling in the top two quadrants, i.e., when a bank angle of 90 degrees is not exceeded. The short duration of a roll within this limiting bank angle precludes the build-up of conditions characteristic of the inertia coupling effects. It may be possible, however, to encounter the violent lateral and longitudinal accelerations resulting from inertial coupling in roll by a series of rapid reversals.

In the evaluation, the aircraft was rolled as rapidly as possible from approximately a 70 degree bank to a 70 degree bank in the opposite direction followed by a reversal to the original bank angle. These rolls were accomplished at various speeds and altitudes, with and without the turn coordinator and yaw damper operating. Examples of these rolls are presented as time histories in Figures 1 through 5 of Appendix I, and a summary of the data is presented in the following table:

ROLLS FROM 70° BANK TO 70° BANK AND REVERSE

INHERENT AIRCRAFT

(No Stability Augmentation)

Altitude (feet)	Indicated Speed (knots)	Ind. Mach No.	Total Aileron Defl. (degrees)	Rate of Roll (deg/sec)	Max.Angle of Yaw (degrees)	Normal Accel. (g)
35,000	395	1.1	10L - 12R	135L - 85R	6L	3.5 to .2
34,000	334	.94	12L -12.6R	135L - 75R	3L	3.6 to 0.6
35,500	283	.82	13L - 13R	125L - 70R	5R	
34,500	254	.75	14L - 12R	125L - 70R	8R	2.9
34,500	232	.68	12L - 11R	115L - 65R	6R	1.7
16,000	475	.95	8.6L-7.0R	120L - 85R	6L	4.2 to 0
16,000	305	.61	10L-10.8R	120L -120R	3R	2.4
15,000	225	.45	12L-11.0R	80L - 90R	7R	2.5 to .8

ROLLS FROM 70° BANK TO 70° BANK AND REVERSE

(Yaw Damper and Turn Coordinator Operating)

Altitude (feet)	Ind. Speed (knots)	Ind. Mach. No.	Total Aileron Defl. (degrees)	Rate of Roll (deg/sec)	Max. Angle of Yaw (degrees)	Normal Accel. (g)
34,000	380	1.05	10.3L	150L	3.5L	3.9
33,000	345	.95	13L - 12.5R	145L - 80R	1°L-1°R	2.6 to 0.7
16,000	475	.93	8°L - 10°R	125L -115R	1°L-2°R	1 to 3.2

Ailerons were unrestricted in these roll maneuvers; the pilot used as much aileron as possible in the roll without exceeding a 70 degree bank angle. At 35,000 feet altitude, 10 to 14 degrees of total aileron were used with resultant rates of roll in the region of 135 degrees per second; and at 16,000 feet altitude, eight to twelve degrees of total aileron were used with rates of roll in the region of 120 degrees per second. Angles of yaw resulting from these maneuvers were minimized with the turn coordinators operating. This can be seen in the rolls presented in Figures 4 and 5. The rates of roll with the turn coordinator functioning were approximately 15 degrees per second higher than those roll rates without the coordinator.

2. Rolls Restricted to 180 Degrees Change of Bank Angle:

The airplane should be unrestricted in rolling to an inverted flight position provided that the change in bank angle is no greater than 180 degrees. The duration of a roll with this limiting condition precludes the build-up of conditions characteristic of inertial coupling. It may be possible, however, to aggravate the yawing tendencies in these rolls by setting up a yawing condition prior to entering the roll.

Rolls were accomplished from a wings level position to an inverted flight condition, stopped at 180 degrees bank change, and rolled back to a wings level position. Various elevator deflections were made during the rolls to determine if the yawing and pitching tendencies could be aggravated sufficiently to warrent a restriction. These rolls were accomplished at 16,000 feet, 35,000 feet, and 46,000 feet altitude through a band of speeds with the inherent aircraft (no stability augmenters), with the yaw damper and turn coordinator operating and with the sideslip feed-back in addition to the yaw damper and turn coordinator. Typical examples of these rolls are presented as time histories in Figures 7 through 13 of Appendix I. A summary of the rolls from wings level to the 180 degree position is presented in the following table:

- ROLLS FROM WINGS LEVEL TO 180 DEGREES AND REVERSE

INHERENT AIRPLANE

(No Dampers or Turn Coordinators)

Altitude (feet)	IAS (knots)	Ind. Mach No.	Total Aileron (degrees)	Peak Roll Rate (deg/sec)	Max. Angle of Yaw (degrees)	Normal Acceleration (g)	Elevator Deflection (degrees)
37,000	360	1.1	11.5R	180R - 100L	4R - 6L	0.3	0
			8.0R	150R - 110L	3R - 4L	3.1 to 0.5	1.9 up
			7.4R	120R - 70L	4R - 4L	1.0	2.3 down
			7.0L	120L - 60R	7L - 2R	1.5	1.5 down
36,000	305	0.9	8.5R	140R - 120L	1R - 3L	2	0
			13.5R	190R	1R	3.25 to 0.5	0.5 up
			8.5R	140R	3L	2.9	0.5 up
			8.0R	130R	0	3.8	0.6 up
			7.5L	140L	9L	2.3	1.65 down
			7.0L	110L	13L	1.5 to -1.2	2.4 down
			7.0R	130R	6R	1.8 to -1.2	1.2 down
35,000	250	0.74	12.5R	140R	5L	1.8	0
			11.5R	120R	6L	2.0 to -1.7	0
			13.0R	170R	6L	1.8	0.5 up
			16.5R	120R	14L	3.8	2.75 up
			14.4R	120R	10.5L	2.3	2.5 up
			14.0R	110R	11L	2.0	2.0 up
			12.0R	160R	5R	2.3 to 0	1.25 down
			9.0L	130L	2L	1.7 to -.5	0.5 down

(Yaw Damper and Turn Coordinator Operating)

Altitude (feet)	IAS (knots)	Ind. Mach No.	Total Aileron (degrees)	Peak Roll Rate (deg/sec)	Max. Angle of Yaw (degrees)	Normal Acceleration (g)	Elevator Deflection (degrees)
46,000	240	.9	12.6L	180L	2.5R	1.0	0
			14.5R	130R	9L - 4R	1.2	2.75 up
			15.5R	120R	10L - 5.5R	1.6	3.25 up
			13.0R	180R	2R	1.1	0.5 down
36,000	360	1.05	13.0R	170R	1L - 3R	4.0	4.0 up
			11.5L	180L	5L	2.7	1.0 down
36,000	310	.91	10.0R	180R	1R	2.0	0
			9.0L	170L	4.2L	1.6	1.5 up
			8.7L	120R	2.0R	1.5	.0
			14.0R	190R	5.5R	2.4	0.2 up
			14.5R	170R	6.0L	2.3	2.65 up
			12.6L	120L	6.0R	3.5	3.0 up
			8.0R	170R	4.0R	1.0	0.5 down

ROLLS FROM WINGS LEVEL TO 180 DEGREES AND REVERSE (Continued)

(Yaw Damper and Turn Coordinator Operating)

Altitude (feet)	IAS (knots)	Ind. Mach No.	Total Aileron (deg)	Peak Roll Rate (deg/sec)	Max. Angle of Yaw (degrees)	Normal Acceleration (g)	Elevator Deflection (degrees)
35,000	250	.74	15.0R	120R	12L	1.2	3.0 up
			14.7R	120R	16L	3.2 to 0	3.35 up
			14.0R	120R	13L - 9R	4.1 to .8	7.0 up
			13.7R	120R	12.5L	2.6 to .4	2.45 up
			13.8R	120R	14.0L	3.1 to .2	1.8 up
			10.5R	120R	9.0L	2.6	0.85 up
			11.2L	110L	2.0R	2.4 to .6	0.35 down
			11.0L	100L	10.0R	3.1 to .3	2.45 up

These rolls indicate that negligible yaw is encountered with fixed elevator with speeds in the region of 300 knots IAS. Adverse yaw is encountered at lower speeds and favorable yaw (yawing in the direction of the roll) at higher speeds. Longitudinal pull forces during the roll tend to induce adverse yaw while the converse is true of push forces so that a pull force tends to aggravate rolls with speeds in the region of 250 knots while a push force tends to offset the adverse yaw. In the same respect, at 360 knots IAS, a pull force offsets the yaw into the roll while a push on the longitudinal control tends to aggravate the yawing condition. This is illustrated graphically in Figure 6 of Appendix I. It may also be noted in Figure 6 that the turn coordinator makes no significant difference in the yaw angles attained in rolls through 180 degrees. The yawing condition with fixed elevator appears to be dependent more upon IAS than upon Mach number. This is demonstrated in Figure 6 in that the angle of yaw characteristics at 240 knots at 46,000 feet altitude is more similar to that at 250 knots and 35,000 feet than at a similar Mach number (310 knots) at 35,000 feet. During these tests, greater pull and push forces were employed during the roll than would be used at other than test conditions without encountering the violent conditions of inertia coupling. A more complete picture of the speed and altitude envelope is desirable by obtaining similar rolls at higher speeds and lower altitudes.

3. Rolls Beyond 180° Change in Bank Angle Through 360°:

A limit rate of roll of 80 degrees per second is recommended for the aircraft with or without the yaw damper and turn coordinator operating when aileron rolls in excess of 180 degrees of bank angle are made. Higher roll rates are possible without adverse effect, however an inadvertent elevator deflection at these higher roll rates could easily cause a divergence in pitch and yaw resulting in excess load factors with possible damage to the aircraft. With increased directional stability (as obtained with sideslip feed-back operating) the yawing tendencies in rolls are minimized. The limit roll rate is 120 degrees per second, although further exploratory tests are required with this improvement to determine if 120 degrees per second is a realistic limit, or if the limit could be raised.

Rolls were accomplished from a wings level position through a 360 degree change of bank angle with various elevator positions. These tests were conducted at 16,000 feet and 35,000 feet through a range of speeds. Aileron deflection was restricted in these tests so that the previously established roll limits were not exceeded. The limits with the inherent airplane were demonstrated to be realistic during the Phase II tests when a 25 degree angle of yaw was encountered during a roll. A summary of the aileron rolls through 360 degree bank change is presented in the following table with a few characteristic rolls presented as time histories in Figures 14 through 18 of Appendix I.

ROLLS FROM WINGS LEVEL THROUGH 360 DEGREES

INHERENT AIRCRAFT

(No Dampers or Turn coordinator)

Altitude (feet)	IAS (knots)	Ind. Mach No.	Total Aileron (degrees)	Rate of Roll (deg/sec)	Elevator (degrees)	Angle of Yaw (degrees)	Normal Acceleration (g)
35,000	320	.92	7L	120L	0.15 down	2L	2.0
36,000	310	.92	5L	60L	1.25 up	1R	2.5
37,000	295	.90	5L	85L	1.0 down	7L	1.6 to 0.3
34,500	251	.74	11L	120L	zero	4R to 7.5R	1.8
34,500	250	.74	11L	120L	0.5 down	4.0R to 7.5R	1.8
36,000	245	.74	11L	120L	0.5 up	12R	2.0
34,500	255	.74	11L	115L	1.5 up	14R	0.3 to 2.5
34,500	255	.74	7.5L	100L	2.5 up	10R	2.6
16,000	470	.92	4L	60L	zero	4L	1.9
16,600	440	.88	4.5L	80L	0.75 down	6L	1.7
16,000	470	.92	1.5L	30L	0.4 down	2L	0.5 to 1.2
16,500	465	.92	6L	120L	0.6 up	6L	1.8
17,200	430	.87	5L	100L	0.75 up	2L	3.2
17,500	300	.62	6.5L	80L	0.75 down	8L	2.1 to -.4
17,000	300	.61	5.5L	40L	2.75 up	4R	2.4
16,500	302	.61	4.5L	60L	0.6 down	4L	1.0

ROLLS FROM WINGS LEVEL THROUGH 360 DEGREES

(Yaw Damper and Turn Coordinator Operating)

Altitude (feet)	IAS (knots)	Ind. Mach No.	Total Aileron (degrees)	Rate of Roll (deg/sec)	Elevator (degrees)	Angle of Yaw (degrees)	Normal Acceleration (g)
32,500	404	1.1	8L	165L	1.15 down	9L	2.5 zero
35,500	392	1.11	6.4L	160L	0.9 down	8L 1R	2.6 0.1
34,000	403	1.11	11R	160R	1.5 down	4R	1.8/0.1
33,500	403	1.10	14R	200R	4.0 up	4R/7L	4.1/0.3
36,000	248	.75	12R	115R	zero	4.5L	2.0 1.2
34,500	255	.76	11.5L	130L	1.25 up	4R	2.0
35,500	254	.76	10.3L	120L	zero	4R-5.5L	2.8
35,000	260	.77	12.2R	140R	1.1 up	4.0L 5.5R	2.5
35,000	250	.74	11.8R	130R	0.75 down	3.0R/5.5L	2.0
29,000	312	.80	11.7L	135L	zero	3R	2.7
29,000	317	.81	11.2R	135R	zero	3L	2.3
27,000	337	.82	8L	120L	1 up	4R	1.6
29,000	322	.82	5.3L	120L	0.85 up	2L	2.8
27,500	330	.82	8.0L	110L	0.5 up	4R	1.6
29,500	310	.80	14R	160R	1 up	6.5L	2.7
28,000	326	.83	10.4R	130R	1.2 up	6L	2.0
27,500	336	.83	8.7R	120R	0.85 up	5.5L	2.2
(Yaw Damper, Turn Coordinator, and Sideslip Feed-Back)							
33,500	375	1.05	14R	210R	zero	4R/5L	3.5/0.2
33,000	380	1.05	11L	200L	1.5 down	7.5L	2.8/0.2
34,500	375	1.05	12.5L	205L	2.75 up	8L	3.1
34,000	375	1.05	12.9L	190L	1.6 up	4L	3.7
34,000	375	1.05	12R	180R	1.25 up	2R	3.4
35,500	300	.88	7.5L	120L	1.0 down	2L	1.8
30,000	314	.82	14L	170L	1.7 up	5R	2.5
35,500	315	.92	8L	160L	0.5 up	2.5L	2.0
31,500	290	.79	13.7L	160L	1.65 up	6.5R	1.9
36,500	308	.92	8L	120L	1.75 up	5R	2.8
35,500	308	.90	5L	120L	1.0 up	1R	2.0
35,500	310	.90	7L	80L	1.7 up	3.5R	2.8
30,500	305	.81	15R	175R	2.5 up	9L	2.3
33,000	264	.75	14.8R	150R	zero	6L	1.3
33,000	267	.75	14.5L	160L	1.25 up	4.5R	1.9
31,500	282	.77	15R	155R	1.5 up	8L	2.1
27,000	230	.58	8.5L	90L	1.45 up	1.3R/6L	1.8

These rolls indicate the same characteristics as encountered in rolling through 180 degrees except that less aileron is required for the same roll rates and yawing tendencies with elevator input is more critical. Favorable yaw at high indicated speeds and adverse yaw at low indicated speeds are still evident.

4. Continuous Rolls:

Continuous rolls, beyond a 360 degree change in bank angle, are restricted because of a lack of exploratory tests. During this evaluation, a roll through four complete turns was accomplished with no stability augmenters operating. Seven degrees of aileron were employed at 315 knots (23,000 feet) with only two degrees of favorable sideslip developing. The elevator remained at neutral throughout the roll, however, and it was accomplished near the crossover speed for adverse and favorable yaw. Indications are that rolls at faster or slower indicated speeds will result in highly undesirable maneuvers. This is the danger of inertial coupling in the F-102A.

REQUIREMENTS IN ROLL

1. Air Defense Mission:

The attack phase of the present air defense mission requires a change in bank angle of only 45 degrees. Under extreme conditions, as might be encountered with a bogey below the interceptor, the Control System Tie-In requires a roll of 180 degrees change in bank, a rapid recovery with possible down elevator at the start of the maneuver and up elevator at the completion. The evaluation flights indicate that these maneuvers may be accomplished without restrictions at least in the speed and altitude envelope tested.

Though the inherent aircraft is unrestricted in maneuvers for an ADC mission, it is interesting to note the dynamic characteristics of the aircraft without a yaw damper as seen in Figure 1 of Appendix I. The oscillations encountered at supersonic speeds with the loss of the yaw damper make the success of collision course attack with rocket armament extremely doubtful. A missile attack could probably be accomplished under these conditions, however, this warrants further investigation.

A further consideration is the psychological factor in flying this extremely complicated aircraft which also has unusual roll limitations. Although the effect on individual pilots cannot be measured directly, it is felt that this combination of factors may reduce the combat effectiveness of the aircraft.

An air defense mission in extremes of weather conditions encompasses the danger of inadvertent collision courses with friendly aircraft. The evasive action for this type of incident would be dictated by the existing conditions and pilot's reflexes. However, an escape from this course would require a 180 degree change in bank at most and may be accomplished without restriction.

2. Aerobatics:

Precision flying in aerobatic maneuvers require rolls through at least a 360 degree change in bank angle. The present restriction of 80 degrees per second is entirely too slow to be acceptable for this type of maneuver. The small aileron angles producing this rate of roll at high indicated airspeeds permit this limit to be easily exceeded. The ease with which the F-102A can be rolled at higher rates will provide a strong temptation to exceed this limit. The fact that a certain combination of conditions will provide a well-coordinated roll while another almost similar combination may induce inertial coupling makes aerobatics hazardous.

The limit of 120 degrees per second with improved directional stability (sideslip feed-back on) is considered adequate for rolls through 360 degrees change in bank angle. A requirement for higher rates is not believed essential and this limitation is considered acceptable.

LIMITING AND WARNING DEVICES

The rapid build-up of extreme conditions in aileron rolls because of inertial coupling precludes the use of warning devices. A roll rate limiter would be highly undesirable as full aileron is desired for rapid acceleration in roll, and a "stick snatcher" at the low roll rate of 80 degrees per second would interfere with operation in the region where roll rate should be unrestricted.

The characteristics encountered with sideslip feed-back operating are highly desirable. These characteristics would be preferred if they could be obtained by improving the inherent stability of the aircraft by an optimum size vertical fin without impairing the aircraft performance. If performance were to be impaired too severely, a reliable sideslip feed-back system would be acceptable. If this system were installed, a warning light in the pilot's direct line of vision would be required to indicate when the system were inoperative. A warning light should also be incorporated when the yaw damper and turn coordinator are inoperative.

CONCLUSIONS

The F-102A aircraft with yaw damper and turn coordinator operating can accomplish the basic Air Defense Command intercept problem. Without these devices, tracking is virtually impossible. It is believed that the lack of inherent directional stability in the F-102A aircraft will preclude it from satisfactorily accomplishing a collision course rocket attack when the yaw damper and turn coordinator are inoperative.

The F-102A aircraft, with or without the yaw damper and turn coordinator, is unrestricted in rolls up to 180 degrees change in bank angle. The airplane may be rolled beyond 180 degrees through 360 degrees change in bank with a limiting roll rate of 80 degrees per second. This limit can be easily exceeded inadvertently during a non-operational type mission and could result in damage and possible destruction of the airplane.

The sideslip feed-back as installed on the YF-102A airplane, USAF No. 53-1787, is a definite aid in reducing the inertial coupling effects. The minimum acceptable criteria in roll without inertial coupling is that which can be accomplished with the sideslip feed-back operating. The aircraft in this condition is unrestricted in rolls up to a 180 degree change in bank angle, and may be rolled through 360 degrees of bank with a limit roll rate of 120 degrees per second. A higher roll rate for 360 degrees is not considered mandatory for the mission of this aircraft. A higher roll rate limit may result from further exploratory flight tests in this condition.

Artificial warning devices are impractical for this problem because of the rapid build-up of yaw and normal acceleration with inertial couplings; however, warning lights should be installed to indicate when stability augmenters are inoperative.

Absolute limiting devices would be too restrictive and would limit the ability to accomplish a basic mission.

RECOMMENDATIONS

1. The directional stability of the F-102A aircraft should be improved in order to reduce the susceptibility to inertial coupling effects.

2. An F-102A aircraft should be provided with an increased vertical fin area and should be evaluated as soon as possible by the Air Force Flight Test Center to determine the degree of improvement in stability and loss of performance.

3. The contractor's investigation in inertial coupling should be continued to determine the complete envelope free of adverse inertial coupling effects especially at low altitudes and high dynamic pressures.

4. No absolute limiting device should be incorporated within the lateral control system.

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DISCUSSION

The elevons of the F-102 airplane perform a combined function by operating differentially as ailerons and in unison as elevators. Total aileron was computed by subtracting the left elevon position from the right elevon with down elevon considered negative. The direction of this aileron displacement was chosen as the direction it tends to make the aircraft roll. The time history plots in this appendix have been prepared for one half of the total aileron displacement. It is not a trace of a single elevon.

Elevator displacement is the average of the two elevons with up deflection tending to cause the aircraft to pitch up.

Angle of yaw, as presented, indicates the opposite of sideslip angle as determined with a yaw vane installed on the nose boom. The direction of yaw indicates the direction that the aircraft nose is pointing.

FIG NO 1
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

70° TO 70° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 35,000 FT IAS ~ 395 KNOTS

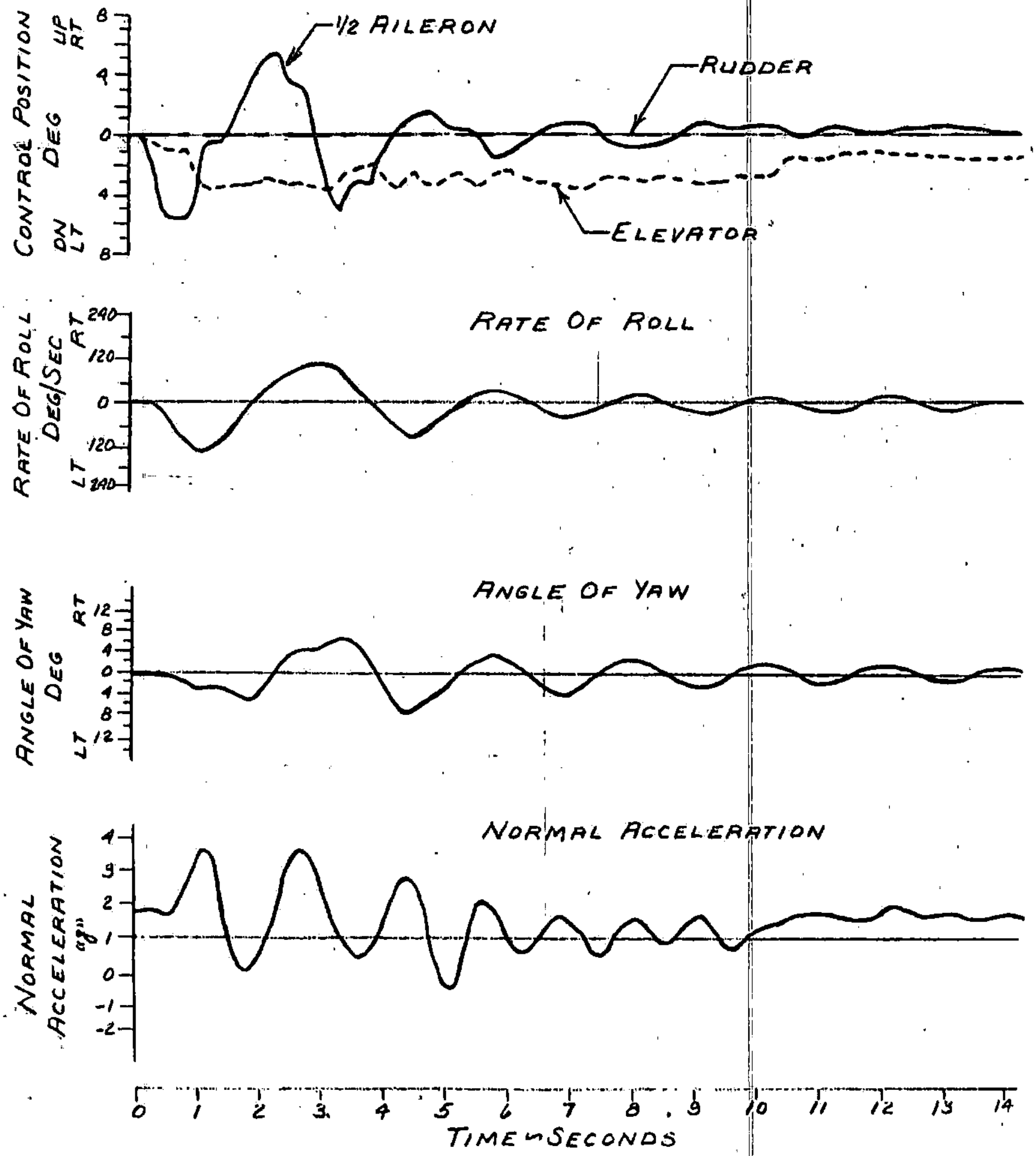


FIG NO 2
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

70° TO 70° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF

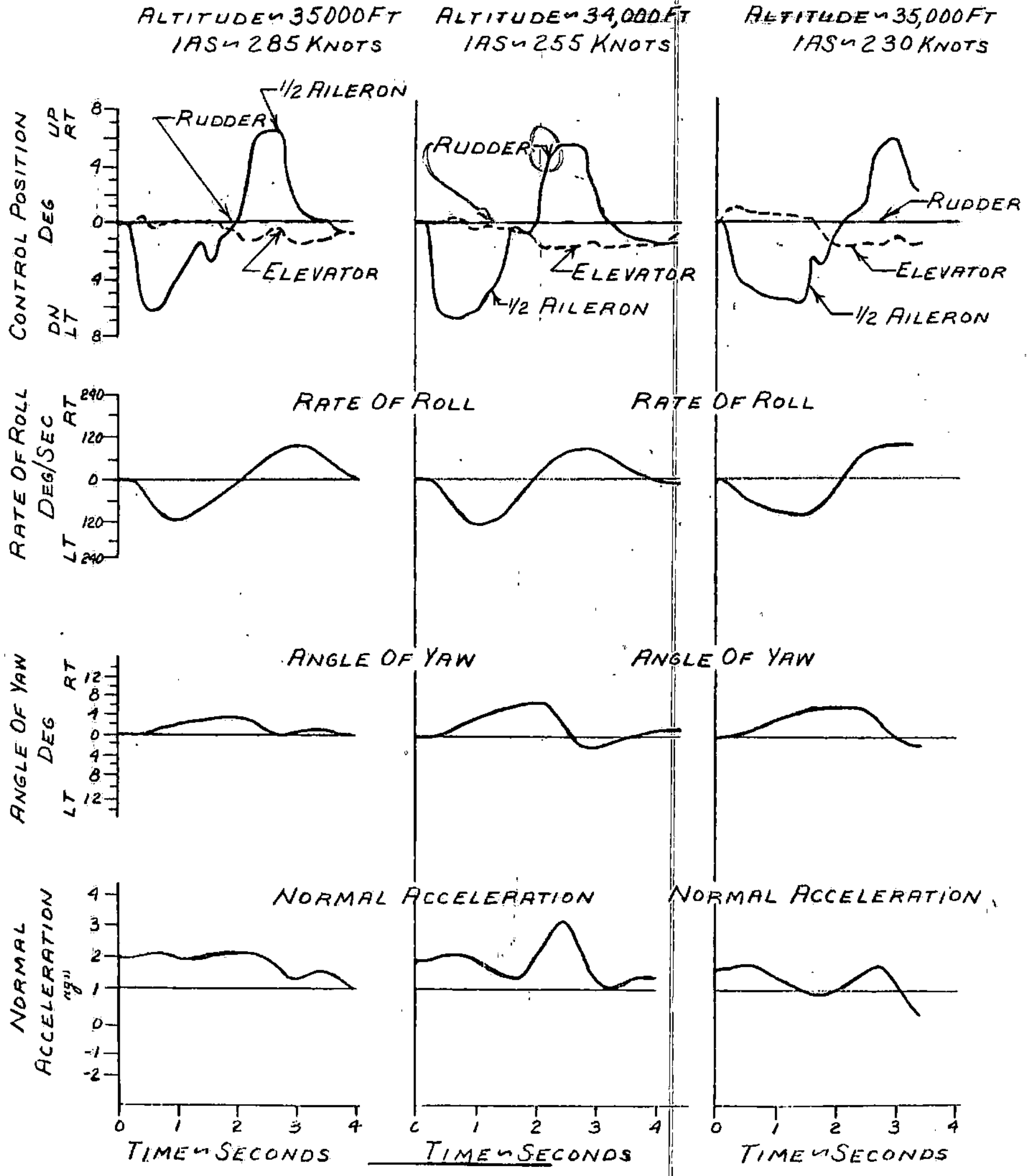


FIG NO 3
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

70° TO 70° BANK & REVERSE	70° TO 70° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF	YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF	SIDESLIP FEED BACK OFF
ALTITUDE ~15,000 FT IAS ~225 KNOTS	ALTITUDE ~16,000 FT IAS ~305 KNOTS

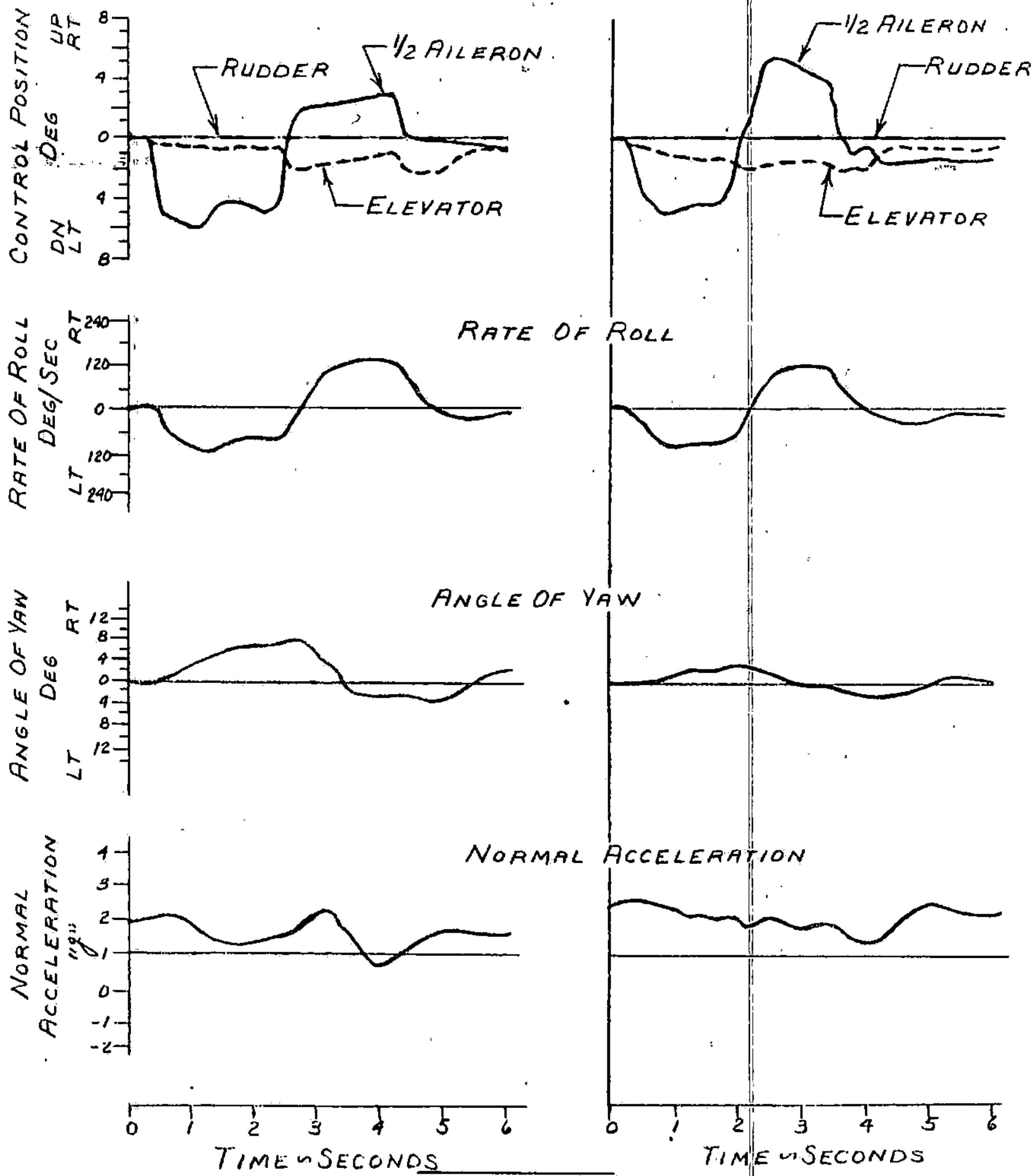


FIG NO 4
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

70° TO 70° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 34,000 FT IAS ~ 335 KNOTS

70° TO 70° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK OFF
ALTITUDE ~ 33,000 FT IAS ~ 345 KNOTS

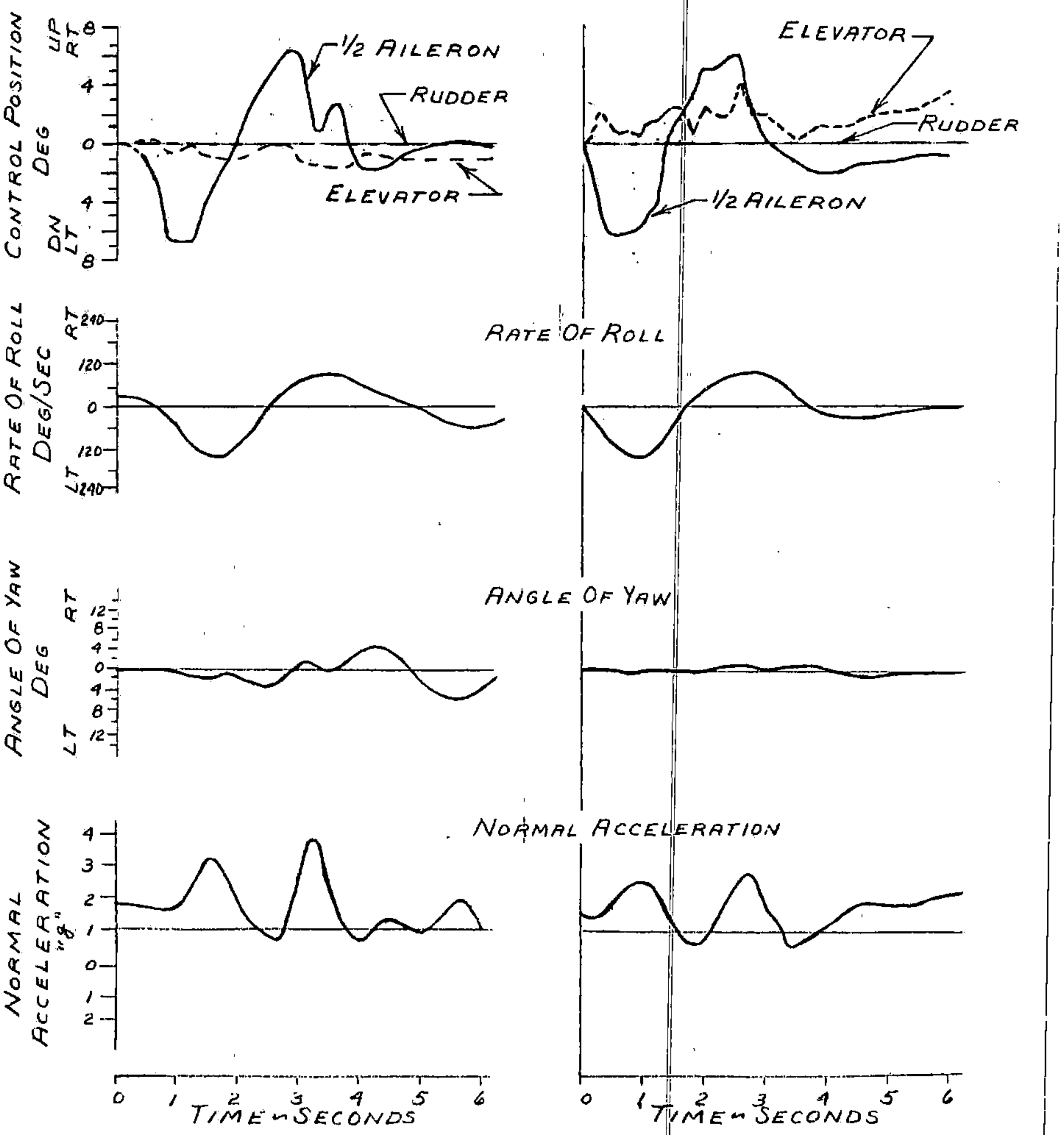
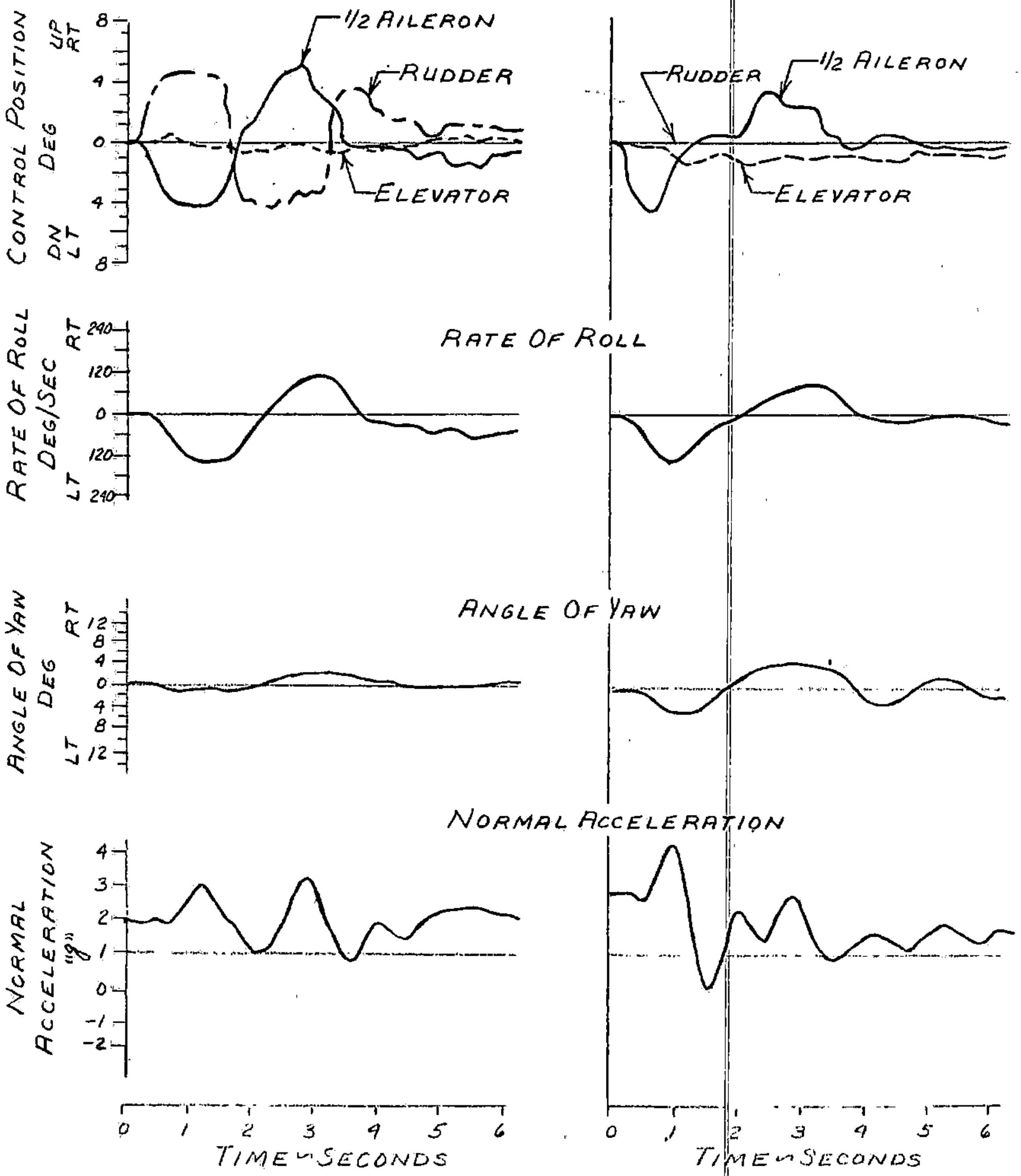


FIG NO 5
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

70° TO 70° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK OFF
ALTITUDE ~16,000 FT IAS ~475 KNOTS

70° TO 70° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~16,000 FT IAS ~475 KNOTS



APPENDIX I

FIG. NO 6
YAW IN ROLLS WITH ELEVATOR DEFLECTION
YF-102A USAF NO 53-1787

NOTE: AILERON ROLLS FROM WINGS LEVEL TO 180 DEGREE CHANGE IN BANK ANGLE.

SYMBOL	YDS.F.C.	IAS	ALTITUDE
◇	OFF	355-365	35000
□	OFF	305-310	35000
◊	OFF	240-260	35000
◈	ON	245-260	35000
⊕	ON	240	46000

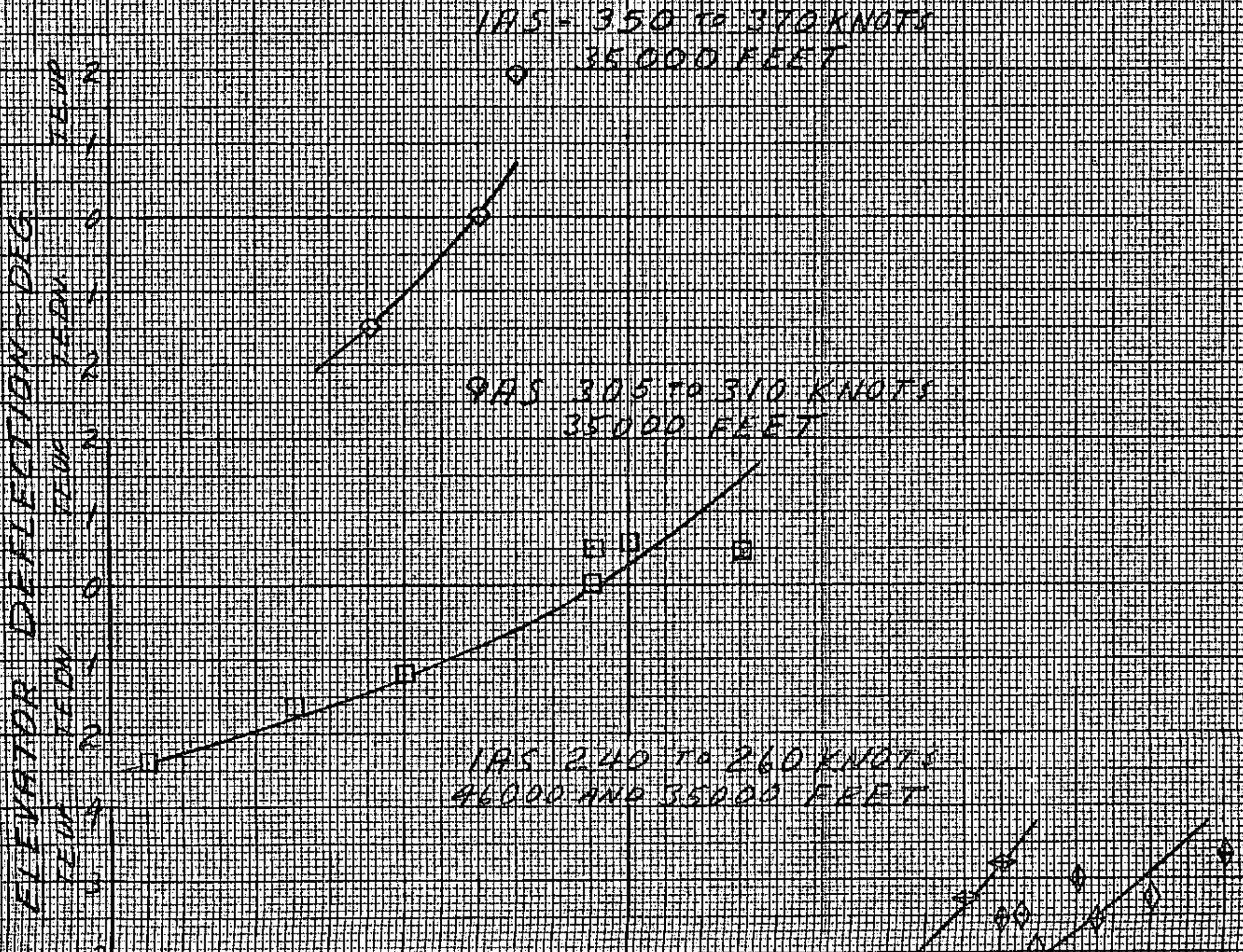


FIG NO 7
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 To 180° & REVERSE	0 To 180° & REVERSE
YAW DAMPER & TURN COORDINATOR OFF	YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF	SIDESLIP FEED BACK OFF
ALTITUDE ~ 36,000 FT IAS ~ 360 KNOTS	ALTITUDE ~ 36,000 FT IAS ~ 365 KNOTS

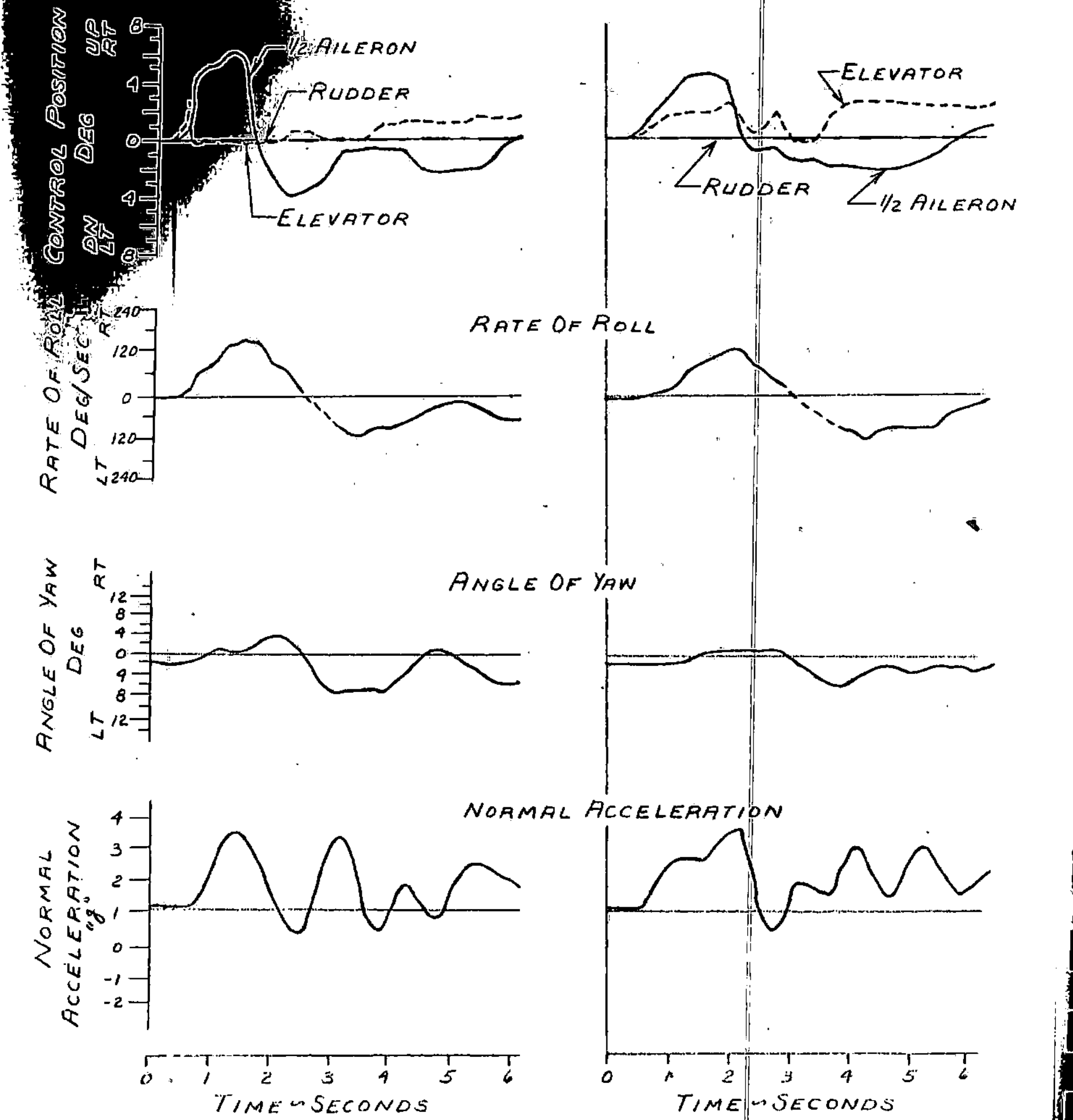


FIG NO 8
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 TO 180° & REVERSE
YAW DAMPER & TURN COORINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 37,000 FT IAS ~ 355 KNOTS

0 TO 180° & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 37,000 FT IAS ~ 360 KNOTS

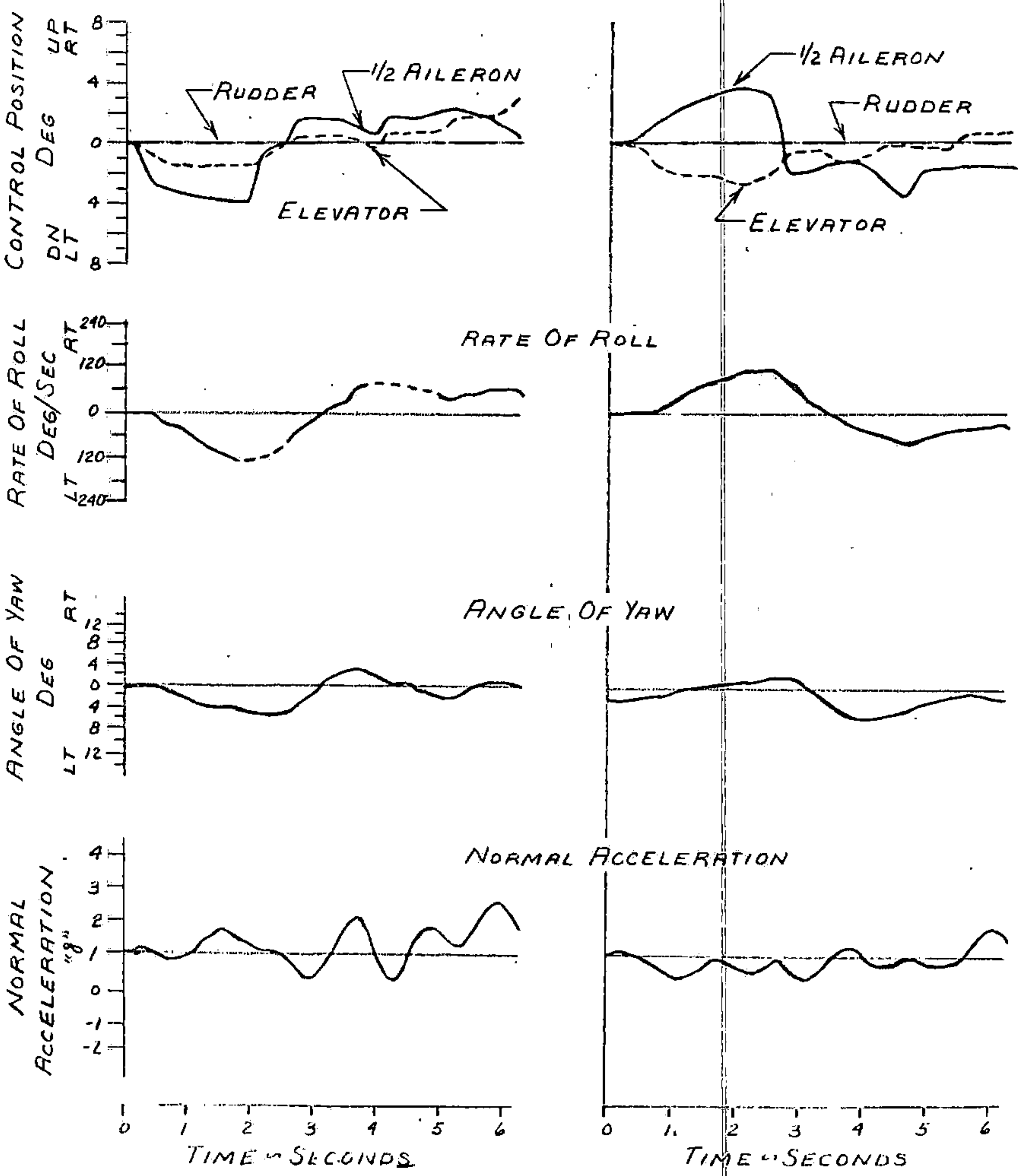


FIG NO 9
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 36,000 FT IAS ~ 310 KNOTS

0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 36,000 FT IAS ~ 310 KNOTS

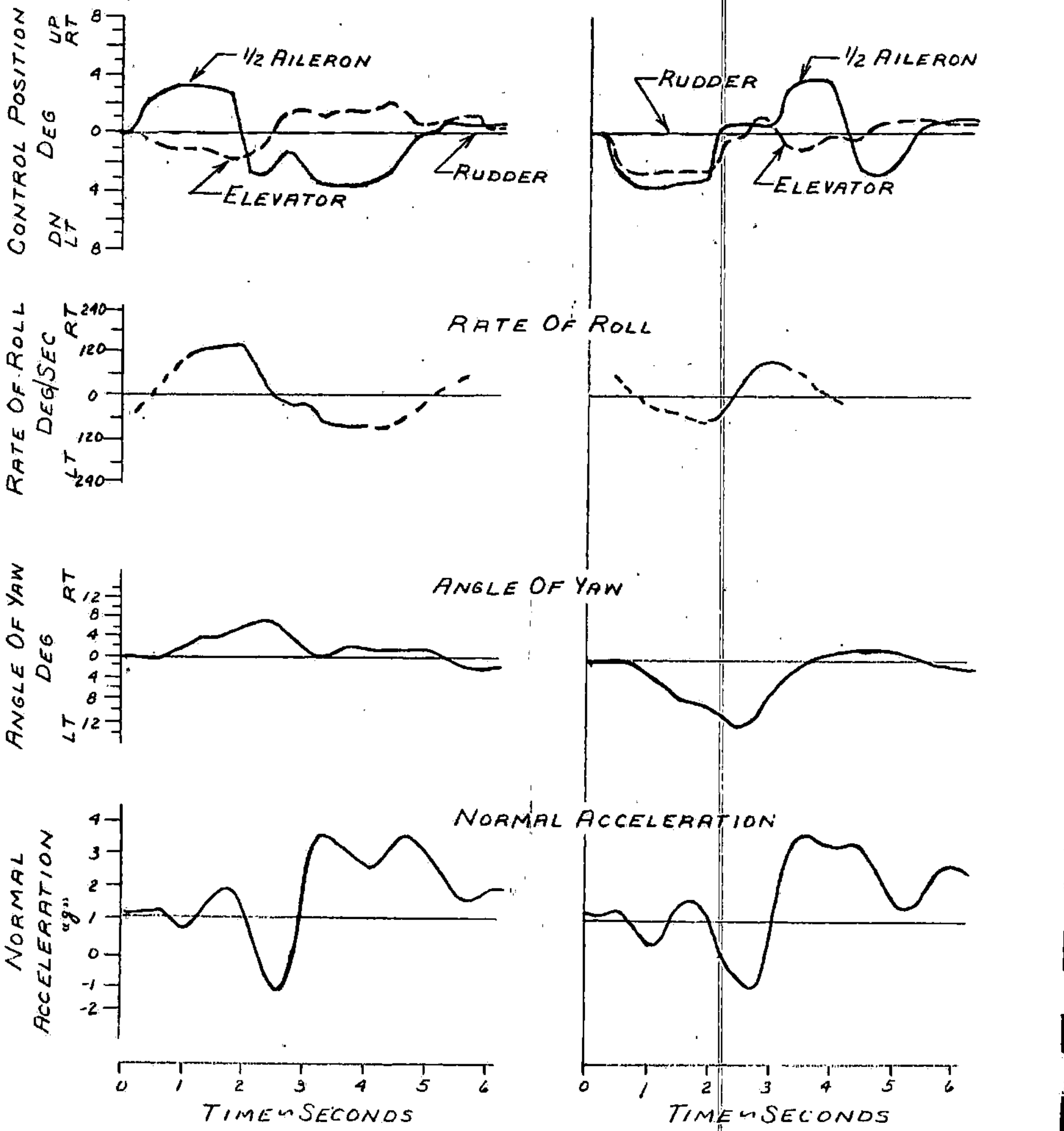
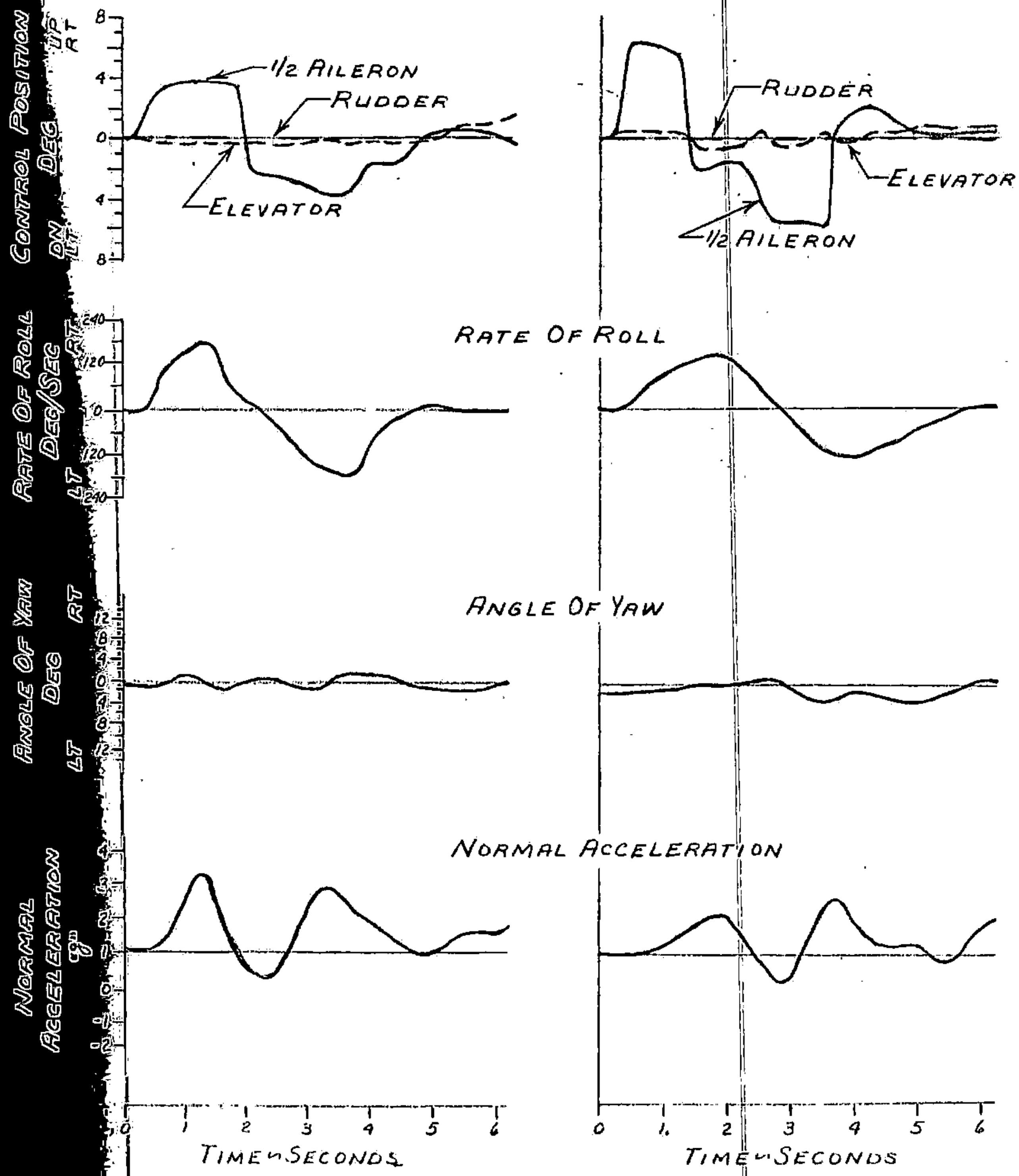


FIG NO 10
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787



0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 35,000 FT IAS ~ 305 KNOTS

0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 36,000 FT IAS ~ 245 KNOTS

FIG NO 10
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 35,000 FT IAS ~ 305 KNOTS

0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 36,000 FT IAS ~ 245 KNOTS

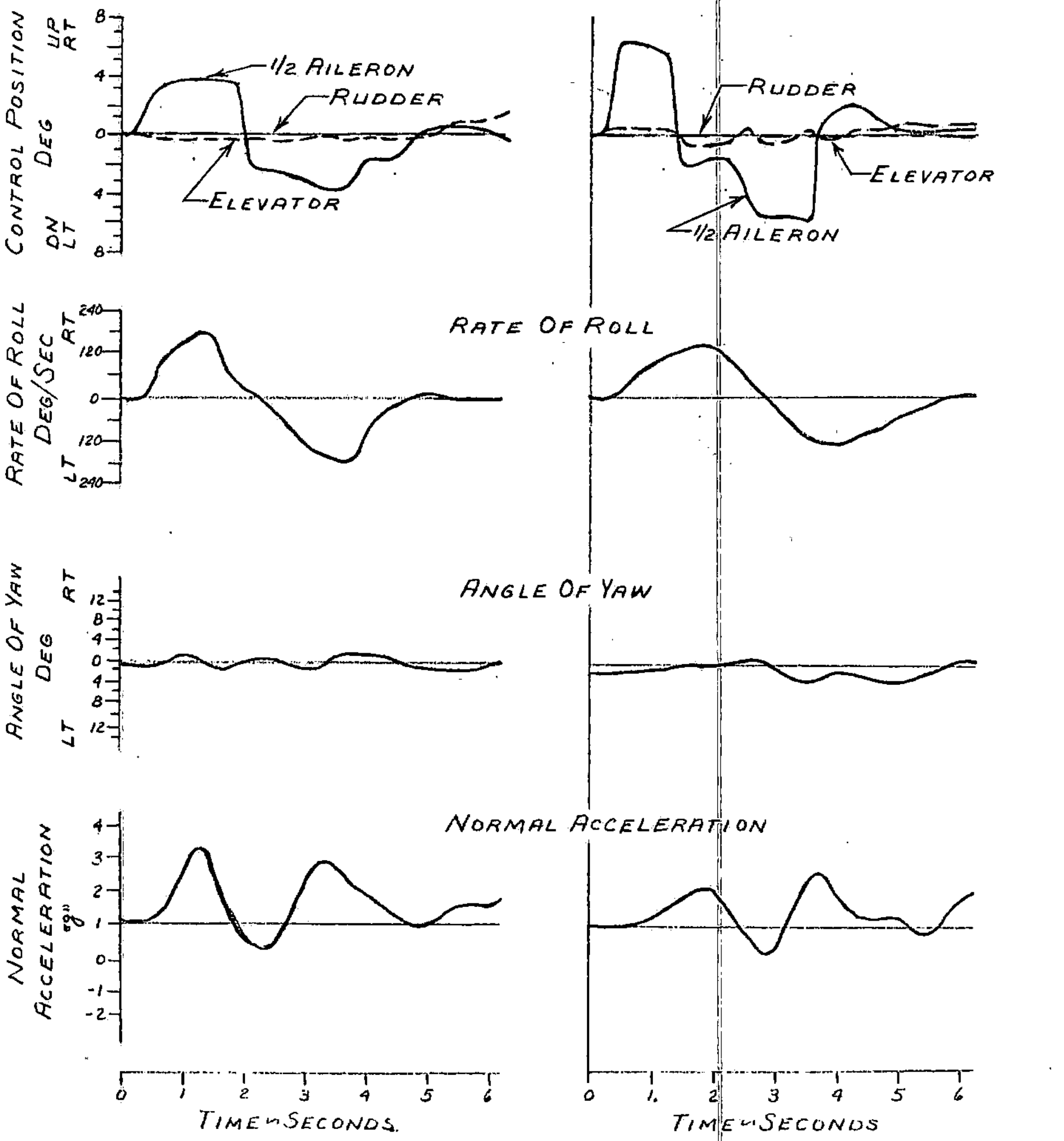
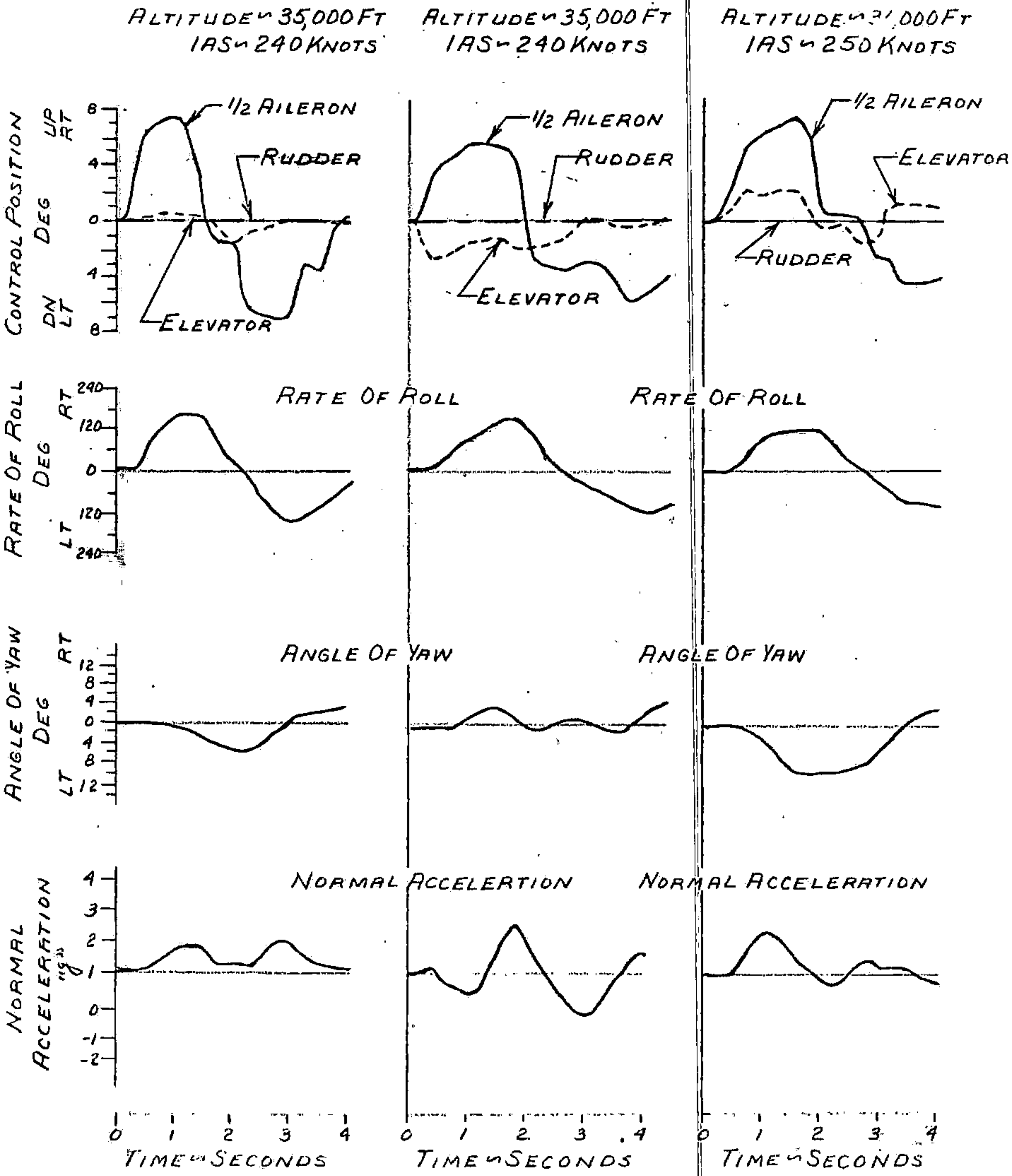


FIG NO 11
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF



APPENDIX I

FIG NO 12

AILERON ROLL HISTORY

YF-102A USAF NO 53-1787

0 TO 180° BANK & REVERSE

YAW DAMPER & TURN COORDINATOR ON

SIDESLIP FEED BACK OFF

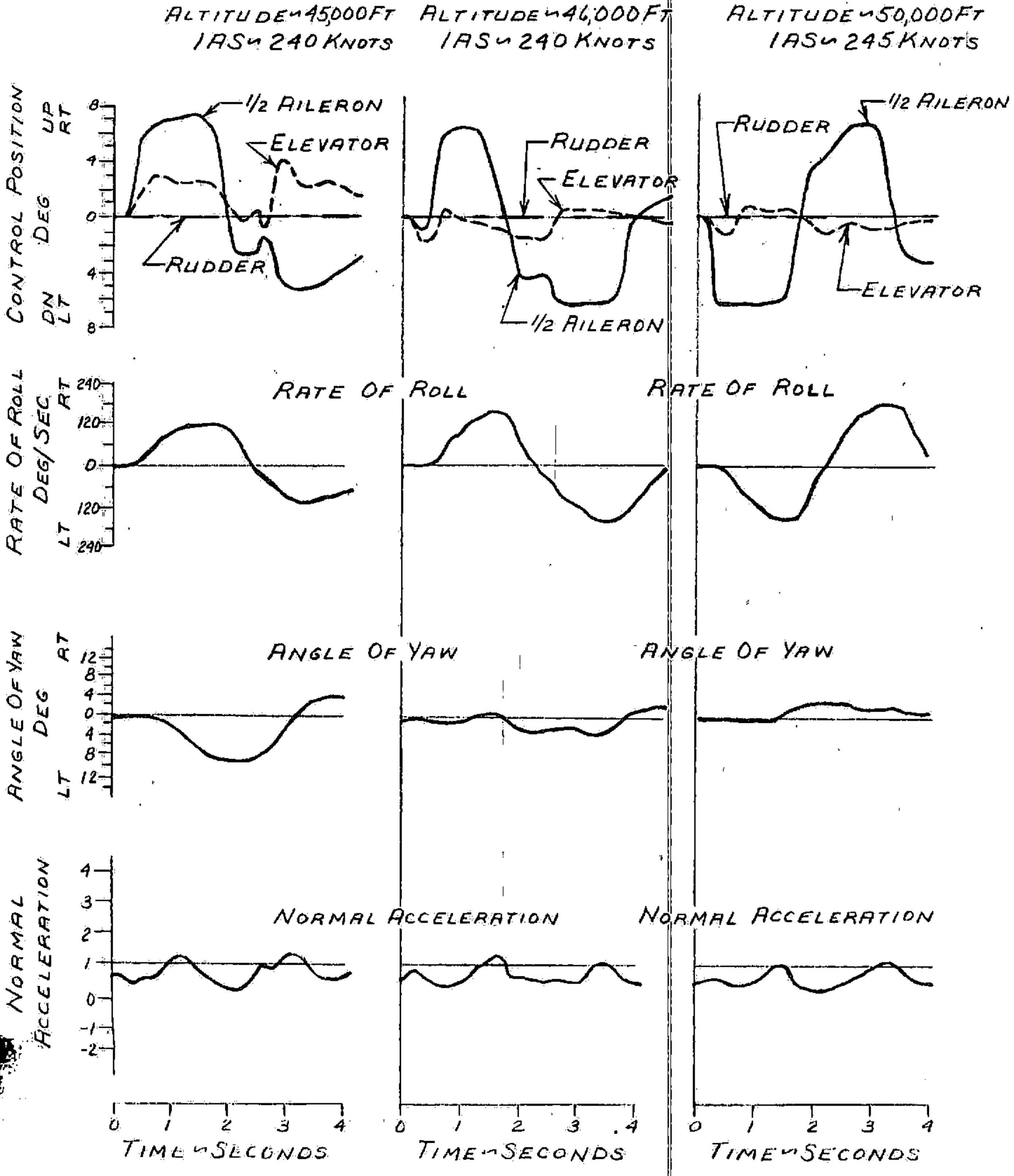
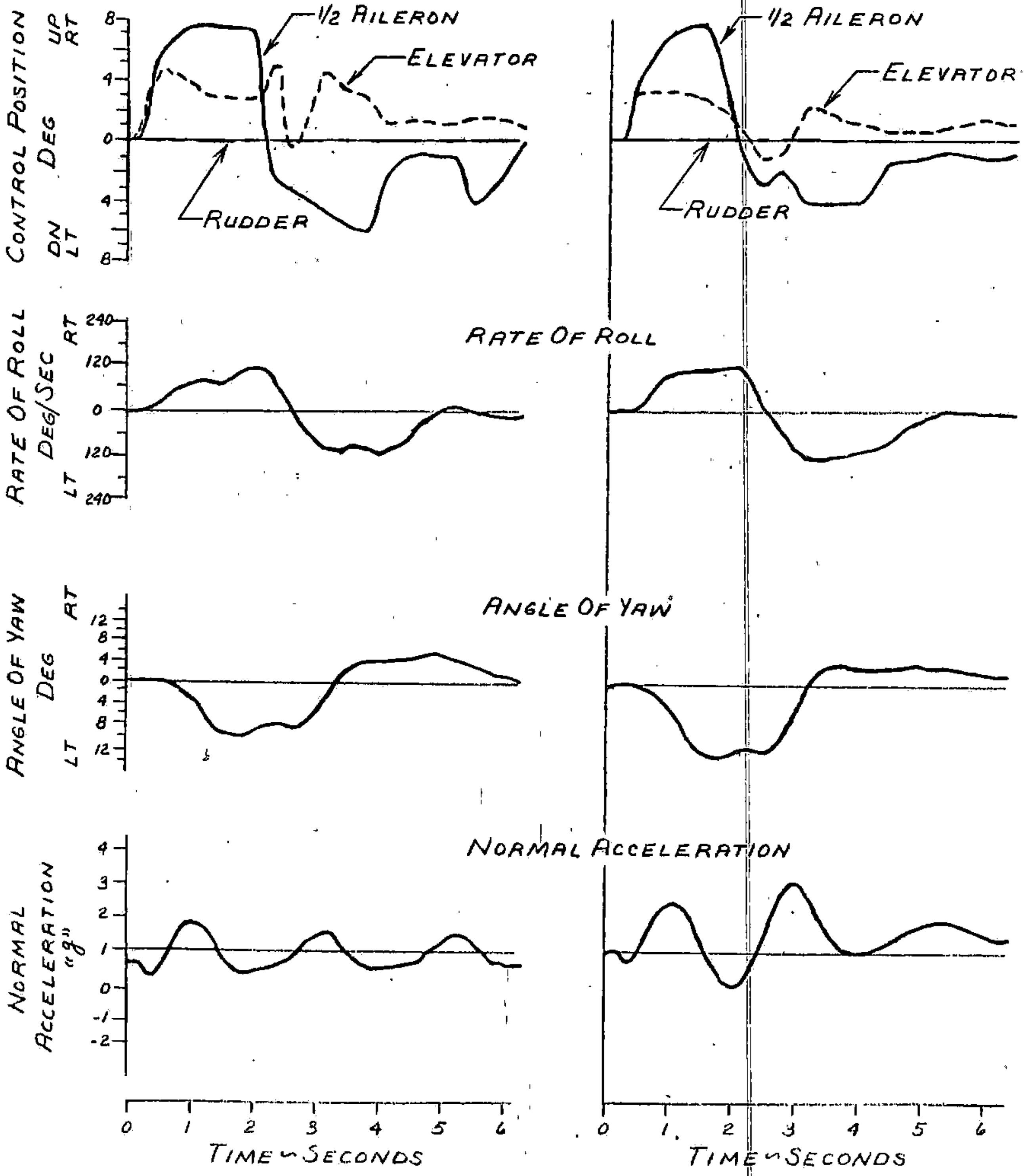


FIG NO 13
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787



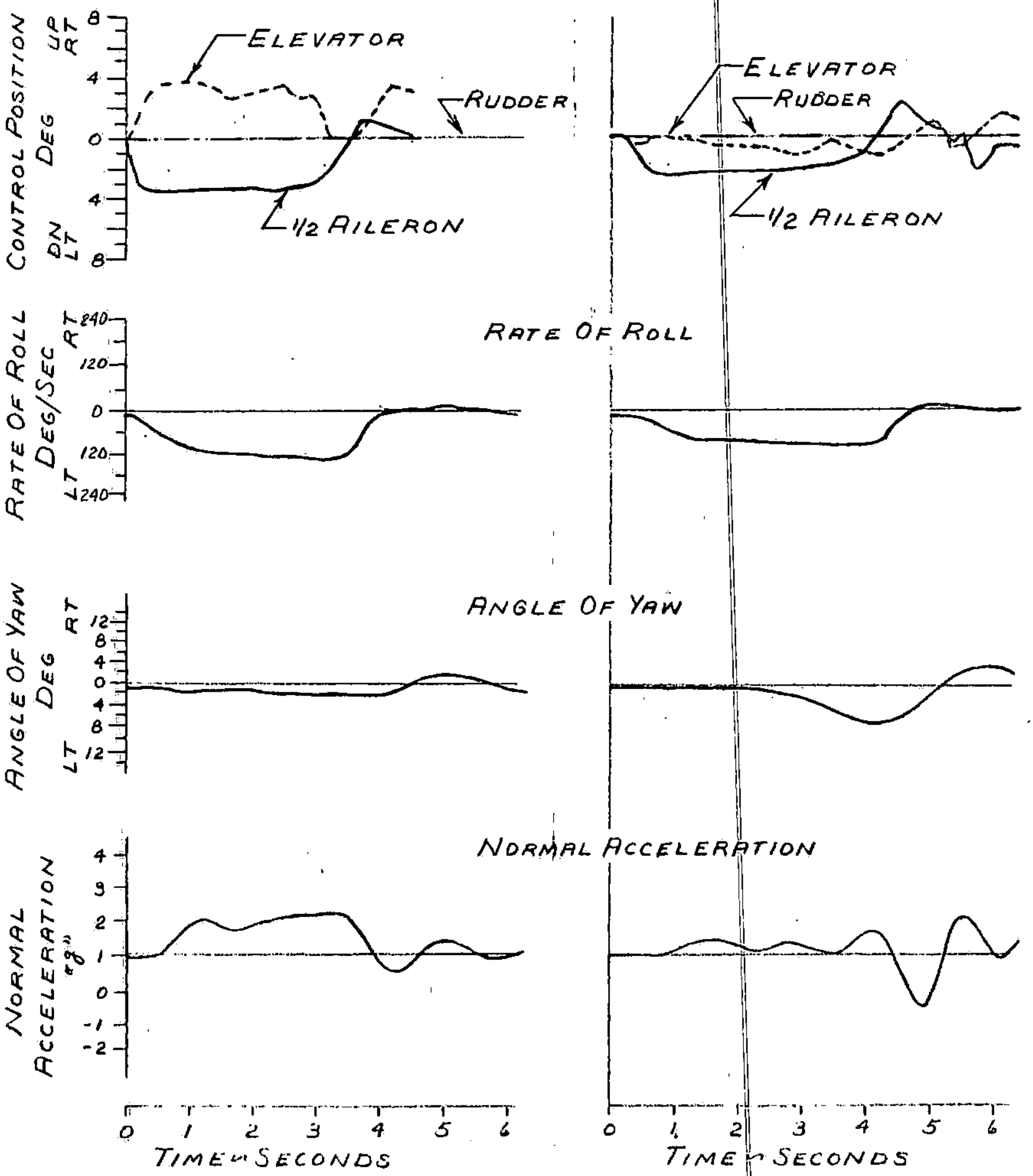
0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK OFF
ALTITUDE ~ 46,000 FT IAS ~ 240 KNOTS

0 TO 180° BANK & REVERSE
YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK OFF
ALTITUDE ~ 35,000 FT IAS ~ 245 KNOTS

FIG NO 14
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 TO 360° ROLL
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 35,000 FT IAS ~ 320 KNOTS

0 TO 360° ROLL
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 37,000 FT IAS ~ 295 KNOTS



APPENDIX I

FIG NO 15
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 TO 360° ROLL
YAW-DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 36,000 FT IAS ~ 245 KNOTS

0 TO 360° ROLL
YAW DAMPER & TURN COORDINATOR OFF
SIDESLIP FEED BACK OFF
ALTITUDE ~ 35,000 FT IAS ~ 250 KNOTS

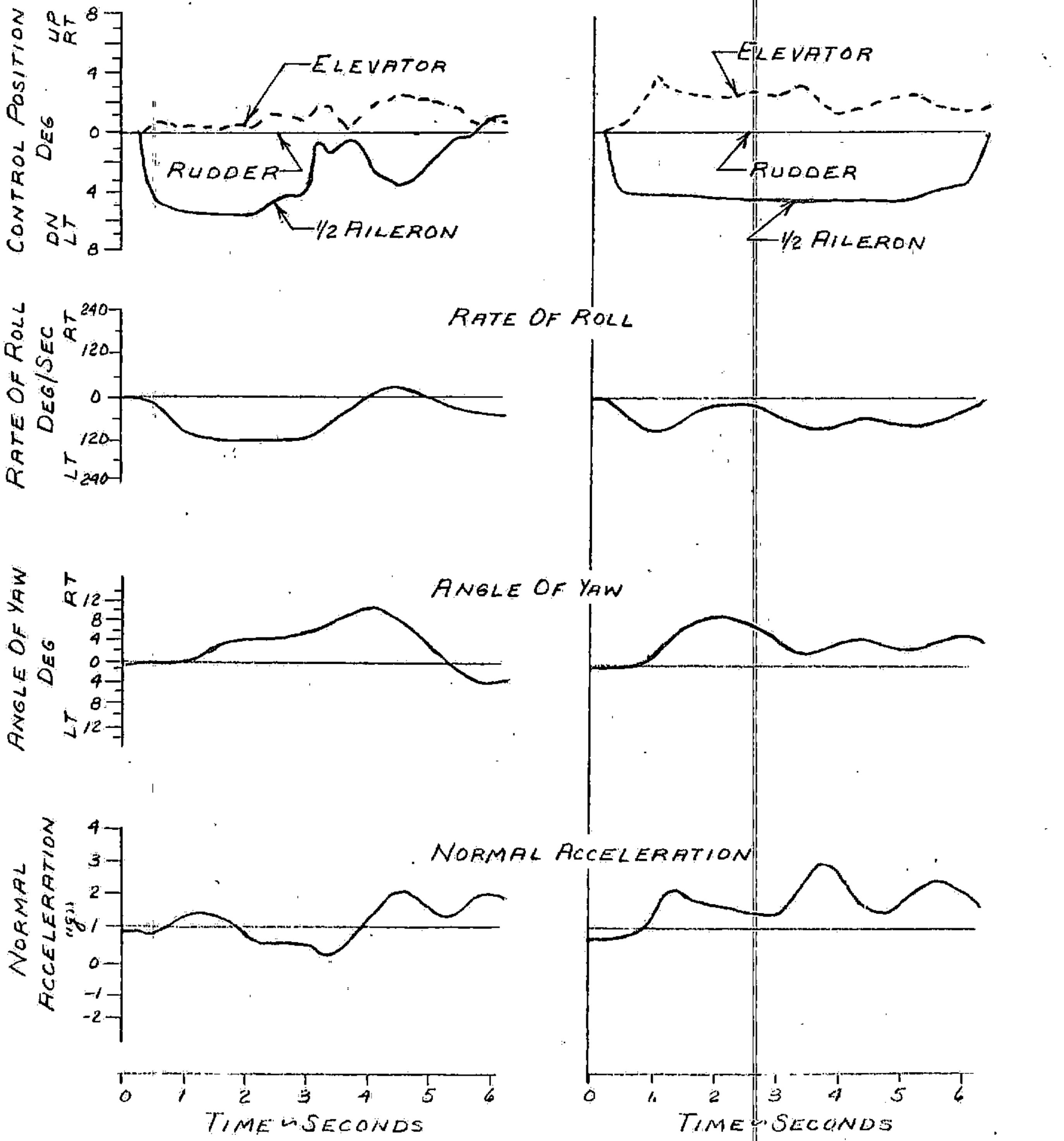
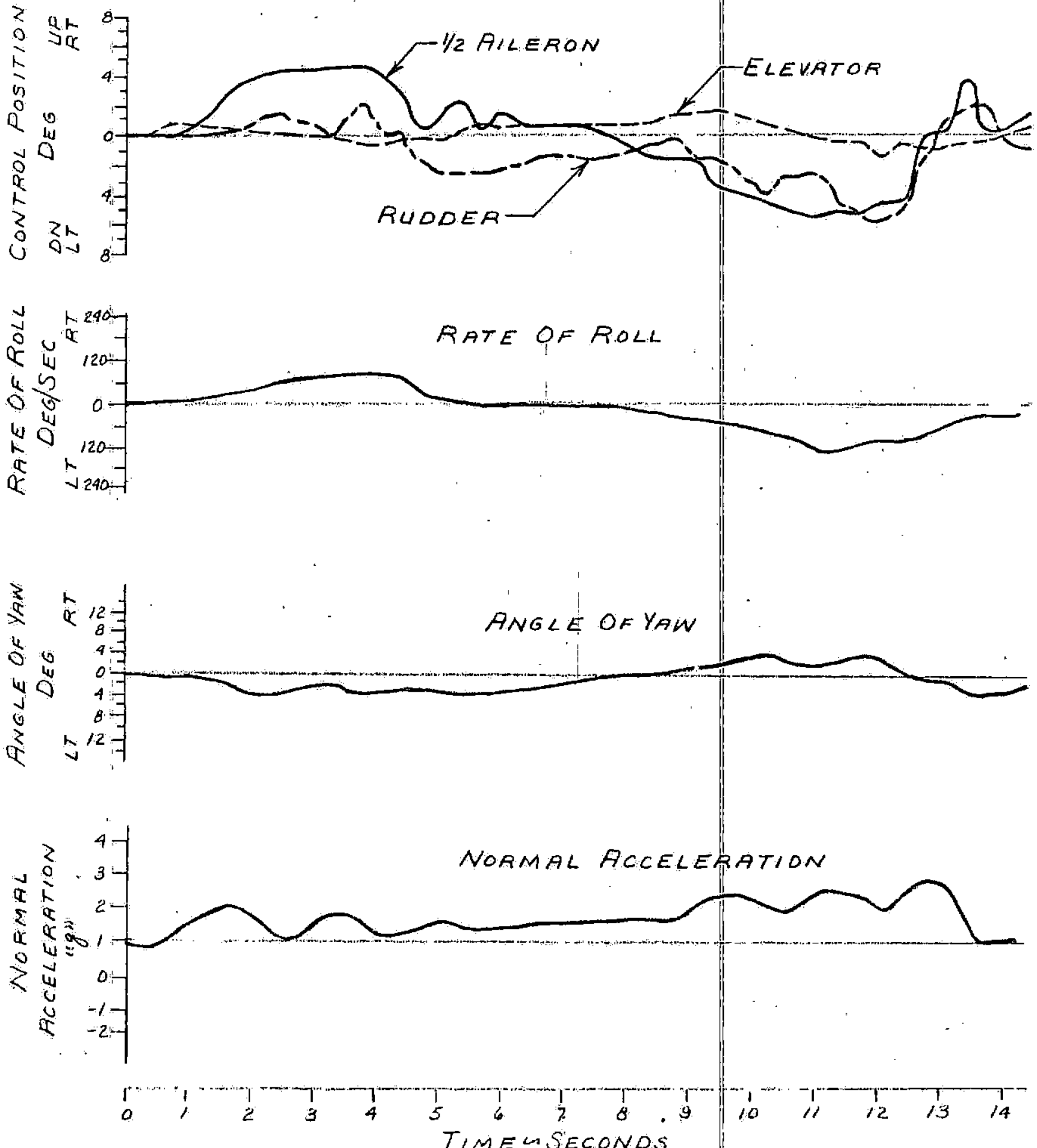


FIG NO 16
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 TO 360° ROLL LEFT & RIGHT
YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK OFF
ALTITUDE ~ 36,000 FT IAS ~ 250 KNOTS



APPENDIX I

FIG NO 17
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

0 TO 360° ROLL	0 TO 360° ROLL
YAW DAMPER & TURN COORDINATOR ON	YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK ON	SIDESLIP FEED BACK ON
ALTITUDE ~ 34,000 FT IAS ~ 375 KNOTS	ALTITUDE ~ 33,000 FT IAS ~ 380 KNOTS

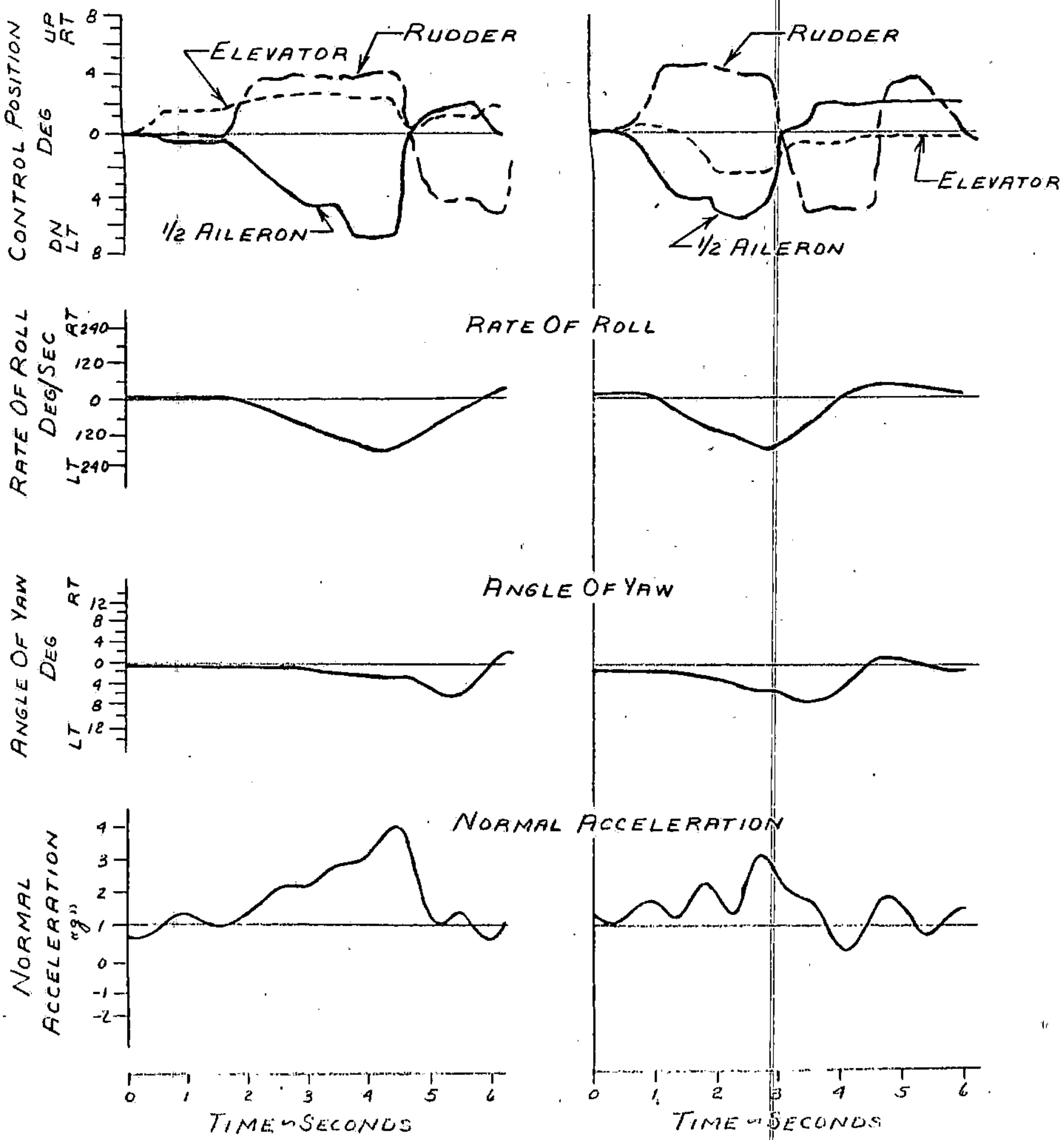
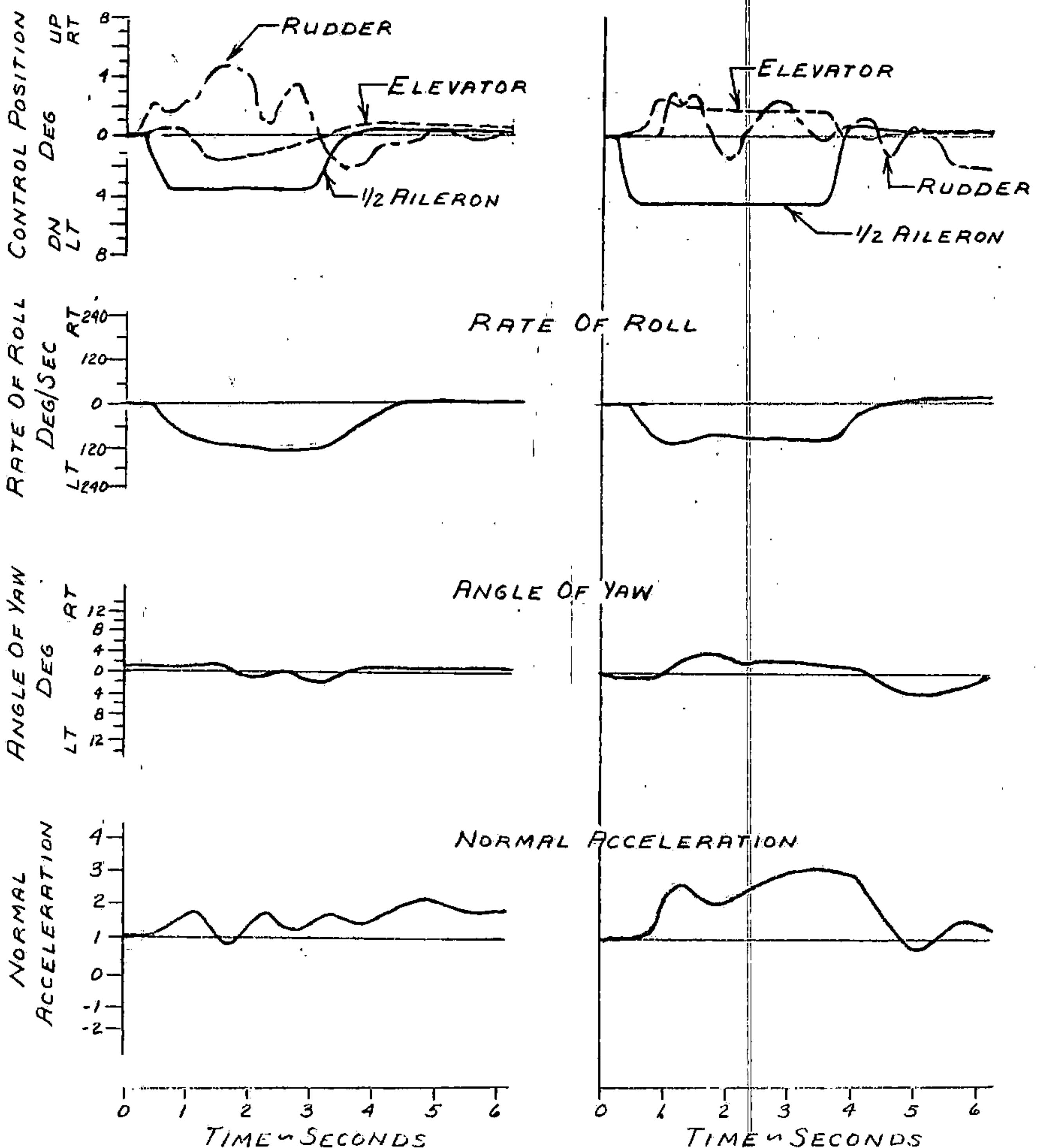


FIG NO 18
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787



0 TO 360° ROLL
YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK ON
ALTITUDE ~ 36,000 FT IAS ~ 300 KNOTS

0 TO 360° ROLL
YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK ON
ALTITUDE ~ 36,000 FT IAS ~ 310 KNOTS

Fig No 19
Aileron Roll History
YF-102A USAF No 53-1787

Extreme CSTI Maneuver
Yaw Damper & Turn Coordinator Off
Sideslip Feed Back Off
Altitude ~ 37,000 Ft   IAS ~ 300 Knots

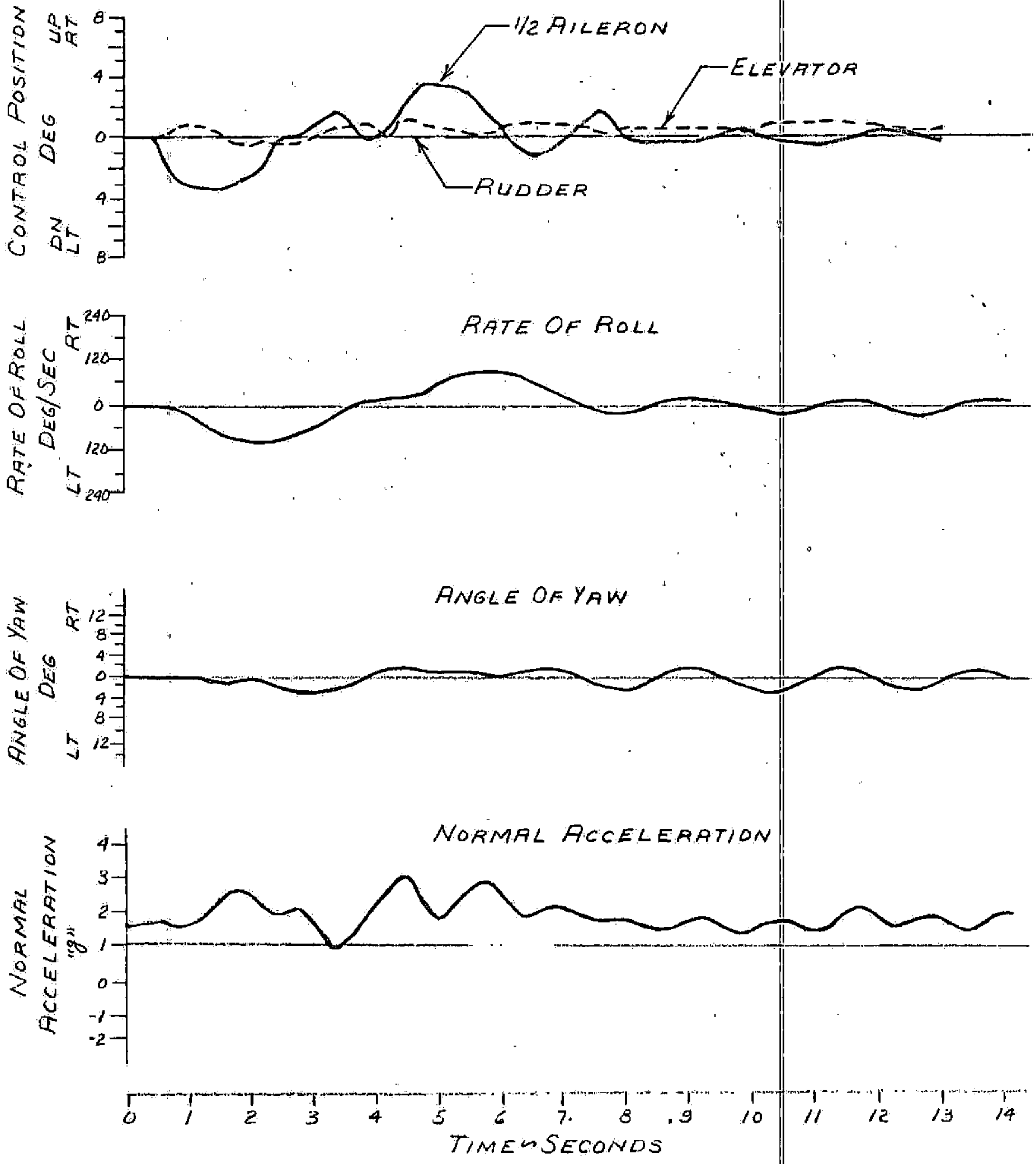
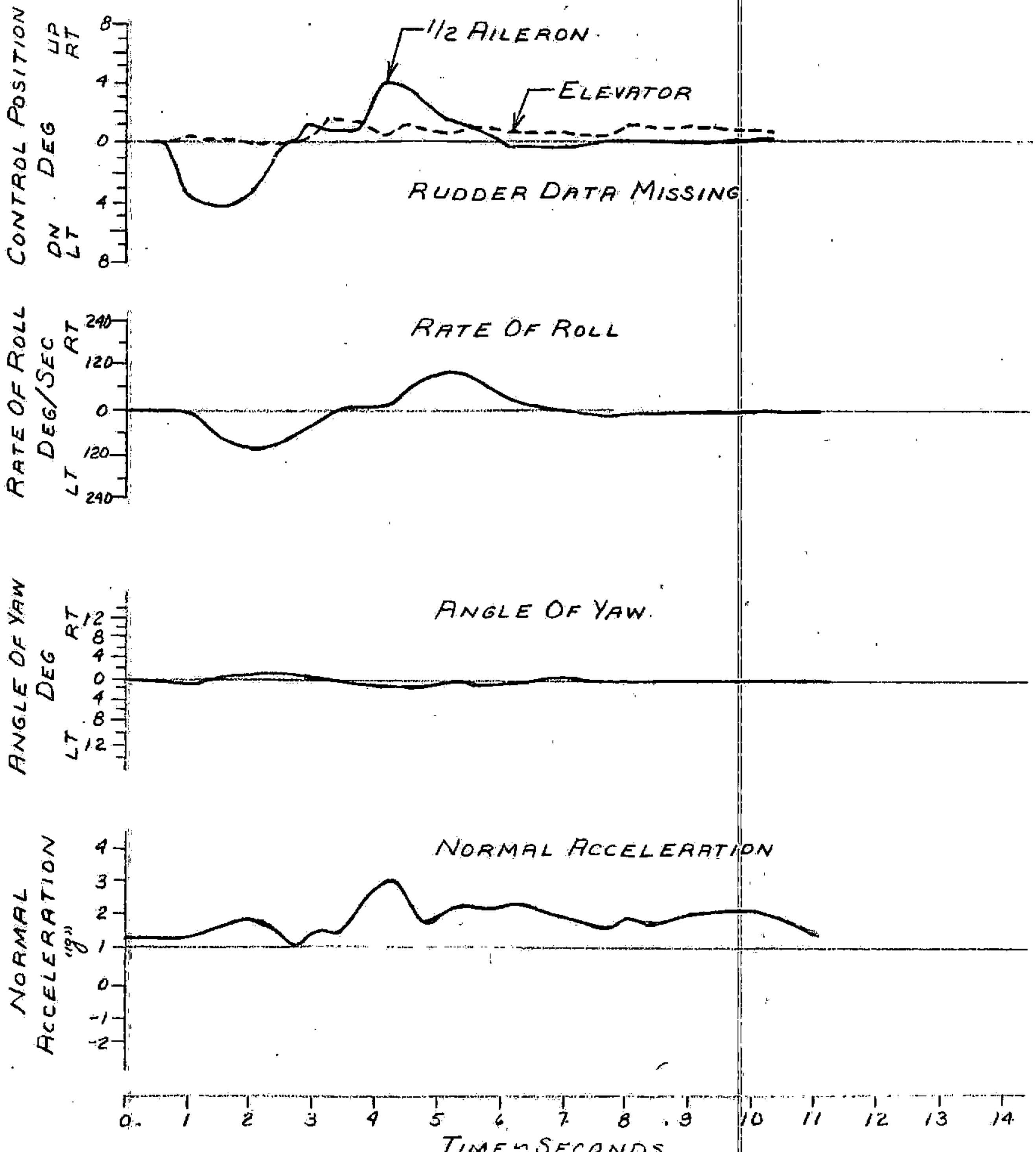


FIG NO 20
AILERON ROLL HISTORY
YF-102A USAF NO 53-1787

EXTREME CSTI MANEUVER
YAW DAMPER & TURN COORDINATOR ON
SIDESLIP FEED BACK ON
ALTITUDE ~ 37,000 FT IAS ~ 300 KNOTS



FLIGHT LOG

YF-102A USAF NO. 53-1787

Flight	Date	Time	Pilot
1	4 Oct 55	1:10	Stephens, R. L., Major, USAF
2	4 Oct 55	:55	Stephens, R. L., Major, USAF
3	5 Oct 55	1:05	Stephens, R. L., Major, USAF
4	5 Oct 55	:45	Everest, F. K., Lt. Col., USAF
5	5 Oct 55	:35	Everest, F. K., Lt. Col., USAF
6	6 Oct 55	1:00	Childs, S. R., Major, USAF
7	7 Oct 55	1:00	Childs, S. R., Major, USAF
8	7 Oct 55	1:00	Hanes, H. A., Col., USAF
9	7 Oct 55	1:00	Baker, R. M., Col., USAF
10	7 Oct 55	:40	Baker, R. M., Col., USAF
11	10 Oct 55	:50	Holtoner, J. S., Brig. Gen., USAF
12	11 Oct 55	1:15	Stephens, R. L., Major, USAF
13	11 Oct 55	1:10	Stephens, R. L., Major, USAF
		12:25	Total

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SAN DIEGO

REPORT ZC-8-535
DATE 27 December 1956
MODEL YF-102C
INDEX 1.9

Classification Changed to:
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Authorized by: SAAMA Date 12-12-66
D D 254 KELLY AFB TEX.,
Reclassified by: Dept., 130-1 Date 4-4-67

TITLE

FLIGHT TEST INVESTIGATION OF
FLUTTER CHARACTERISTICS OF ARMAMENT BAY DOORS

YF-102C USAF S/N 53-1797

PREPARED BY R. E. Dixon
R. E. Dixon

GROUP Engineering Flight Test Reporting and Analysis

CONTRACT NO. REFERENCE AF 33(600)-5942
mK

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R. H. Chausse
Group Engineer

APPROVED BY W W Withee
J.J.M.
W. W. Withee
Assist. Chief of
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Figure 2 - Armament Bay Door Flutter Tests	7

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INTRODUCTION

This report presents the results of flight flutter tests on YF-102C, USAF S/N 53-1797 armament bay doors.

These tests were conducted in compliance with References (1 and 2) on YF-102C, USAF S/N 53-1797 at Holloman Air Force Base, New Mexico during the period 12 March through 10 April 1956, as authorized by CCN MSN 5N-6461 (158) to contract AF 33 (600)-5942.

This report has been prepared in compliance with Air Force Specification 40141, paragraph E-2b (9)b.

PURPOSE

Armament bay door flutter tests were conducted to extend the present restrictions on door operations. Previous speed limitation on this airplane had restricted door operation to 0.80 Mach number or 240 knots IAS, whichever occurred first (Reference 3).

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SUMMARY

A total of ten (10) flights were conducted on YF-102C, USAF S/N 53-1797 to determine the stability of armament bay doors in the Mach number range from 0.80 to 1.20 at altitudes between 20,000 and 40,000 feet.

Analysis of the data indicates that the main armament bay door is free from flutter at Mach numbers to 1.2 or indicated airspeed of 580 knots, whichever occurs first.

CONFIGURATION

The main armament bay door on this airplane is 21 inches wide, 231 inches in length and weighs 60 pounds. This door is shorter, wider and lighter than the standard F-102A armament door. It is also operated by two (2), rather than three (3) actuating cylinders.

The results of flight flutter tests on the standard F-102A armament door (Reference 4) were considered insufficient because of these differences in door configuration.

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DISCUSSION OF RESULTS

Results of flight flutter tests are best described in terms of the damping coefficient (g), which is defined by the following equation:

$$g = \frac{-\log_e \frac{X}{X_1}}{n \pi}$$

where: X = amplitude at some point on the response curve.

X_1 = amplitude at some succeeding point on the response curve.

n = number of cycles between points X and X_1

From the above equation it can be seen that (g) will be negative for decaying oscillations and positive for divergent oscillations. Therefore, a negative damping coefficient (g) denotes stability.

Damping coefficients were computed from oscillograph recordings of decay response measured by strain gages on the rod ends during door operation. When the actuating pistons bottomed in the full-opened position, a transient oscillation was induced. Figure 1 shows a typical oscillograph recording trace of decay response during armament bay door opening sequence. Figure 2 shows the damping coefficient (g) as a function of true Mach number for ground level and at 20,000 and 40,000 foot altitudes.

Analysis of the door flutter tests show that the door possesses satisfactory damping for all conditions tested. The low altitude, high speed tests showed evidence of considerable forced oscillation due to turbulent air flow. However, the amplitudes of the forces were small in comparison with the design loads for the actuator cylinders.

CONCLUSIONS

1. Satisfactory damping has been demonstrated for the Mach number range of 0.80 to 1.20 and for altitudes between 20,000 and 40,000 feet.

2. The armament doors may be operated without objectionable flutter characteristics at any speed below 580 knots I.A.S. or 1.2 Mach number, whichever is lower. This restriction is based on the aircraft equipped with a five (5) foot test boom with a production type pitot static head.

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REFERENCES

1. CONVAIR Report ZC-8-166. Flight Test Program, YF-102C USAF S/N 53-1797. 1 February 1956.
2. CONVAIR Memorandum EFT-P-8-342 (Revised). Flight Test Program, YF-102C USAF S/N 53-1797, Missile Bay Door Flutter. 5 March 1956.
3. CONVAIR Memorandum DF-8-186 (Revised). Missile Bay Door Flutter Restrictions on F-102A Airplanes USAF S/N 53-1797 and S/N 53-1806. 21 February 1956.
4. CONVAIR Report ZC-8-512-83. Flight Operations Report F-102 Development Program. Pages 15, 24 and 25, Missile Bay Door Flutter Characteristics, F-102A USAF S/N 53-1793. 5 October 1956

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REPORT NO. ZC-8-535

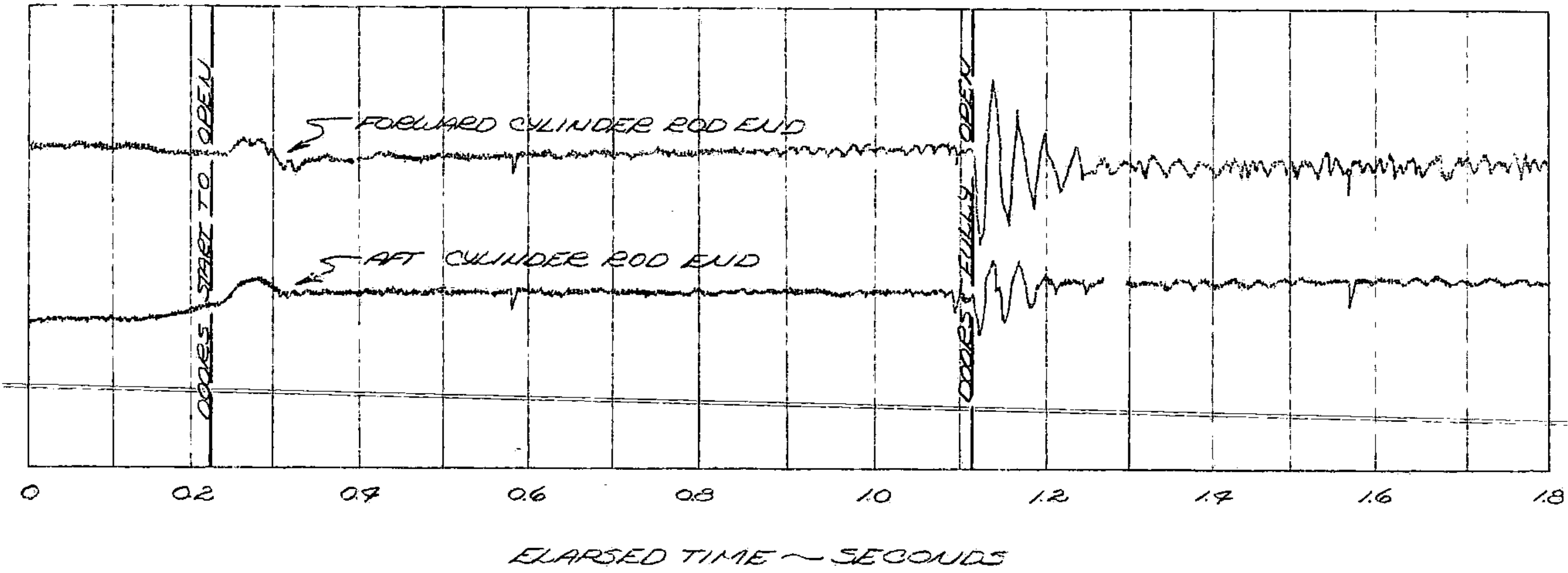
FIGURE 1 PAGE 6

TYPICAL ARMAMENT BAY DOOR FLUTTER HISTORY

YF-102C S/N 53-1797

NOTES:

1. FLIGHT CONDITIONS WERE:
   TRUE MACH NUMBER = 0.94
   TRUE PRESSURE ALTITUDE = 40560 FT.
   INDICATED CORRECTED AIRSPEED = 277 KTS
2. STRAIN GAGES WERE LOCATED ON THE FORWARD & AFT ARMAMENT BAY DOOR ACTUATING CYLINDER ROD ENDS.



BY: mme
CKD: RED
EF-8-1797-236

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FIGURE 2 PAGE 7

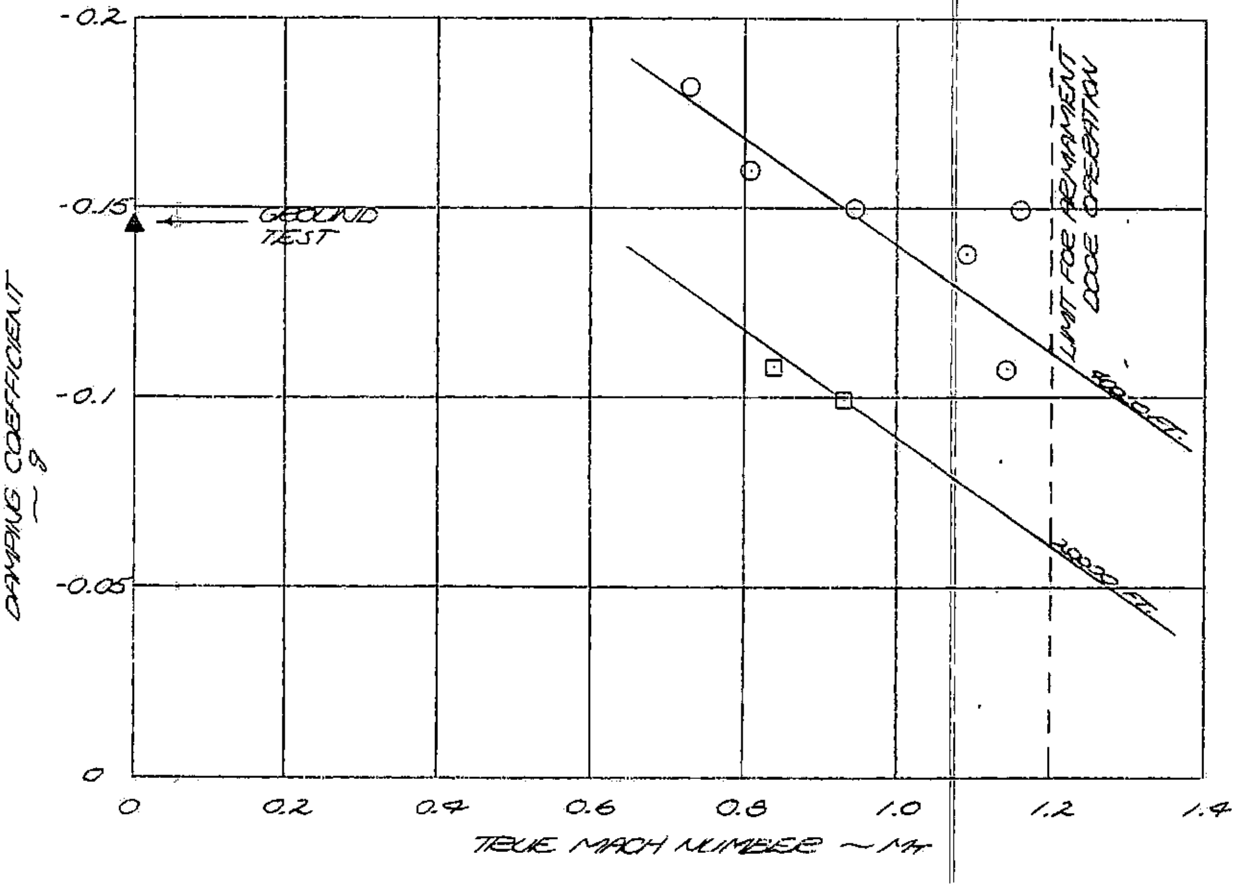
ARMAMENT BAY DOOR

FLUTTER TESTS

YF-102C S/N 53-1797

FLT NO.	g	M_T	V_{IC} ~KNOTS	H_P ~FEET	q_c ~PSF
9	-.162	.724	213	39940	162
	-.160	.810	243	39200	215
	-.150	.942	277	40560	264
10	-.134	1.092	356	38810	462
	-.108	1.142	360	39220	532
12	-.103	.843	376	20750	546
13	-.150	1.157	403	36230	604
	-.097	.931	420	20760	683
GRND TEST	-.146	—	—	—	—

LEGEND:
- ⊡ 20000 FT.
- ⊙ 40000 FT.
- ▲ GROUND TEST



NOTES:

1. DAMPING COEFFICIENT $(g) = \frac{-\log_e \frac{x}{x_1}}{n\pi}$, WHERE $\log_e \frac{x}{x_1}$ IS THE LOGARITHMIC DECREMENT OF THE DECAY RESPONSE, AND n IS THE NUMBER OF CYCLES FROM x TO x_1
2. A NEGATIVE DAMPING COEFFICIENT DENOTES STABILITY.
3. DATA WERE OBTAINED FROM STRAIN GAGE RESPONSE RECORDED ON AN OSCILLOGRAPH DURING DOOR OPENING. STRAIN GAGES WERE LOCATED ON THE FORWARD ARMAMENT BAY DOOR ACTUATING CYLINDER ROD END.

BY: JPB
CK'D. RLO

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A DIVISION OF GENERAL DYNAMICS CORPORATION

SAN DIEGO

REPORT EFT-A-8-480

DATE 13 August 1957

MODEL TF-102A

INDEX NO. 1.13

TITLE

Flight Test Results of TF-102A Stall and Spin Tests

TF-102A S/N 54-1362

PREPARED BY J. Weissman
J. Weissman

GROUP Engineering Flight Test Reporting and Analysis

REFERENCE Contract Number AF 33(600)-5942

CHECKED BY R. H. Chausse
R. H. Chausse
Ass't Flight Test Group Engineer

APPROVED BY M. I. Edelstein
M. I. Edelstein
Flight Test Group Engineer

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INTRODUCTION

To demonstrate the stall and spin characteristics and to determine the optimum recovery procedure for the TF-102A aircraft, testing was accomplished in compliance with USAF specifications 1815B and 1816 (References (a) and (b) respectively). Stall and spin testing were accomplished, concurrently, at Edwards Air Force Base, California, during May and June 1957. TF-102A USAF S/N 54-1362 was utilized for these tests. The purpose of this memorandum is to present and discuss the results of these tests which were done in accordance with Reference (c).

Previously published memoranda (References (d), (e), (f) and (g)) have discussed spin recovery equipment testing, exploratory spin testing of the F-102A aircraft and the specification compliance demonstration of the F-102A.

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SUMMARY

Both ground and inflight deployment and jettison of the spin recovery parachute were accomplished. In each case, functioning of the primary systems and secondary systems was checked. Satisfactory operation of the systems were obtained in all cases. The blossoming of the canopy, too, on every deployment was considered good.

The minimum airspeed during stall testing was 103 knots indicated; the maximum rate of descent was approximately 10,000 feet per minute. The pilot reported conventional stall characteristics of buffet and slight longitudinal, directional and roll oscillations.

With the aircraft loaded fuselage heavy and wings light, aileron against spins resulted in flatter spins developing than were encountered from other types of entry controls. This higher angle of attack (approximately 60°) appears to result from a faster yaw rate. Spins to the left were less severe than spins to the right. When entry was made using rudder alone or rudder and aileron "with", the angle of attack during the spin was at about 35° nose up. The turning rate in this attitude appeared less than in the flatter spin.

Recovery was effected when the pilot released the controls, permitting the surfaces to neutralize. The aircraft stopped turning within 1-1/2 turns after this action. In most cases, less than one turn was sufficient.

With the light fuselage loading, the aircraft spun with a high angle of attack, approximately 60° nose up in the developed spin. In this attitude, the aileron determined the direction of the developed spin. Roll rate and other aircraft characteristics were much the same for similar spins at the more forward cg loading. This was true in both the clean and the landing configuration. Recoverys from such spins in the clean configuration were completed in less than 1-1/4 turn. In the landing configuration, recovery in the first test took almost two turns. On subsequent flights, either when using aileron with or just neutralizing controls, the aircraft stopped turning in less than one turn.

The data obtained during spin testing are tabulated and summarized in Figures 1a, 1b and 1c.

Satisfactory effect of the spin recovery parachute was recorded, when, on the final test, pro spin controls were held and recovery effected by the parachute in less than one turn.

SUMMARY (Cont'd)

Accelerations encountered during the spins were not excessive. The most severe were pitch oscillations, which, on one flight, reached a maximum normal acceleration of +3.25 g's. Lateral and longitudinal accelerations recorded were insignificant on all flights.

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Nomenclature

Aileron Deflection	One-half the included angle between the right and left elevons
Right Aileron	Aileron deflection which produces right wing down in conventional flight attitudes
Aileron With	Aileron which would produce a roll in the direction of the spin if the airplane were in a conventional flight attitude (right aileron in a right spin)
Aileron Against	Aileron which would produce a roll opposite to the direction of the spin if the airplane were in a conventional flight attitude (left aileron in a right spin)

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CONFIGURATION

Aircraft S/N 54-1362 is a production TF-102A, modified to incorporate a spin recovery parachute and an emergency power source to assure operation of the primary hydraulic system and the instrumentation. The spin recovery parachute was installed on a ramp located in place of the speed brakes and drogue chute. It is a 24 foot ring-and-slot type with a 65 foot riser.

A dual RF FM/FM telemetering package and a 28-hole photopanel were installed in the aircraft to obtain data. Space positioning radar plotter, theodolite camera, Askania cameras and chase plane photo coverage were also utilized.

Spin specification demonstration flights were accomplished with the aircraft loaded in accordance with paragraphs D-2a(1)(a), D-2a(1)(b), and D-2a(1)(c) of Reference (b). These loadings simulate most aft cg condition; full armament load and low fuel (heavy fuselage, light wing); and high fuel remaining but armament expended (heavy wing, light fuselage) respectively.

DISCUSSION

Checkout of Spin Recovery Equipment

Several operations were conducted prior to the start of the spin test program for a functional operation check and evaluation of the spin recovery parachute. Deployment of the parachute was accomplished both on the ground and in flight.

Both primary and secondary deploy systems were used, on separate taxi runs, to check their operation. The signal to deploy was initiated as the aircraft approached an indicated airspeed of 120 knots. Ejection of the pilot chute appeared normal on both tests. The spin recovery parachute canopy was observed to blossom with no difficulty. Approximately five seconds later the signal to jettison was initiated. The systems operated to the satisfaction of the pilot and the engineers observing the test.

Two inflight deployments of the spin recovery parachute were also accomplished. Stabilizing the aircraft at approximately 110 knots indicated airspeed; at an indicated altitude of about 35,000 feet, the deploy signal was initiated. The pilot then pushed forward on the stick to gain airspeed. At an indicated airspeed of 150 knots, the chute was jettisoned. Testing was accomplished using first the secondary system. This was repeated using the normal system for the operation. Typical data are shown in Figure 2.

The operation of both the deploy and jettison systems were satisfactory. The results were much the same as obtained during the taxi runs.

Stall Testing

Testing to demonstrate the stall characteristics of the TF-102A was conducted concurrently with the flights for the functional checkout of the spin recovery parachute and the spin tests. Typical data are shown on Figures 3, 4 & 5.

Both power on and power off stalls were demonstrated. In each case the airspeed was gradually reduced as up elevator was introduced. As the indicated airspeed approached 100 knots, the pilot permitted the elevators to neutralize, breaking the stall. The rate of descent, computed from the indicated altitude, while the pilot was holding up elevator, was as much as 10,000 feet per minute.

DISCUSSION (Cont'd)

Stall Testing (Cont'd)

The pilot reported conventional stall characteristics. Slight buffeting occurred at 175 knots indicated airspeed. This increased in its intensity with decreasing airspeed. Slight directional oscillations associated with slight longitudinal and roll oscillations were also present. Rudder effectiveness was found to be very low at speeds approaching 100 knots indicated airspeed.

Spin Entry

The aircraft was entered into one incipient spin before beginning the specification compliance demonstration. A summary of the data obtained are shown on Figure 6.

The maneuver was entered from a wings level stall at approximately 38,000 feet indicated altitude. At an indicated airspeed of approximately 100 knots, while holding up elevator, right rudder was introduced. Left aileron (aileron against) was initiated about two (2) seconds later. The aircraft began a slow, steep spin to the right. As it completed approximately 1-1/4 turns, the controls were released, permitting the surfaces to neutralize. Recovery was complete within an additional 1/4 turn.

Spins In Heavy Fuselage Light Wing Condition

Loaded to simulate full armament and light fuel, the aircraft was flying with a forward cg. (25.3% mac). Six (6) spins were accomplished for specification compliance demonstration with this configuration.

(a) "Aileron against" for spin entry

Figure 7 summarizes the data obtained during the first specification demonstration flight. Spin entry was accomplished at an altitude of approximately 40,000 feet and an indicated airspeed of about 100 knots. Holding up elevator, right rudder, followed by left aileron (aileron against for a right spin) were introduced. The aircraft began a slow, steep spin to the right. After completing about 1-1/2 turns, the aircraft pitched up into a higher angle of attack and appeared to be spinning at a faster rate. The aircraft remained in this flatter spinning mode for the remaining three turns. During these turns some oscillations in angle of attack

DISCUSSION (Cont'd)

Spins In Heavy Fuselage Light Wing Condition (Cont'd)

(a) "Aileron against" for spin entry (Cont'd)

were recorded, (between 44° and 64° nose up). The pilot released the controls after the aircraft had completed a total of 4-1/4 turns. Recovery from the spin was complete in less than an additional 1/2 turn.

A spin to the left, also was accomplished from an "aileron against" entry i.e. left rudder and right aileron. Spinning characteristics and aircraft response were much the same as encountered on the spin to the right. When the pilot released the controls to recover, the aircraft continued turning. An aileron pulse in the direction of the spin did effect recovery within 1/2 turn after it was initiated. An analysis of the data (Figure 8) showed effective aileron remained though the controls had been released. Approximately 3° of right aileron had inadvertantly been trimmed in during the spin.

On a subsequent flight, the spin to the left was repeated. Data are shown on Figure 9. Recovery this time was completed within 3/4 turn after the control surfaces returned to their trim position.

As was noted during testing of the F-102A, spinning to the left is no more critical than spinning to the right. Pilot comment and data obtained indicate the left spin as less severe than the spin to the right. With this information, all subsequent tests for specification compliance demonstration would be spins to the right (Reference b).

(b)"Aileron with" for spin entry

Two spins were made in this configuration entered using "aileron with" (right rudder and right aileron). In each case the aircraft began a slow steep spin to the right, with an indicated angle of attack of approximately 30° nose up. (Figure 10) On the first test, this attitude continued throughout the spin. Prospin controls were maintained for 3-3/4 turns and recovery completed in less than 1/2 turn after the surfaces returned to their trim position.

On the second test, the aircraft attitude and spinning characteristics were much the same at the start. As approximately 3 turns to the right were completed, the

DISCUSSION (Cont'd)

Spins In Heavy Fuselage Light Wing Condition (Cont'd)

(b) "Aileron with" for spin entry (Cont'd)

direction of the spin reversed. With an increasing yaw rate to the left (the new direction of spinning) the angle of attack increased to about 60° nose up. Data are shown on Figure 11. The pilot released the controls after about 1-1/2 turns to the left and the aircraft recovered from the spin within an additional 1-1/4 turns.

Data obtained shows the rudder will cause the aircraft to enter a spin from a wings level stall, and also determine the initial direction. Where up elevator is held, rudder alone is sufficient to hold the aircraft in a spin and determine the direction for a spin with a low angle of attack (approximately 30° nose up or less). When the spin tends to flatten (angle of attack increases) the rudder loses its effectiveness and the aircraft will develop a spin opposite to the aileron input i.e. right aileron for a left spin. Data also indicate a faster spin at the higher angle of attack.

(c) Rudder Only (aileron neutral) for spin entry

Entry characteristics with fuselage loading were the same when the aileron was held neutral as was encountered on previous tests. The aircraft began a slow steep spin to the right. As the yaw rate increased, angle of attack built up, momentarily, dropping again to approximately 25° nose up with a decreasing yaw rate. (Figure 12) This cycle repeated approximately 1-1/2 turns later.

The aircraft made approximately 4-3/4 turns to the right, recovering in less than an additional 1/3 turn when the elevators and rudder were released. The aircraft had apparently been maintained in the spin by the rudder at the lower angles of attack. When the aircraft pitched up into a flatter spin, the rudder was ineffective. This higher angle of attack was apparently not of sufficient duration to allow the neutral aileron to effect recovery and as the spin steepened, the rudder once more became effective.

DISCUSSION (Cont'd)

Spins in the Heavy Wing, Light Fuselage Condition

With a heavy wing light fuselage configuration, the aircraft is loaded to simulate armament expended with a heavy fuel load remaining. The cg at spin entry, in this configuration, was approximately 26.3% mac, and 3 spin tests were made. Data are shown on Figures 13, 14 & 15,

In each case, the nose pitched up and the spin developed in a flatter mode; (nominal 60° nose up). This was much the same as encountered in the more forward cg where "aileron against" was initiated as entry control. Roll rates and pitch rates in the developed spin were also very similar. With the high angle of attack, the spin developed in the direction opposite to the aileron. For example, on flight 12, right rudder and right aileron were introduced while holding up elevator. The aircraft began a slow steep spin to the right. As the yaw rate began to decrease, the angle of attack increased at a slow rate. The aircraft then began to spin to the left accompanied by an increasing yaw rate. The aircraft pitched up to an angle of attack of approximately 60° nose up, remaining essentially at this attitude until the controls were released. (Figure 15) As was observed on previous flight, a spin to the left appears to be less severe than a spin to the right.

Recovery was accomplished in every case when the pilot released the controls, permitting the surfaces to return to their trim position. Within 1 turn after such action the aircraft had stopped turning.

Spins in the Light Wing, Light Fuselage Condition.

As a mission has been completed, a tactical aircraft returns to its base with light fuselage and light wing. Its armament has been expended and fuel is low. In such a condition the cg is at the most aft position since takeoff and climb out. For these tests, the cg at spin entry was approximately 27.3% mac.

Three spins were entered from a wings level stall (Figures 16, 17 & 18) and one from a 2g turn (Figure 19) with the aircraft in the clean configuration, unloaded to give the most aft cg obtainable at the entry altitude.

DISCUSSION (Cont'd)

Spins in the Light Wing, Light Fuselage Condition. (Cont'd)

Spin and spin recovery characteristics in this configuration were similar to the spins with heavy wing loading. Recovery was completed in 1-1/4 turns or less after the controls were neutralized.

Spins in the Landing Configuration

For spin testing in the landing configuration, the aircraft was loaded to simulate armament expended and fuel low (light fuselage and light wing). With such a loading, the cg, at spin entry, was approximately 27.3% mac, the most aft position.

One spin was conducted for specification compliance demonstration. Entry was accomplished when right rudder and left aileron were introduced while holding up elevator. Pro-spin controls were held for about one turn, then released, permitting the surfaces to neutralize. (Figure 20) Approximately two additional turns were made before recovery was complete.

Two additional spins were made in the landing configuration to verify recovery characteristics and procedures obtained on the previous flight. Both were spins to the right, entered, "aileron against" (right rudder, left aileron). In the first of these, pro-spin controls were held for approximately two turns. (Figure 21) For recovery, the elevators and rudder were neutralized and an "aileron with" (right aileron) pulse initiated. The aircraft stopped turning in less than an additional turn. On the second test, the pro-spin controls were held but one turn, (Figure 22) then neutralized. In this case, recovery was completed in less than one turn after the pilot released the controls, "aileron with" was not necessary. Characteristics of spins in the landing configuration were much the same as those in the clean configuration loaded for the most aft cg.

Effectiveness of Spin Recovery Parachute

After completing all the testing required for the demonstration of specification compliance, one additional test was made to determine the effectiveness of the spin recovery parachute. Spin entry was accomplished at an indicated altitude of about 45,000 feet and approximately 90 knots. Holding up elevator, right rudder and left aileron were initiated.

DISCUSSION (Cont'd)

Effectiveness of Spin Recovery Parachute (Cont'd)

After spinning right about 3 turns, the pilot deployed his spin recovery parachute while continuing to hold the pro-spin control. The effect of the chute was sufficient to reduce the angle of attack and decrease the yaw rate. Recovery was completed in less than one additional turn; and the ailerons were neutralized as the parachute was jettisoned. Data from this test are shown on Figure 23.

Accelerations Encountered During Spins

Accelerometers to measure normal, lateral and longitudinal accelerations were installed at the aircraft cg and in the nose wheel well. Figures 24 & 25 present data obtained during the most severe conditions encountered. As shown, normal accelerations of up to 3.25 g's were encountered, momentarily. Lateral and longitudinal accelerations on these flights were very small. Accelerations encountered, in general, during the spin test program, were insignificant and are therefore not a factor in spin or spin recovery characteristics of the TF-102A aircraft.

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CONCLUSIONS

1. The TF-102A can be entered into a spin maneuver from a stall attitude when pro-spin controls are initiated.

2. Rudder control is sufficient to start the aircraft spinning from a stall attitude.

3. At an indicated angle of attack of 35° nose up or less the rudder determines the direction in which the aircraft will spin.

4. Where the angle of attack is greater than 35° nose up, the spin will develop opposite to the aileron input.

5. Type of entry has no effect on any of the characteristics of the developed spin.

6. The extension of the landing gear does not effect the entry, recovery or spinning characteristics of the aircraft.

7. As on the F-102A, to recover from a spin, the controls should be neutralized. If recovery is not completed within 1 to 1-1/2 turns, "aileron with" should be initiated.

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REFERENCES

(a) USAF Specification 1815B. Flying Qualities of Piloted Airplanes.
Dated 1 June 1948.

(b) USAF Specification 1816. Spinning Requirements for Airplanes.
Dated 15 June 1945.

(c) Convair Report ZC-8-259. Stability and Control, Spin, Development and Demonstration Testing TF-102A USAF S/N 54-1362.
Dated 10 October 1956. (Flight Test Program)

(d) Convair Memorandum EFT-A-8-348. Ground Deployment Test of Spin Recovery Parachute.
Dated 16 January 1957.

(e) Convair Memorandum EFT-A-8-403. In-flight Evaluation of Spin Recovery Equipment.
Dated 18 April 1957.

(f) Convair Memorandum EFT-A-8-423. Evaluation of Flight Data Obtained During Preliminary Spin Maneuvers.
Dated 27 May 1957.

(g) Convair Memorandum EFT-A-8-430. Flight Test Results of F-102A Stall and Spin Tests.
Dated 5 June 1957.

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Flight Number and Date	Loading and Config.	Para. of USAF Spec 1816 ①	Entry	Controls for Spin & Duration	Total Turns (including recovery) & Direction	Rate of Turning During Spin	Maxi
							Roll
5 24 May 1957	Heavy Fuselage Light Wing cg at 25.3% mac Clean	Not Applicable	From wings level stall 37,800 feet indicated altitude 104 knots indicated	17½° up elev. reduced to 13° 16° rt. rudder 5° lft. aileron Held 1¼ turns	1-1/3 rt. then 1/3 lft.	5.5 rpm	← Dat
5 24 May 1957	Heavy Fuselage Light Wing cg at 25.3% mac Clean	D-2a (1)(b) & D-2a (2)(c)	From wings level stall 39,400 feet indicated altitude 103 knots indicated	15° up elev. reduced to 12° 17° rt. rudder 5° lft. aileron Held 4¼ turns	4½ right	8.8 rpm	← Da
6 29 May 1957	Heavy Fuselage Light Wing cg at 25.3 % mac Clean	D-2a (1)(b) & D-2a (2)(c)	From wings level stall 35,000 feet indicated altitude 100 knots indicated	15° up elev. reduced to 12½° 17° lft. rudder 5½° rt. aileron Held 3-3/4 turns	6-1/3 left	Approx. 8.3 rpm	← Da
8 10 June 1957	Heavy Fuselage Light Wing cg at 25.3% mac Clean	D-2a (1)(b) & D-2a (2)(b)	From wings level stall 38,000 feet indicated altitude 105 knots indicated	16° up elev. reduced to 13° Full rt. rudder 7° rt. aileron Held 3-3/4 turns	4½ right	5 rpm increasing to 7.5 rpm	145°/ right
8 10 June 1957	Heavy Fuselage Light Wing cg at 25.3% mac Clean	D-2a (1)(b) & D-2a (2)(a)	From wings level stall 37,000 feet indicated altitude 103 knots indicated	19° up elev. 20° rt. rudder Neutral aileron Held 4-2/3 turns	5-1/3 right	7 rpm	90°/ righ
9 11 June 1957	Heavy Fuselage Light Wing cg at 25.3% mac Clean	D-2a (1)(b) & D-2a (2)(e)	From wings level stall 34,500 feet indicated altitude 102 knots indicated	19° up elev. reduced to 13° 18° lft rudder 7° rt. aileron Held 4 turns	4-3/4 left	3 rpm increasing to 4 rpm	45° lef

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Figure: *1a* Page 15
SUMMARY OF SPIN TESTS
TF102A A/C - S/N 54-1362
Sheet 1 of 3

um Rates During Spin			Maximum Angle of Attack ②	Recovery Controls & Turns Required	Altitude Lost in Spin (feet)③	REMARKS
	Pitch	Yaw				
not obtained on this flight →				Neutral Less than ¼ turn	4300 3500	This flight was to further investigate spin entry and to explore recovery characteristics from an incipient spin. Figure 6
not obtained on this flight →				Neutral Less than ¼ turn	10,100 900	Figure 7
not obtained on this flight →				Neutral 2 turns then aileron with 2/3 turn	8800 5600	Approximately 3° of right aileron remained after the pilot released the controls. This had inadvertantly been "trimmed in" during the spin. Figure 8
o	30°/sec up	40°/sec right	35° Nose Up	Neutral 2/3 turn	13,000 4,000	The rudder remained effective during the spin when the angle of attack remained at a nominal 20° to 30° nose up during the maneuver. Figure 10
	20°/sec up	70°/sec right	62° Nose Up	Neutral ½ turn	13,500 5,500	When the angle of attack remained at approximately 25° nose up, the rudder alone was sufficient to keep the airplane spinning as long as up elevator was maintained. Figure 12
e	5°/sec up	15°/sec left	25° nose up	Neutral 3/4 turn	15,800 2,400	Repeat of flight 6 (where aileron inadvertantly trimmed in by the pilot held the aircraft in a spinning attitude after the controls had been neutralized) Recovery in this instance was complete in less than one turn after controls were neutralized. Figure 9

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Flight Number and Date	Loading and Config.	Para. of USAF Spec No.6 ①	Entry	Controls for Spin & Duration	Total Turns (including recovery) & Direction	Rate of Turning During Spin	
9 11 June 1957	Heavy Fuselage Light Wing cg at 25.3% mac Clean	D-2a (1)(b) & D-2a (2)(b)	From wings level stall 38,000 feet indicated altitude 100 knots indicated	18° up elev. Full rt. rudder 6° rt. aileron Held 2-2/3 right & 1-1/4 left	2-2/3 right then 2-3/4 left	8 rpm (right) 5.5 rpm (left)	8 ri 11 le
10 12 June 1957	Heavy Wing Light Fuselage cg at 26.3% mac Clean	D-2a (1)(c) & D-2a (2)(c)	From wings level stall 38,500 feet indicated altitude 98 knots indicated	16° up elev. reduced to 10° 13° rt. rudder 4° lft. aileron Held 3½ turns	4½ right	8 rpm	9 r
10 12 June	Light Wing Light Fuselage cg at 27.3% mac Clean	D-2a (1)(a) & D-2a (2)(c)	From wings level stall 41,000 feet indicated altitude 96 knots indicated	16° up elev. reduced to 11° 18° rt. rudder 5° lft. aileron Held 3½ turns	4½ right	6.2 rpm	11 ri
10 12 June 1957	Light Wing Light Fuselage cg at 27.3% mac Clean	D-2a (1)(a) & D-2a (2)(a)	From wings level stall 43,000 feet indicated altitude 94 knots indicated	17° up elev. Full rt. rudder Neutral aileron Held 3½ turns	3-3/4 right	5.9 rpm	80 ri
11 14 June 1957	Heavy Wing Light Fuselage cg at 26.3% mac Clean	D-2a (1)(c) & D-2a (2)(a)	From wings level stall 39,500 feet indicated altitude 95 knots indicated	20° up elev. Full rt. rudder Neutral aileron Held 3-3/4 turns	4-1/3 right	5 rpm	12 ri
11 14 June 1957	Light Wing Light Fuselage cg at 27.3% mac Landing Gear down	D-2c (1) & D-2a (2)(c)	From wings level stall 41,500 feet indicated altitude 96 knots indicated	17° up elev. 18° rt. rudder 5° lft. aileron Held 3/4 turn	2-3/4 right	7.5 rpm	15 ri

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Figure, 1b Page 16
SUMMARY OF SPIN TESTS
TF102A A/C - S/N 54-1362
Sheet 2 of 3

imum Rates During Spin			Maximum Angle of Attack ②	Recovery Controls & Turns Required	Altitude Lost in Spin (feet)③	REMARKS
	Pitch	Yaw				
ec sec	30°/sec up & down	45°/sec right 35°/sec left	70° Nose Up	Neutral 1½ turn	14,000 4,000	Figure 11
ec	15°/sec up	85°/sec right	65° Nose Up	Neutral 1 turn	8500 2500	Figure 13
ec	20°/sec up	95°/sec right	70° Nose Up	Neutral 1 turn	9500 4500	Figure 16
c	20°/sec up	60°/sec right	73° Nose Up	Neutral ½ turn	10,200 1,800	Figure 17
ec	20°/sec up	50°/sec right	70° Nose Up	Neutral 2/3 turn	12,000 2,500	Figure 14
ec	30°/sec up	60°/sec right	70° Nose Up	Neutral 2 turns	2000 7500	Review of photo coverage indicates pro-spin controls were not held long enough to meet specification compliance requirements. Figure 20

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Flight Number and Date	Landing and Config.	Para. of USAF Spec 1816 ①	Entry	Controls for Spin & Duration	Total Turns (including recovery) & Direction	Rate of Turning During Spin	Ma
							Rel
11 14 June 1957	Light Wing Light Fuselage cg at 27.3% mac Landing Gear down	D-2c (1) & D-2a (2)(b)	From wings level stall 37,000 feet indicated altitude 95 knots indicated	15° up elev. Full rt. rudder 4.5° rt. aileron Held for 3/4 turn right, 2-2/3 turn left	3/4 right 2-3/4 left	5.4 rpm	75°/ left
12 14 June 1957	Heavy Wing Light Fuselage cg at 26.3% mac Clean	D-2a (1)(c) & D-2a (2)(b)	From wings level stall 39,500 feet indicated altitude 96 knots indicated	20° up elev. reduced to 16° Full rt. rudder 8° rt. aileron Held 1/3 turn rt., 4½ turns lft.	1/3 right 5-1/4 left	7.5 rpm	70°/ left
12 14 June 1957	Light Wing Light Fuselage cg at 27.3% mac Clean	Not Applicable	From a 2g turn 39,000 feet indicated altitude 125 knots indicated	20° up elev. Full rt. rudder Neutral aileron Held for 3¼ turns	4½ right	6.2 rpm	100° rig
12 14 June 1957	Light Wing Light Fuselage cg at 27.3% mac Clean	D-2a (1)(a) & D-2a (2)(c)	From wings level stall 44,500 feet indicated altitude 90 knots indicated	16° up elev. 16° rt. rudder increased to full right 3° lft. aileron increased to 6°	4 right	6.2 rpm	60°/ rig
13 19 June 1957	Light Wing Light Fuselage cg at 27.3% mac Landing Gear down	D-2c (1) & D-2a (2)(c)	From wings level stall 38,000 feet indicated altitude 90 knots indicated	15° up elev. reduced to 10° 16° rt. rudder 5° lft. aileron Held 2 turns	3 right	6 rpm	115 rig
13 19 June 1957	Light Wing Light Fuselage cg at 27.3% mac Landing Gear down	D-2c (1) & D-2a (2)(c)	From wings level stall 40,500 feet indicated altitude 90 knots indicated	18° up elev. 15° rt. rudder 4° lft. aileron Held 1 turn	1-3/4 right	6.5 rpm	55° rig

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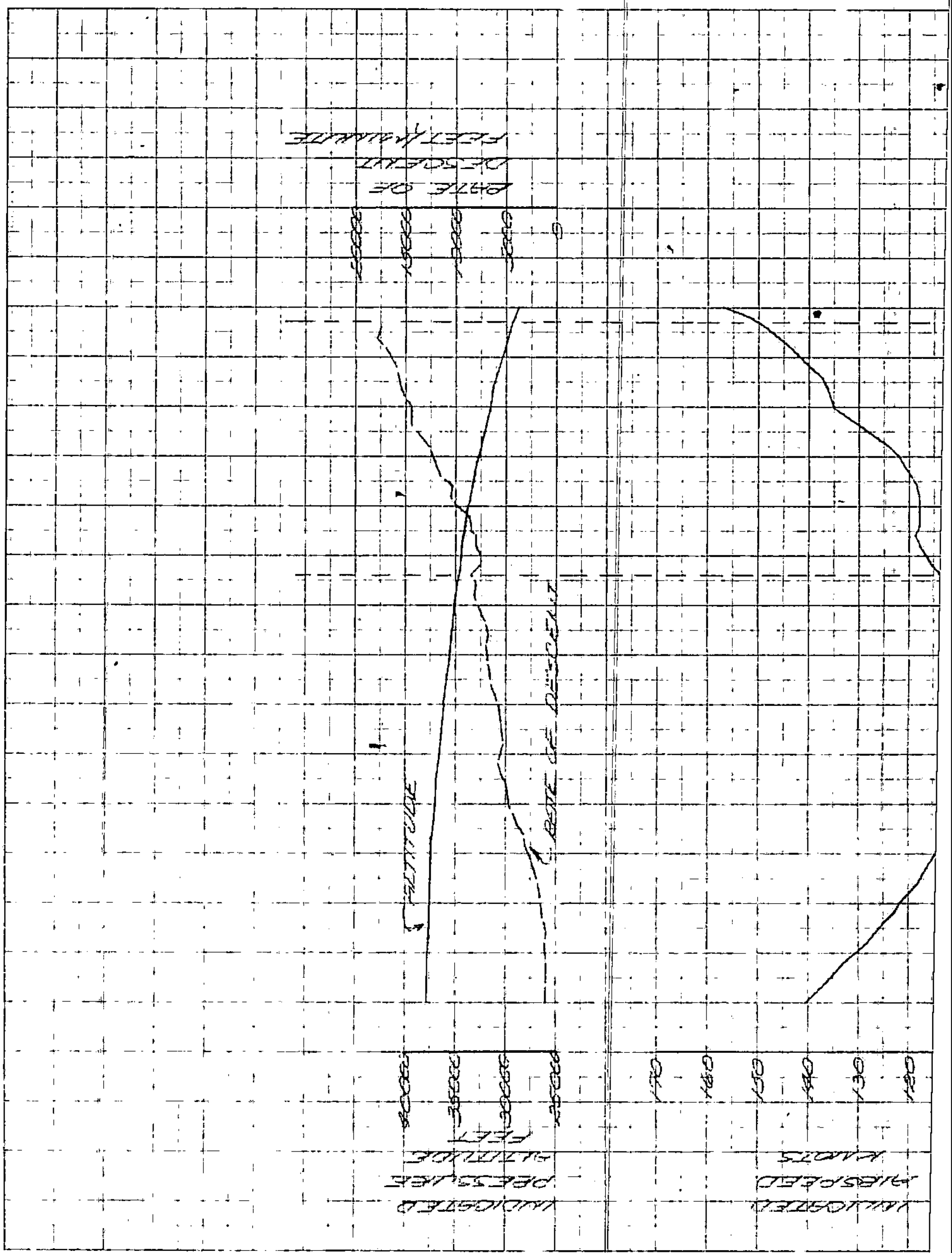
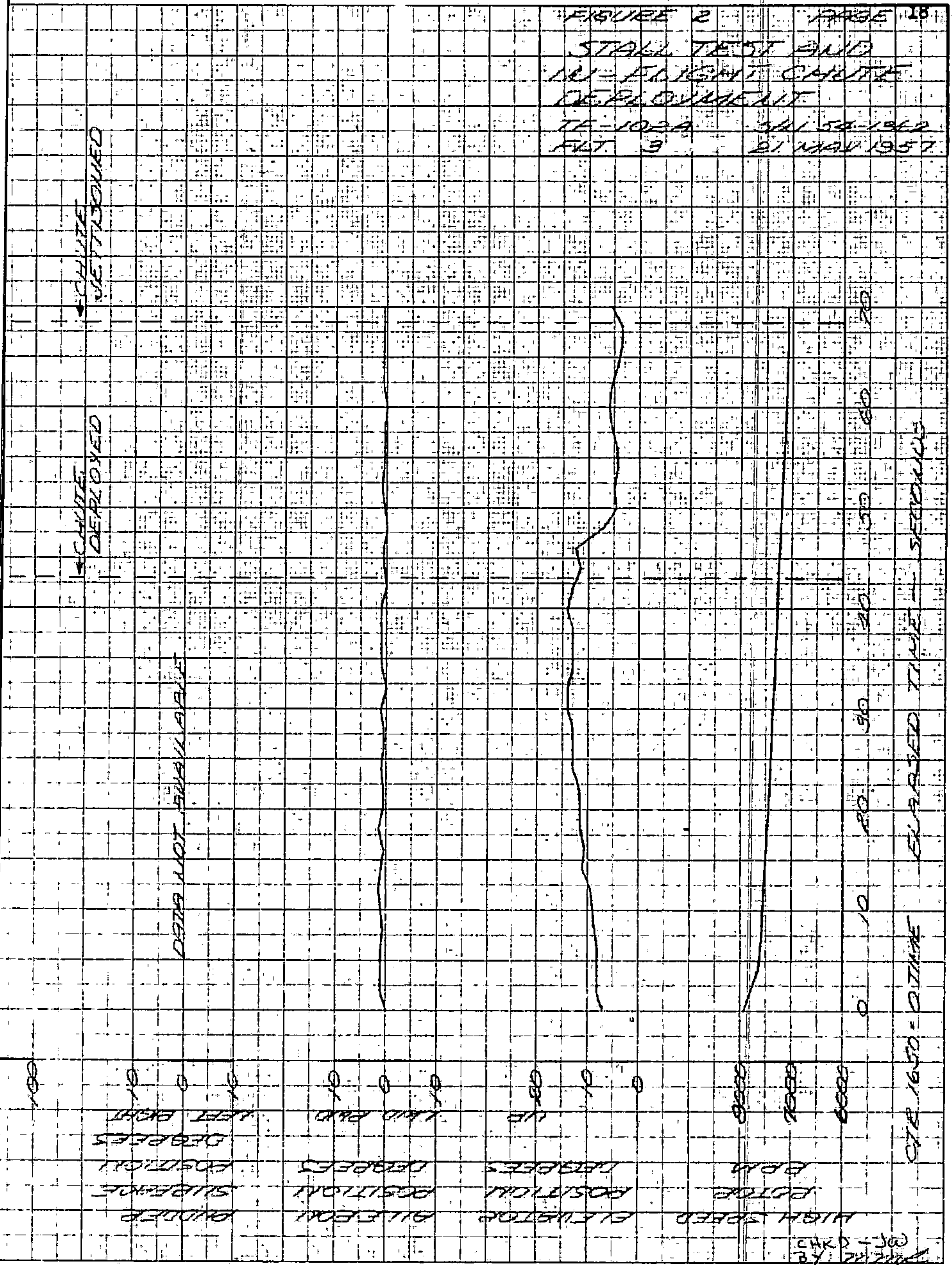


K+E 10 X 10 TO THE CM 359T-14LG
KEUFFEL & ESSER CO.

FIGURE 2 PAGE 18

STALL TEST AND
IN-FLIGHT CHUTE
DEPLOYMENT

TF-102A 56-1362
FLT. 3 21 MAY 1957



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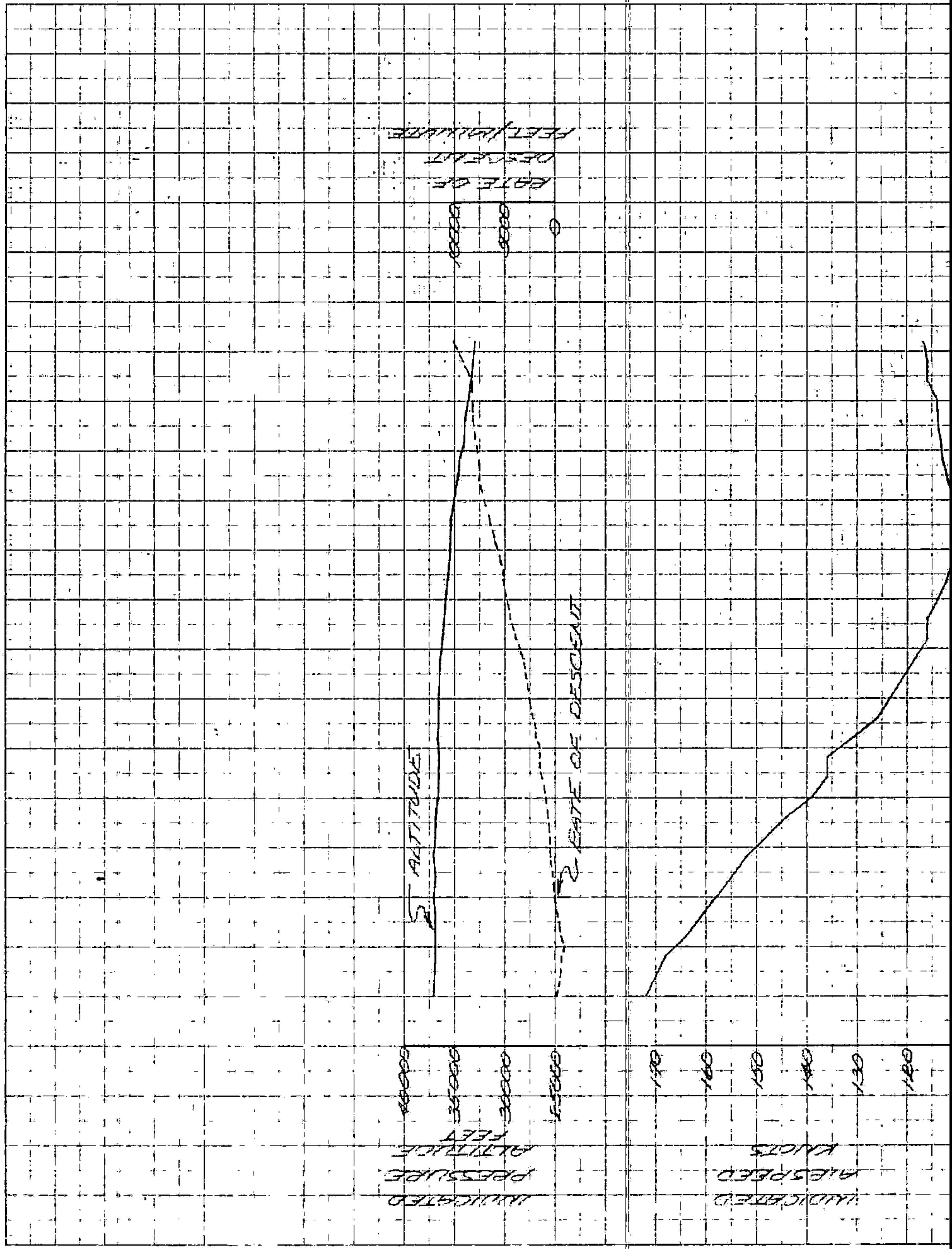


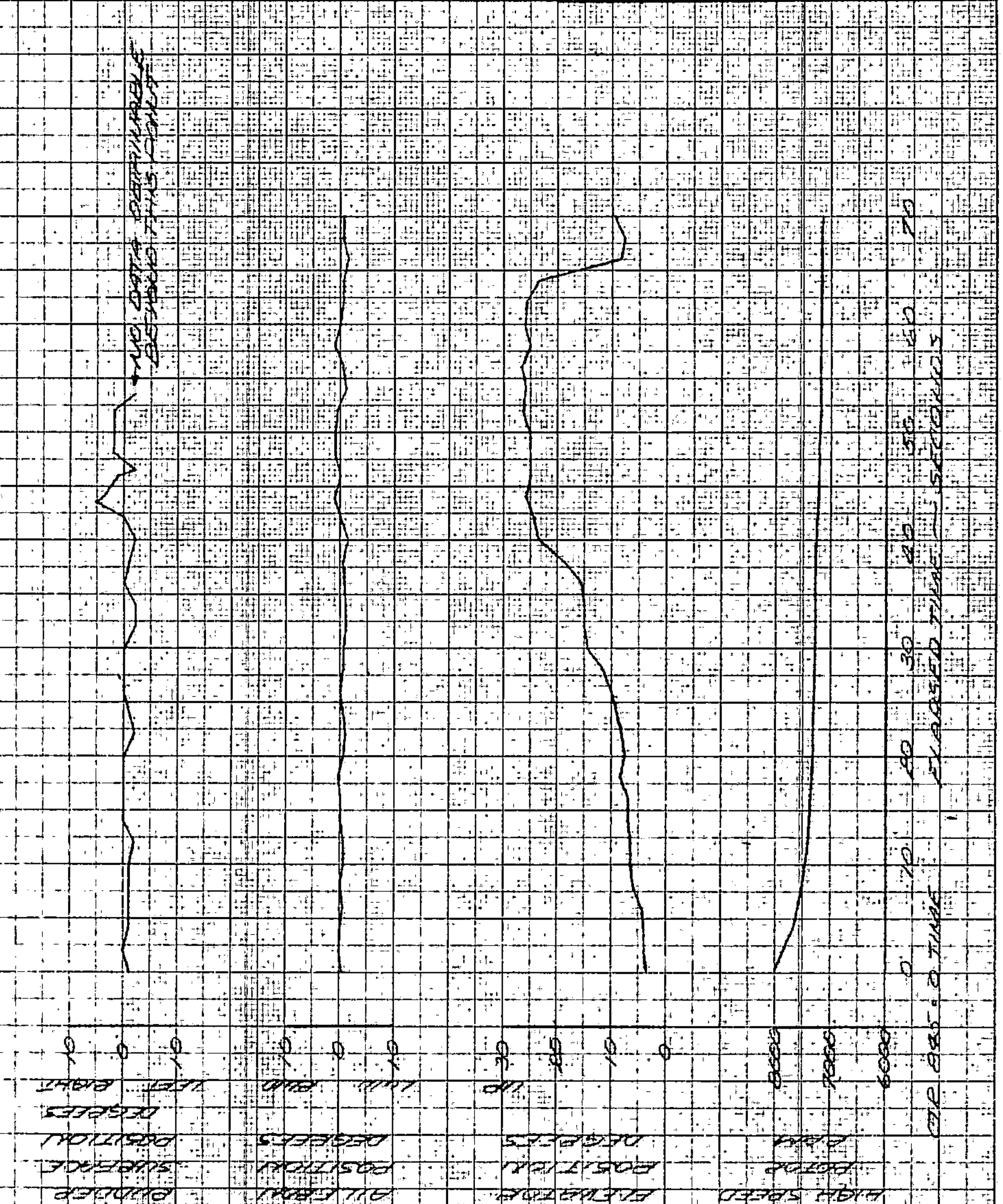
K&E 10 X 10 TO THE CM 359T-14LG KEUFFEL & ESSER CO. MADE IN U.S.A.

FIGURE 3

STALL TEST

TF-102A SN 54-1362

FLT 8 15 MAY 1957



CHKD JW
BY

EF-8-1362-47

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CONFIDENTIAL

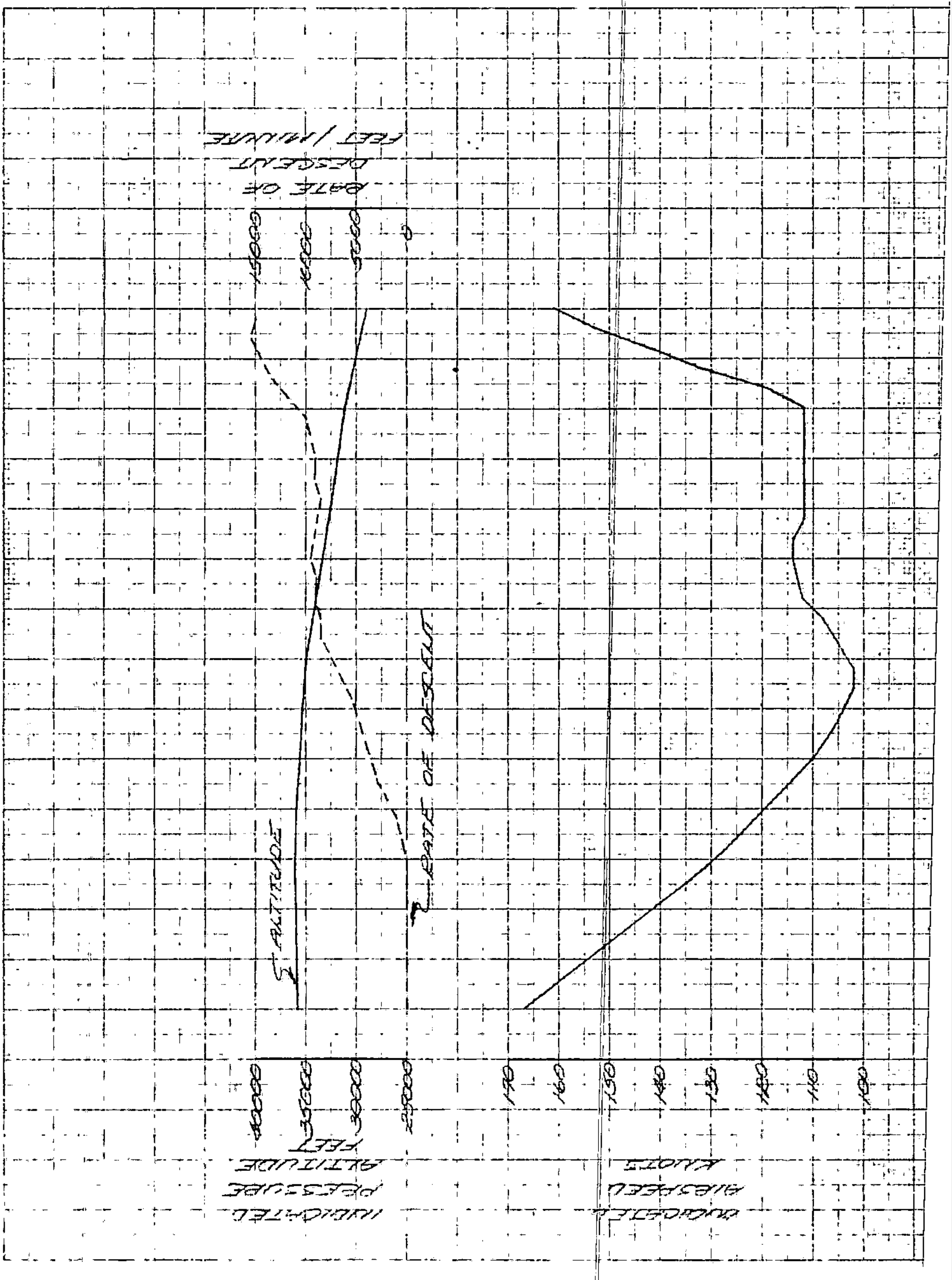
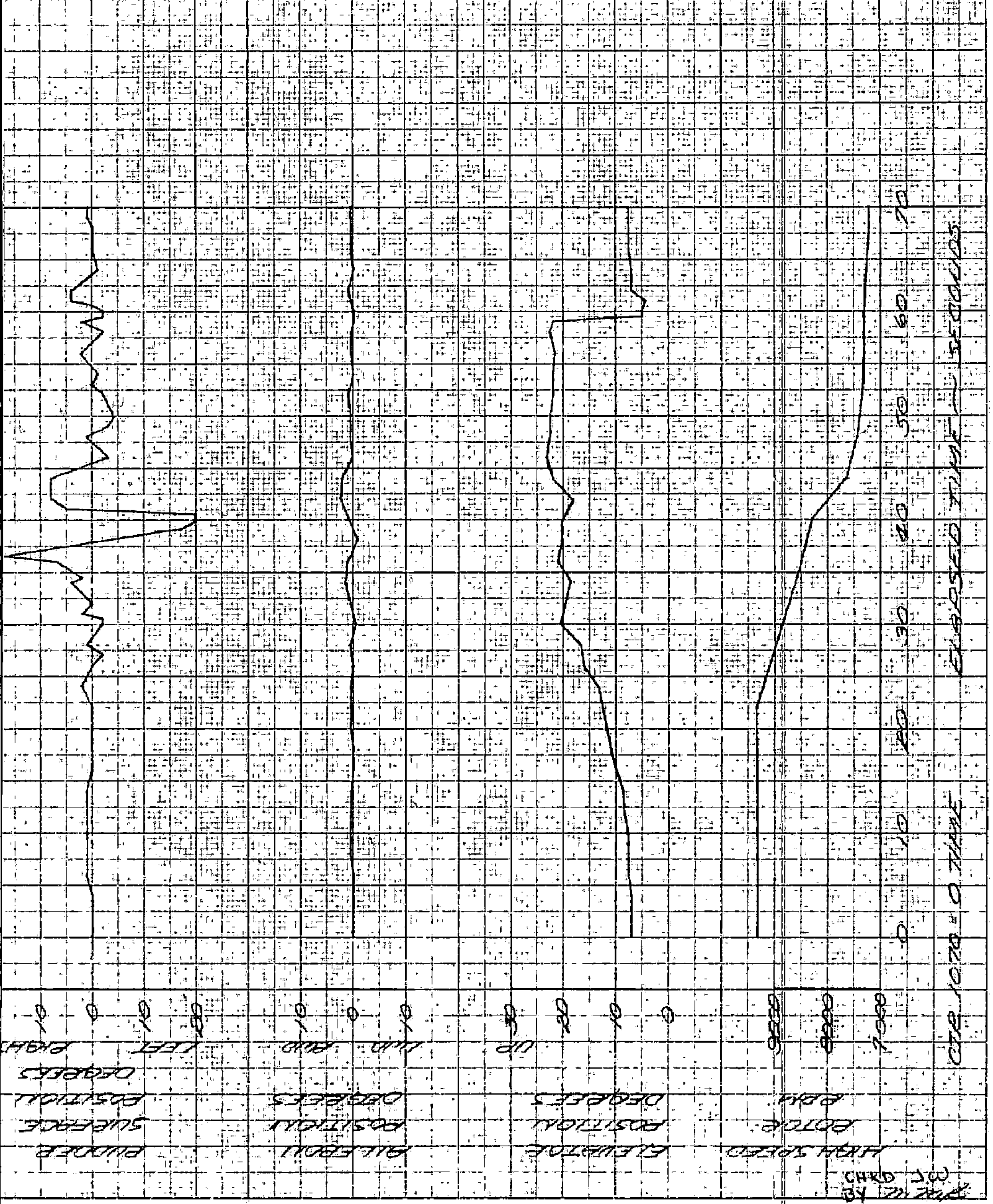


FIGURE 4 PAGE 20

STALL TEST

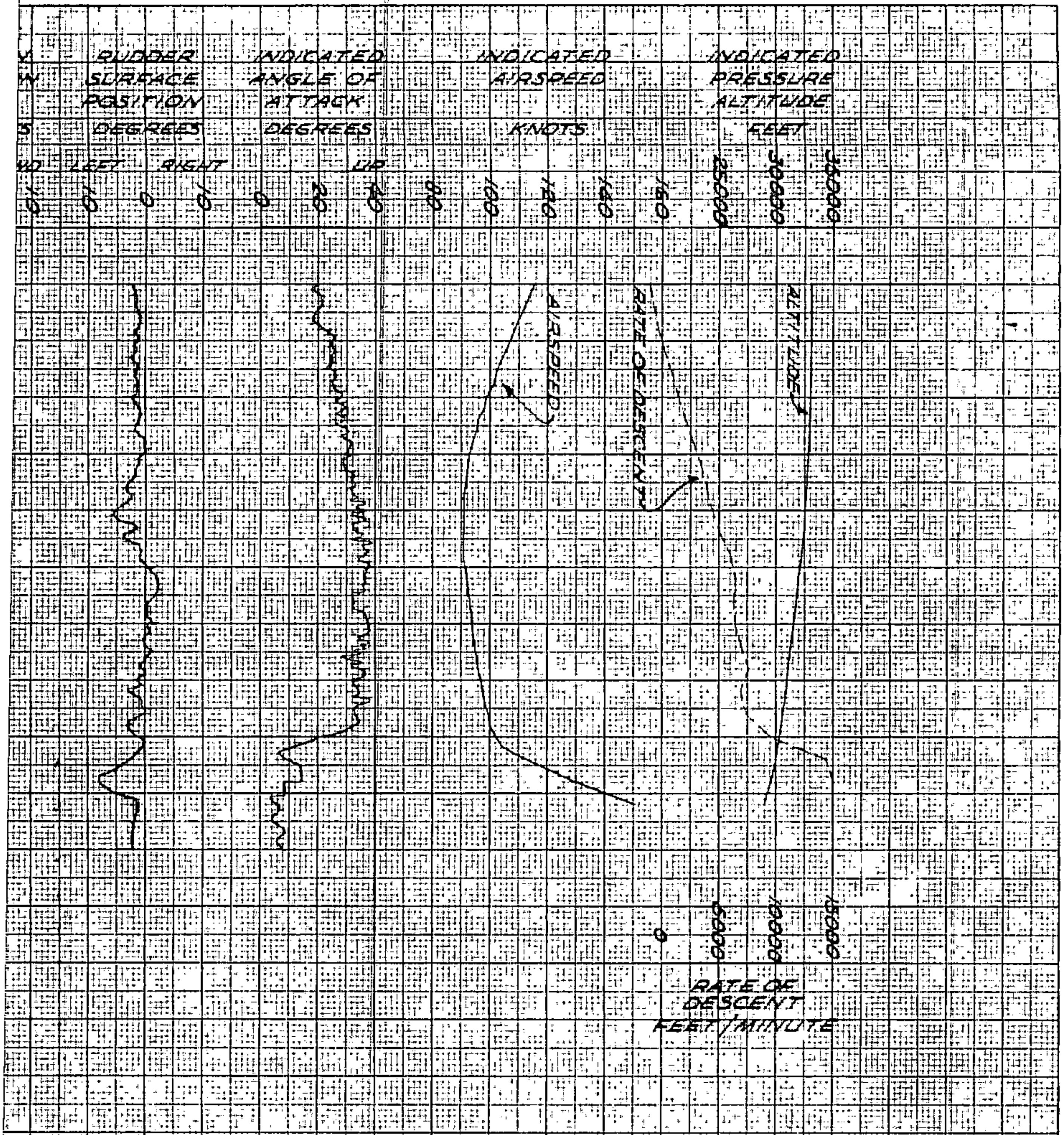
TE-1024 S/N 54-1368

FLT 3 21 MAY 1957



CHKD JW
BY

CONFIDENTIAL



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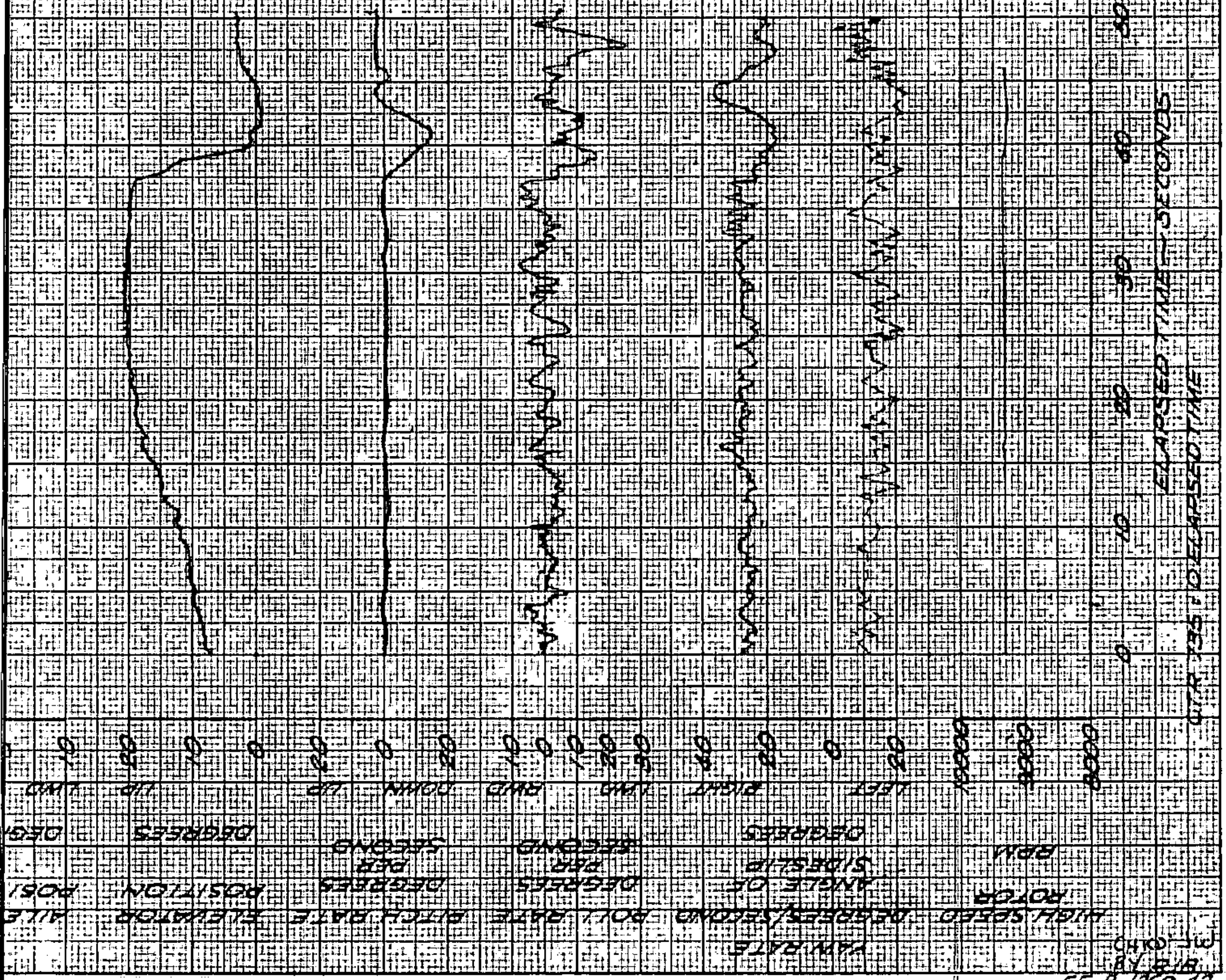
K&E 10 X 10 TO THE CM 359T-14LG
KEUFFEL & ESSER CO. ALBANENE

FIGURE 5 PAGE 21
STALL TEST DATA
TF-102A S/N 54-1362
FLT 13 19 JUNE 1957



EF-8-1362 43

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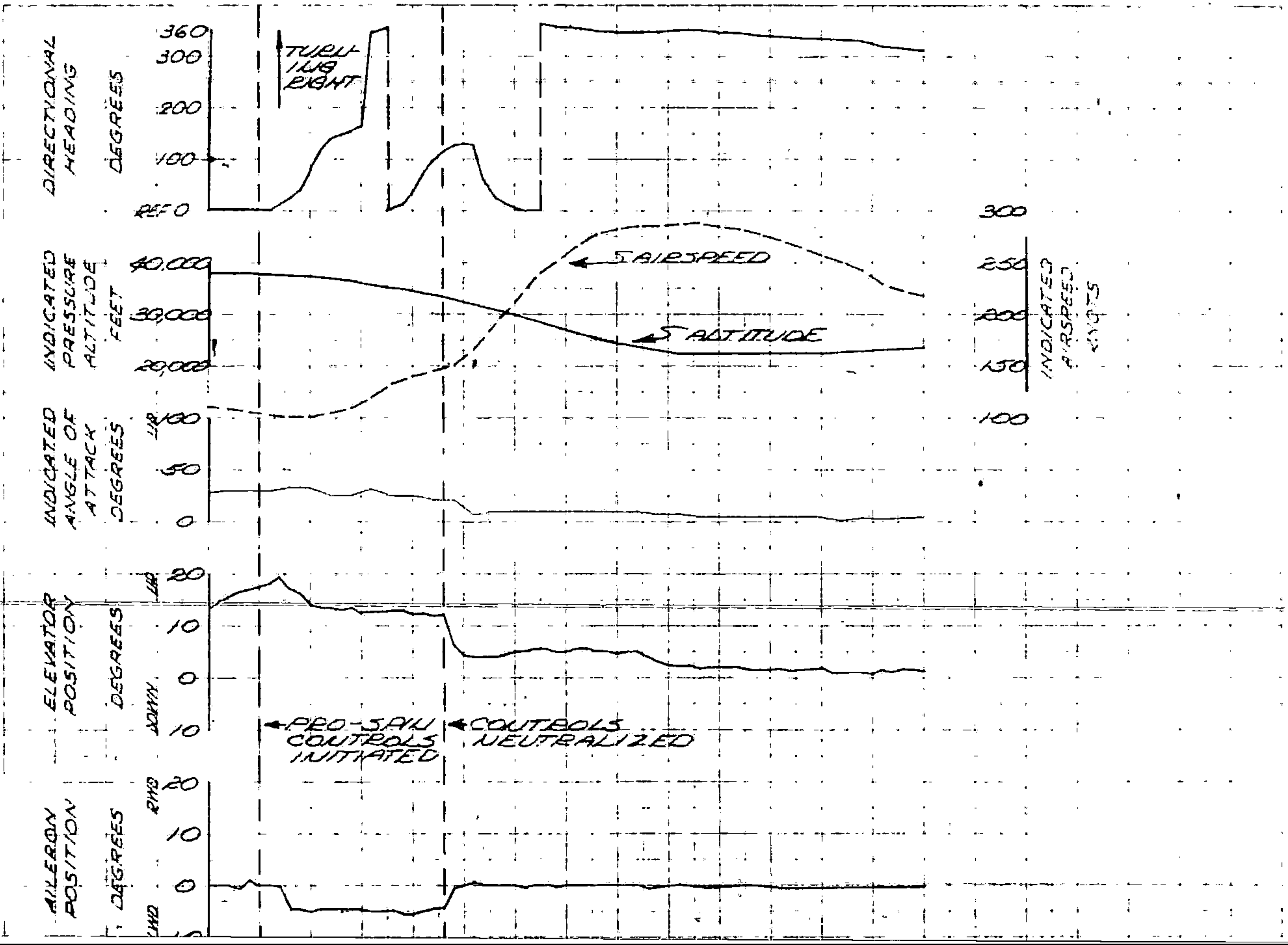
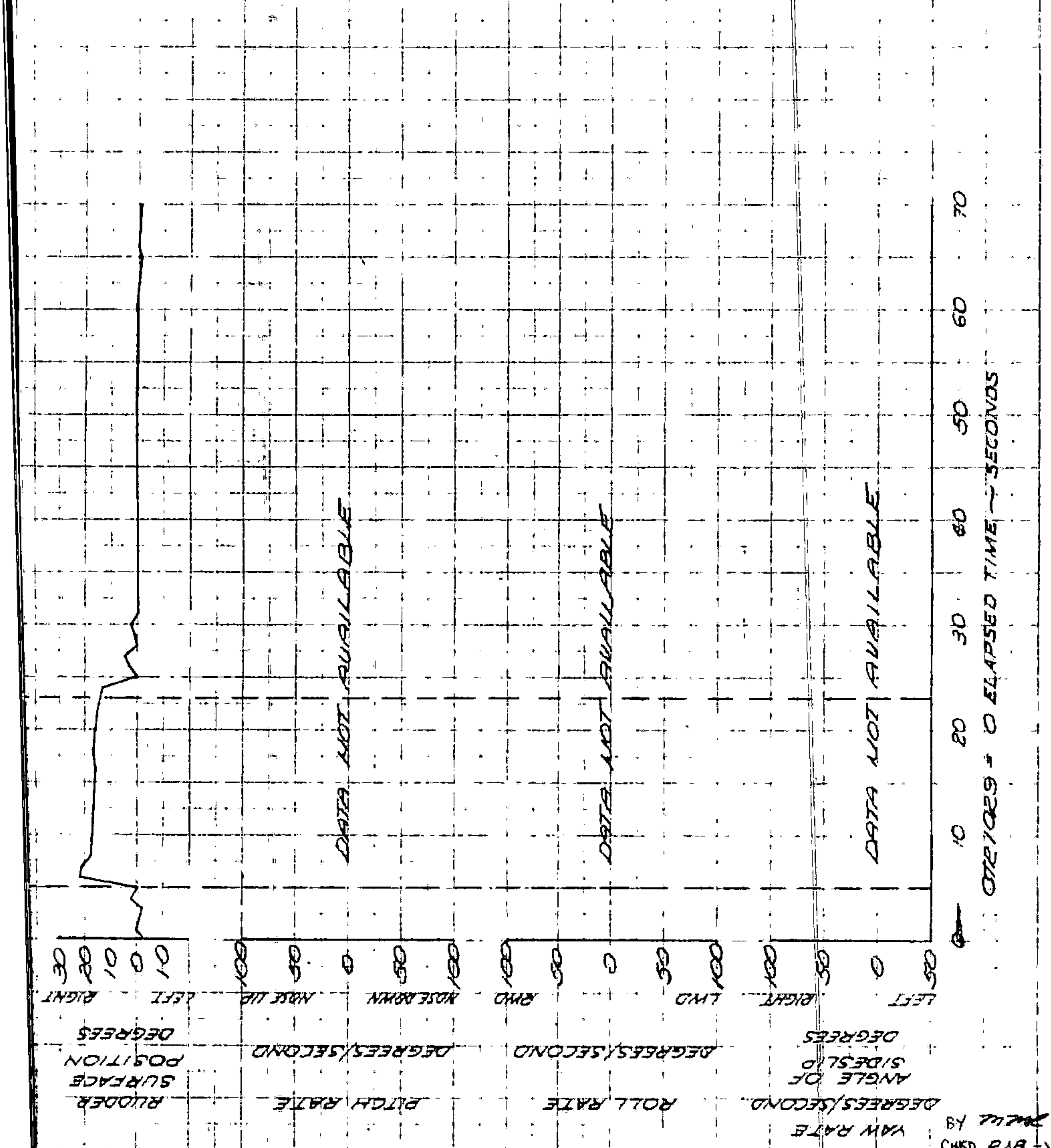


FIGURE 6 PAGE 22

SPIN TEST DATA

SIMULATED FULL ARMAMENT FUEL LOW, AILERON AGAINST FOR ENTRY (RIGHT RUDDER, LEFT AILERON)

TF-102A S/N 54-1362
FLT. 5 24 MAY 1957



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CHKD RAB
EF-8-1362-44

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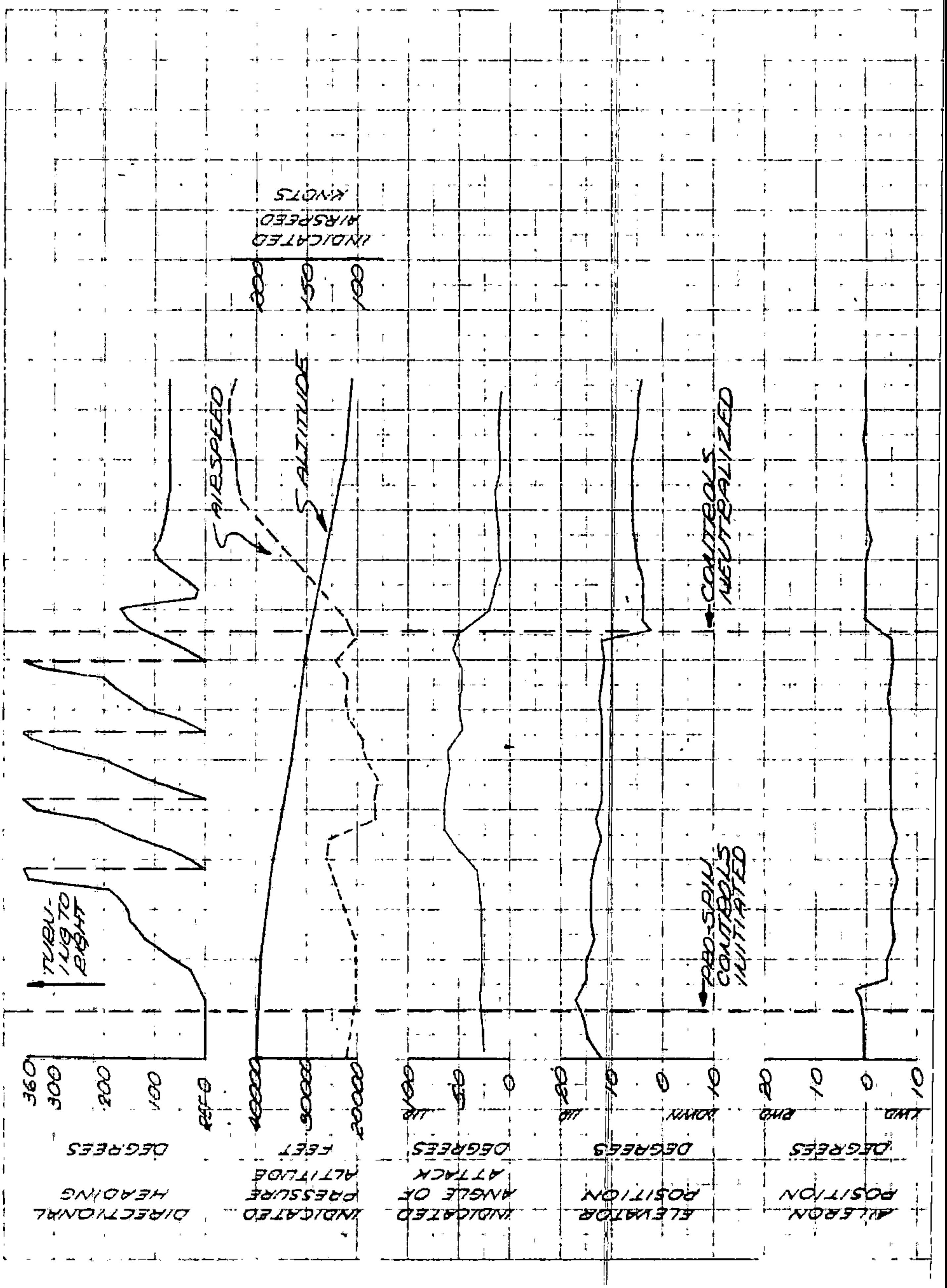
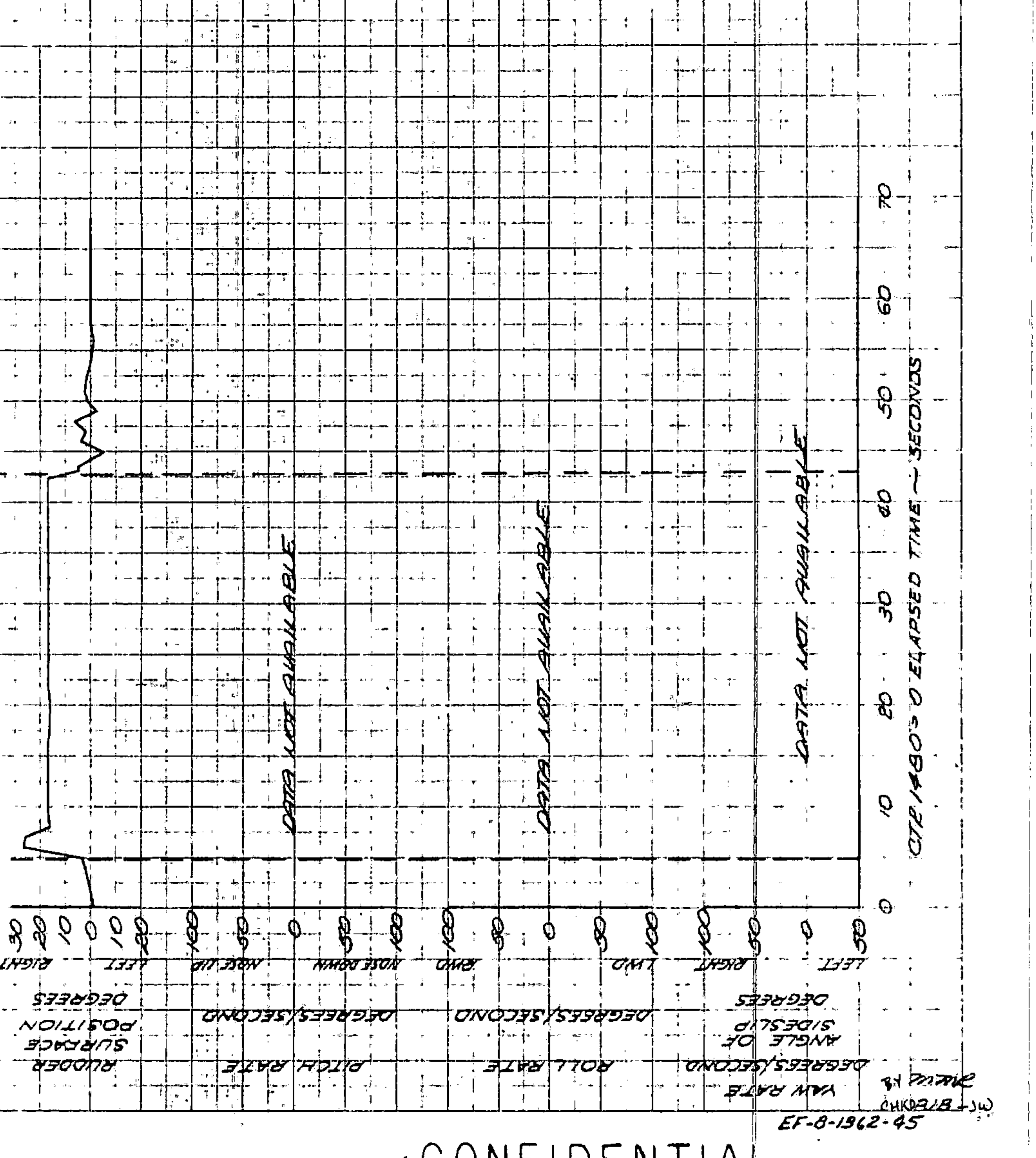


EFT-A-8-480

FIGURE 7 PAGE 23

SPIN TEST DATA

SIMULATED FULL ARMAMENT, FUEL
LOW, AILERON AGAINST FOR ENTRY
(RIGHT RUDDER, LEFT AILERON)
TF-102A S/N 57-1362
FLT. 5 24 MAY 1957



BY PMH
CHKD BB - JW

EF-8-1962-45

CONFIDENTIAL

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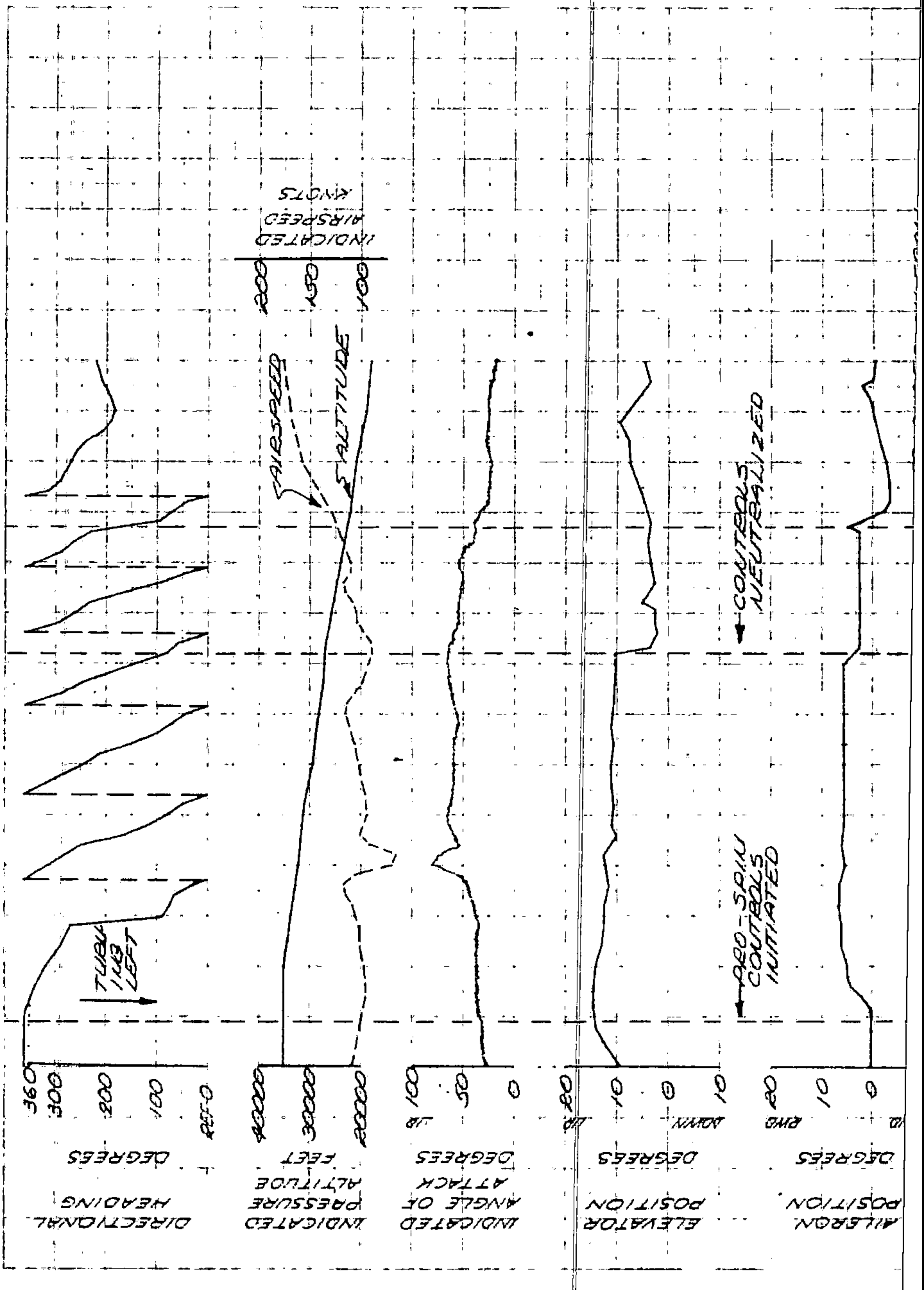


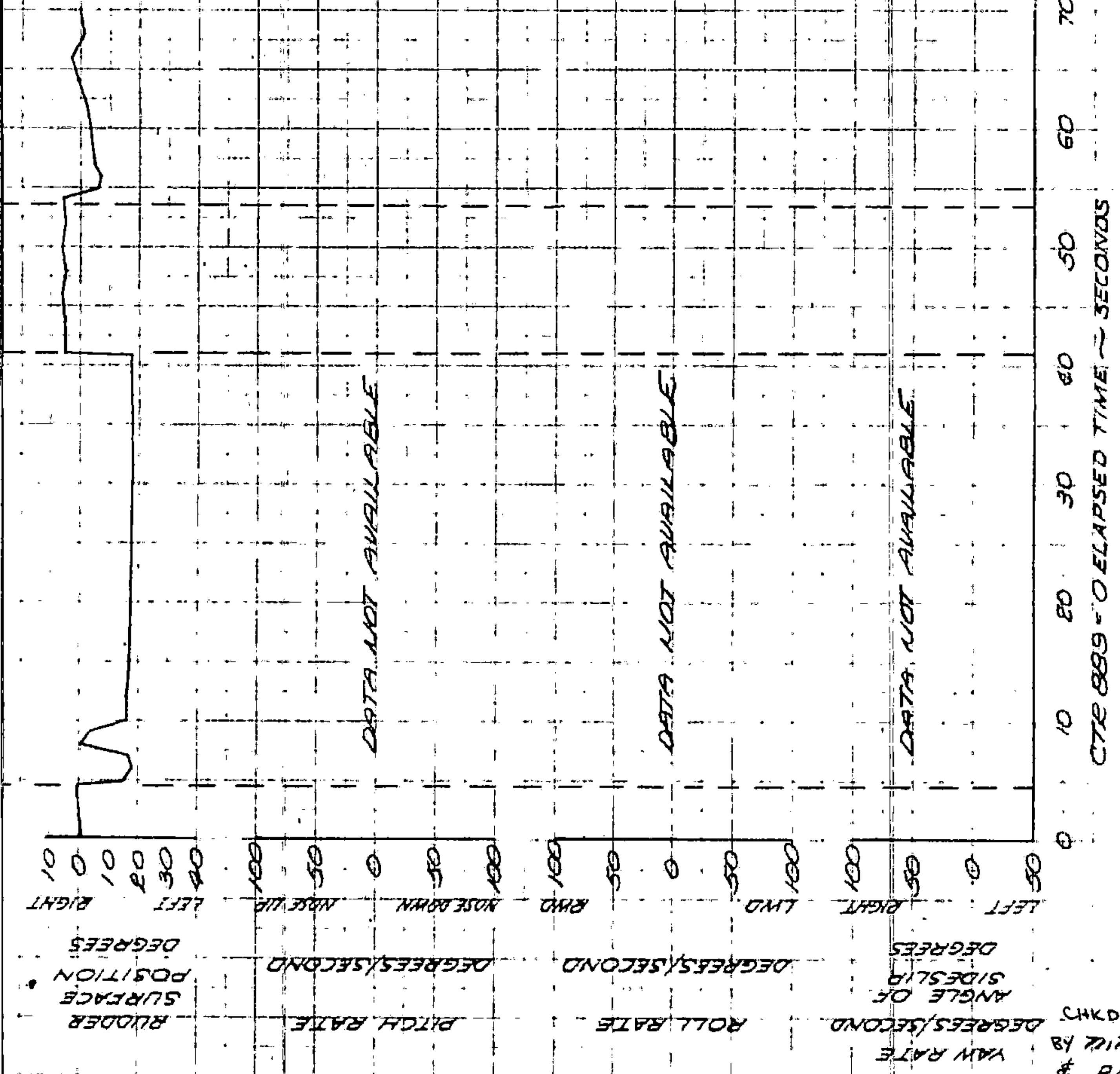
EFT-A-8-480

FIGURE 8 PAGE 24

SPIN TEST DATA

SIMULATED FULL ARMAMENT FUEL
LOW AILERON AGAINST FOR ENTRY
(LEFT RUDDER, RIGHT AILERON)

TF-102A S/N 54-1362
FLT 6 29 MAY 1957



CHKD JW
BY [illegible] & RJB
EF-8-1362-46

CONFIDENTIAL

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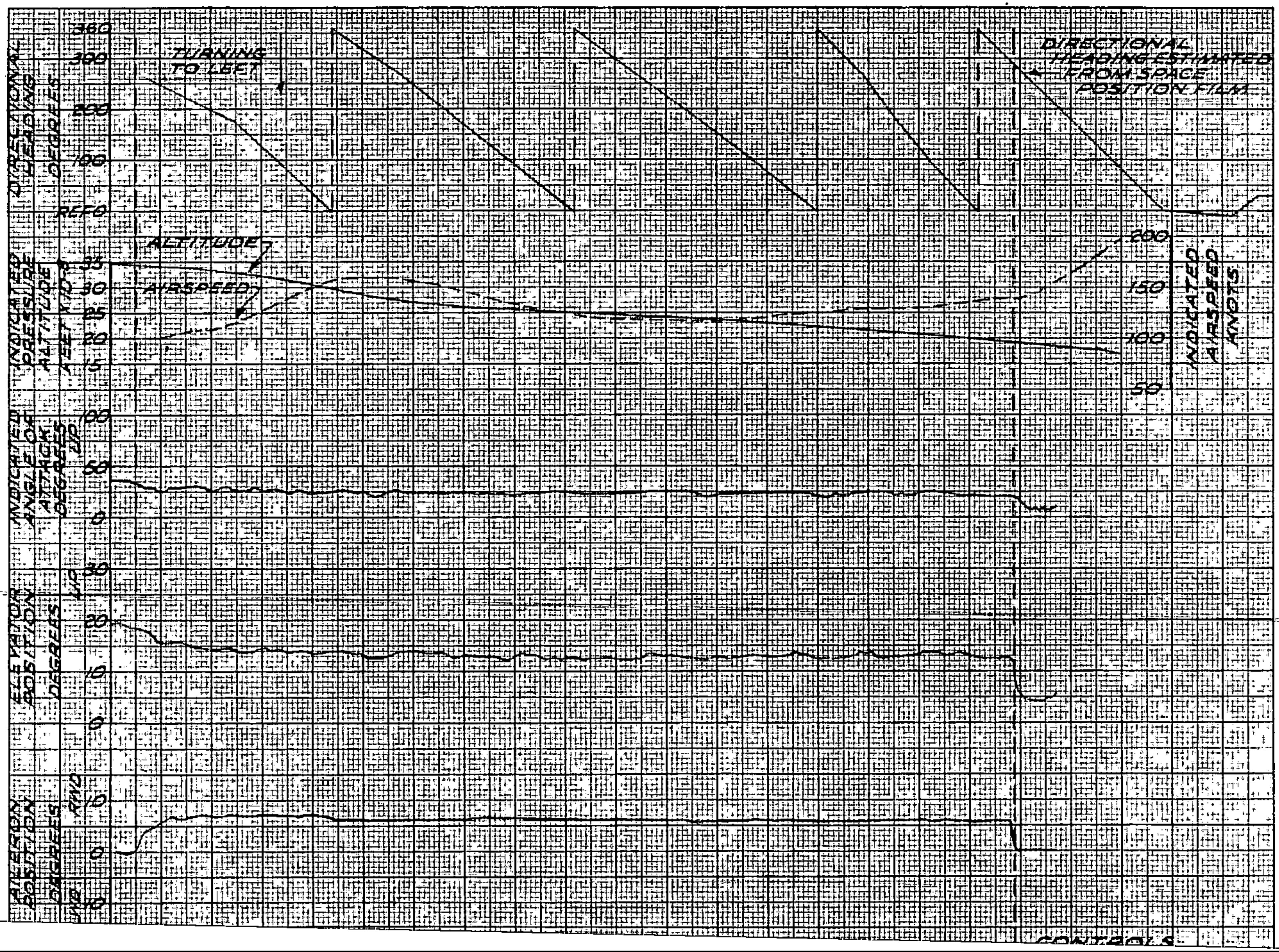
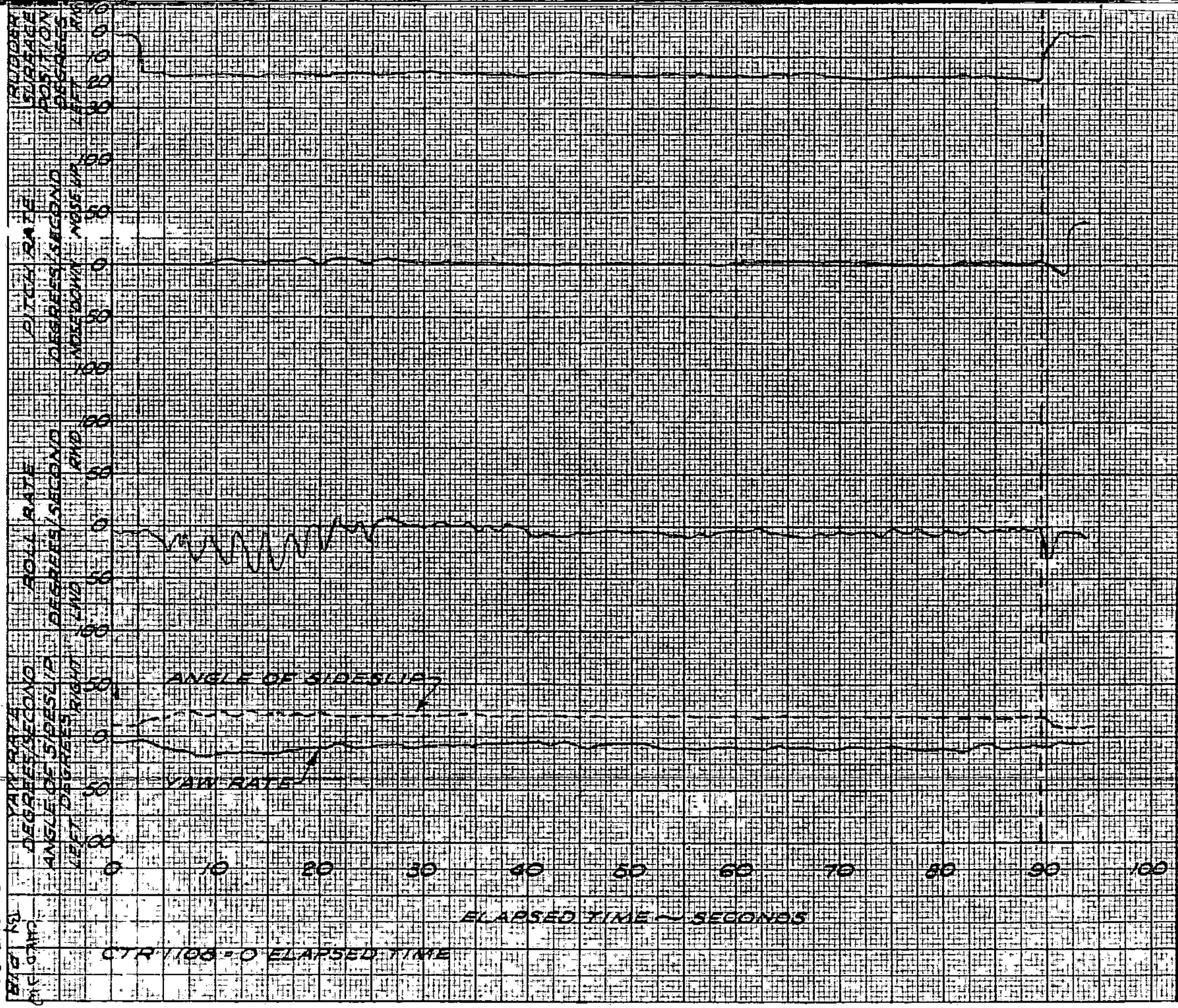


FIGURE 9 PAGE 25

SPIN TEST DATA

SIMULATED FULL ARMAMENT, ELEV. LOW, AILERON AGAINST FOR ENTRY
(LEFT RUDDER, RIGHT AILERON)

TF-102A S/N 54-1362
FLT 8 11 JUNE 1957



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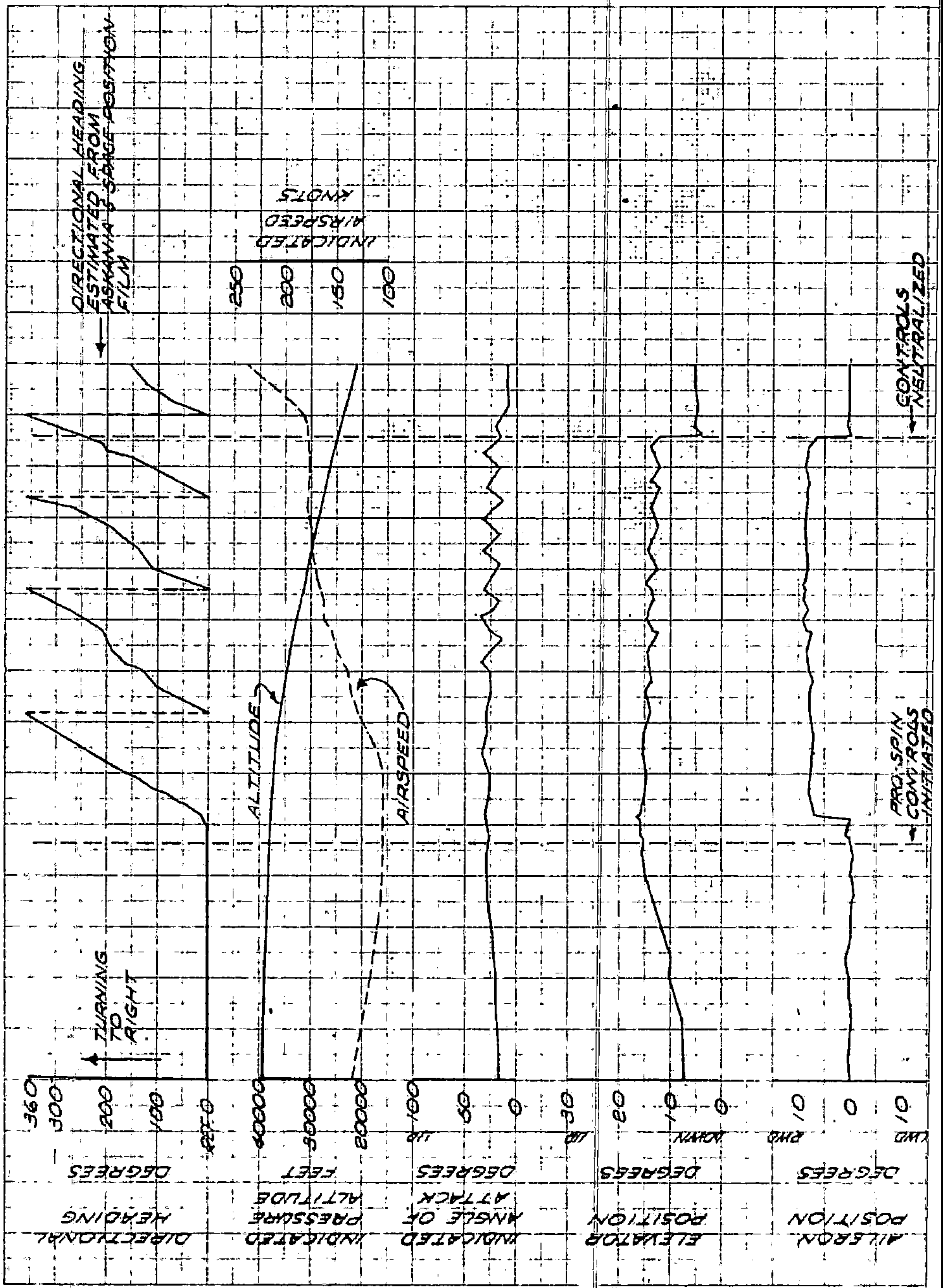
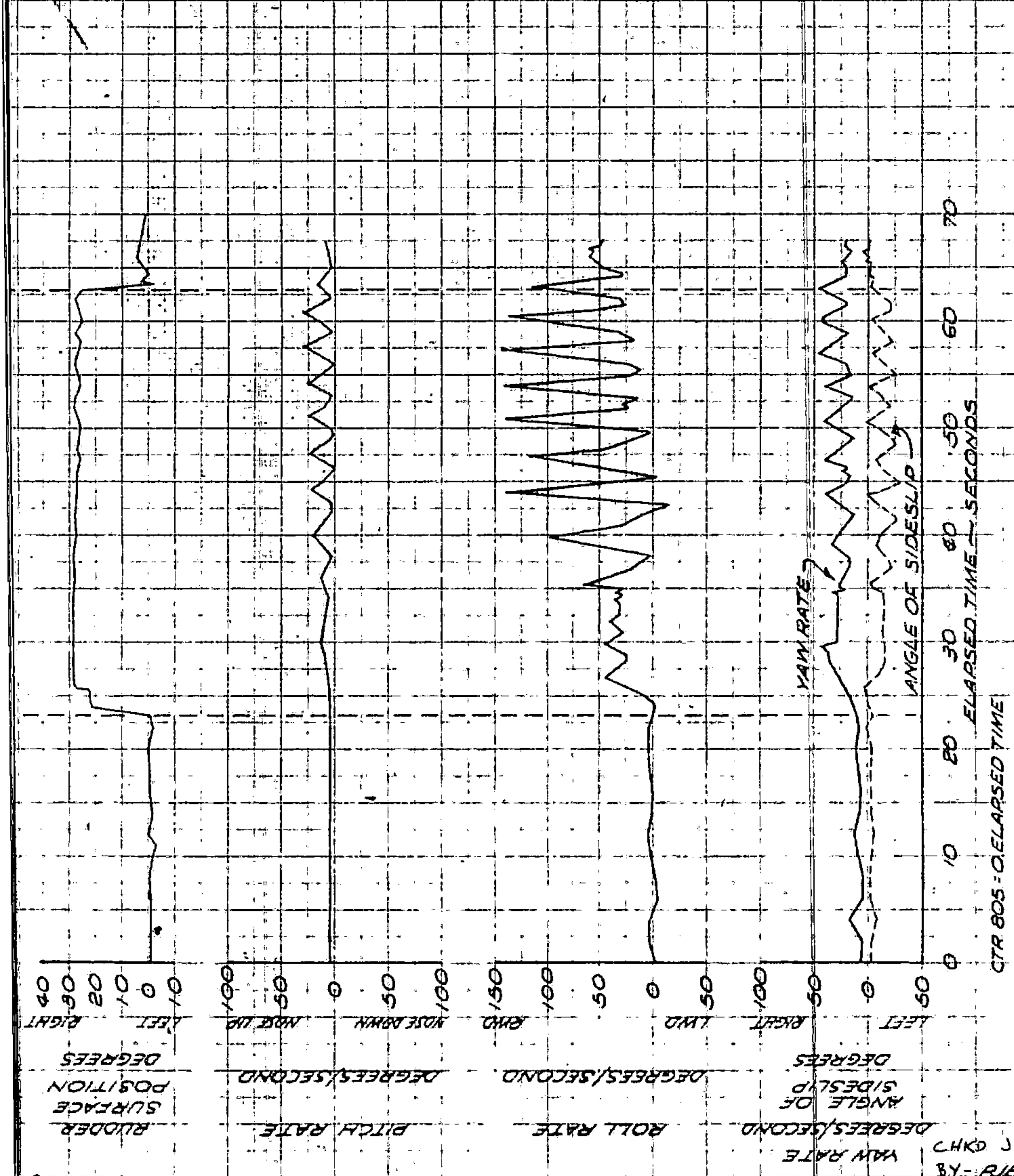


FIGURE 10

SPIN TEST DATA

SIMULATED FULL ARMAMENT, FUEL
LOW, AILERON WITH FOR ENTRY.
(RIGHT RUDDER, RIGHT AILERON)

TF-102A S/N 54-1362
FLT 8 10 JUNE 1957



CHKD JW
BY- RJB

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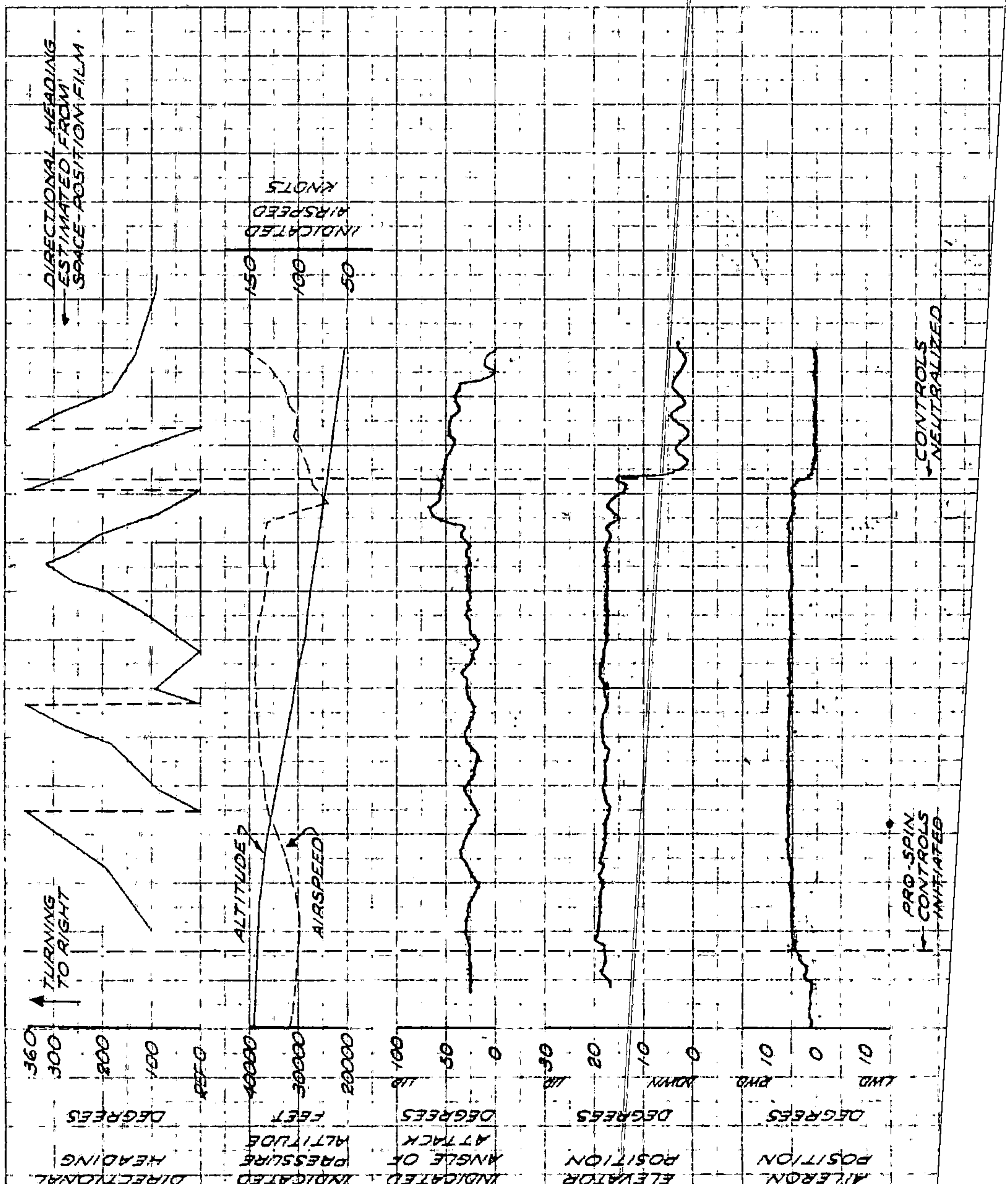
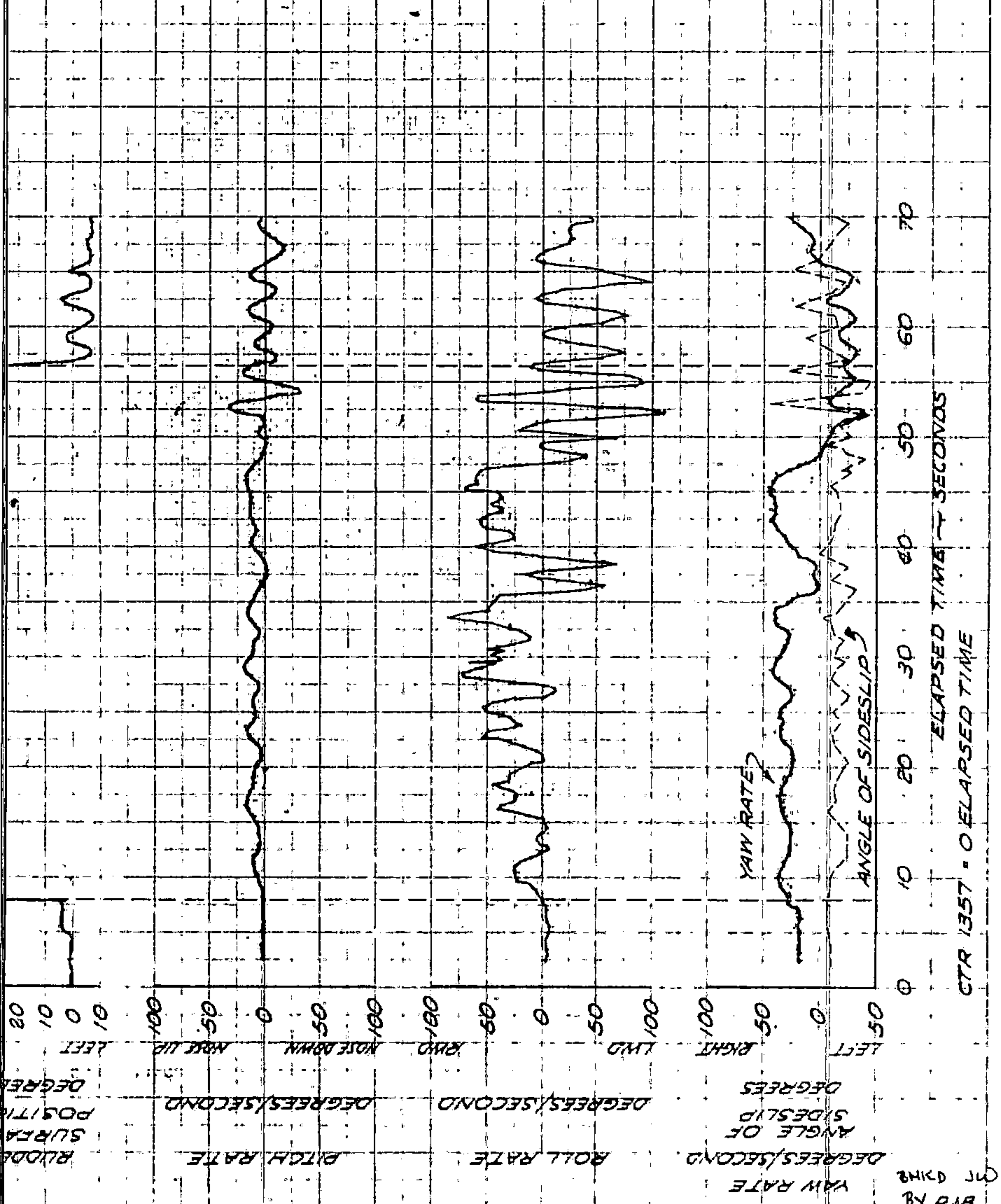


FIGURE 11

SPIN TEST DATA

SIMULATED FULL ARMAMENT, FUEL LOW, AILERON WITH FOR ENTRY (RIGHT RUDDER, RIGHT AILERON)

TF-102A S/N 54-1362
FLT 9 11 JUNE 1957



CHKD JW
BY PJB
EF-8-1362-31

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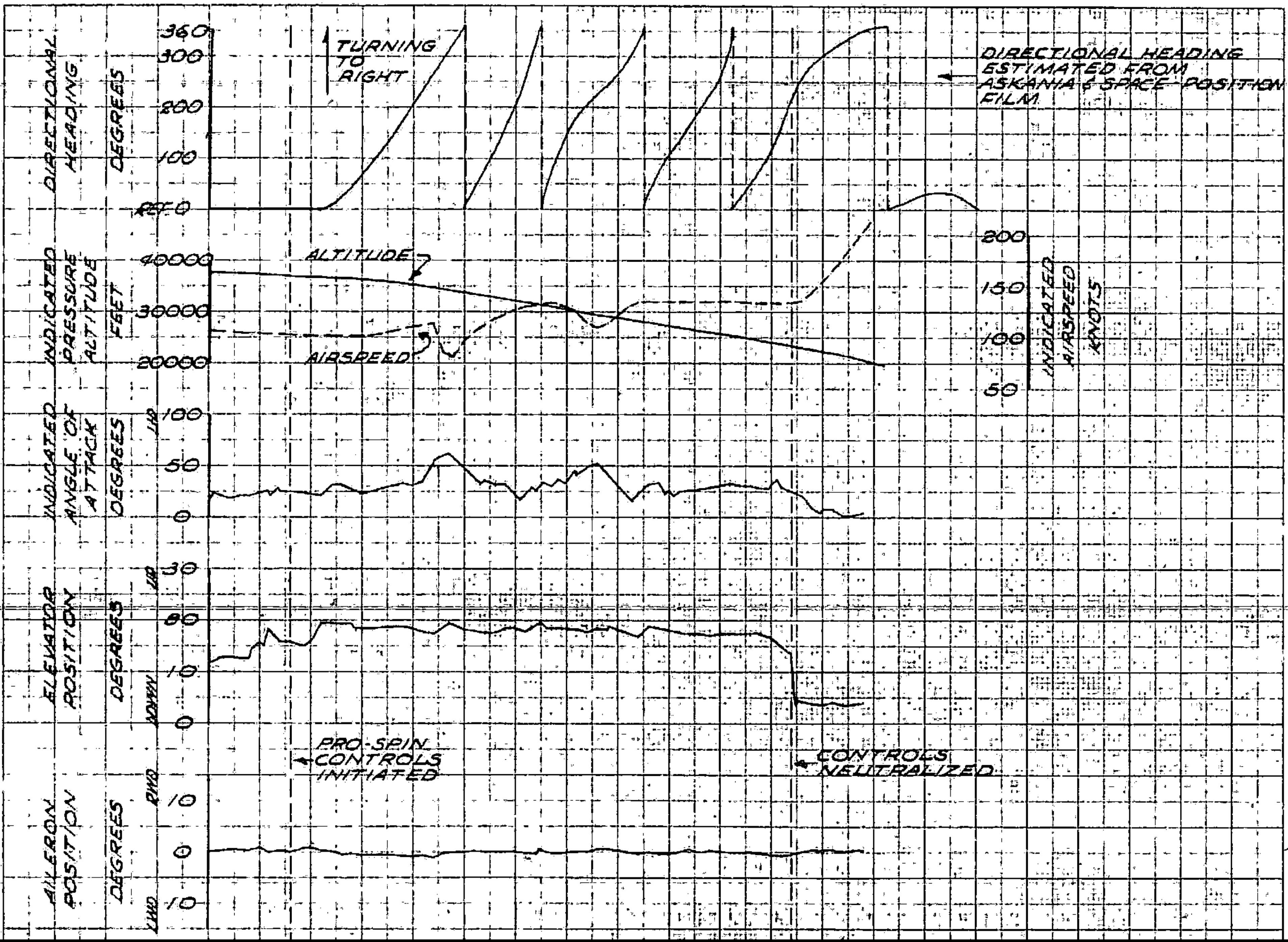


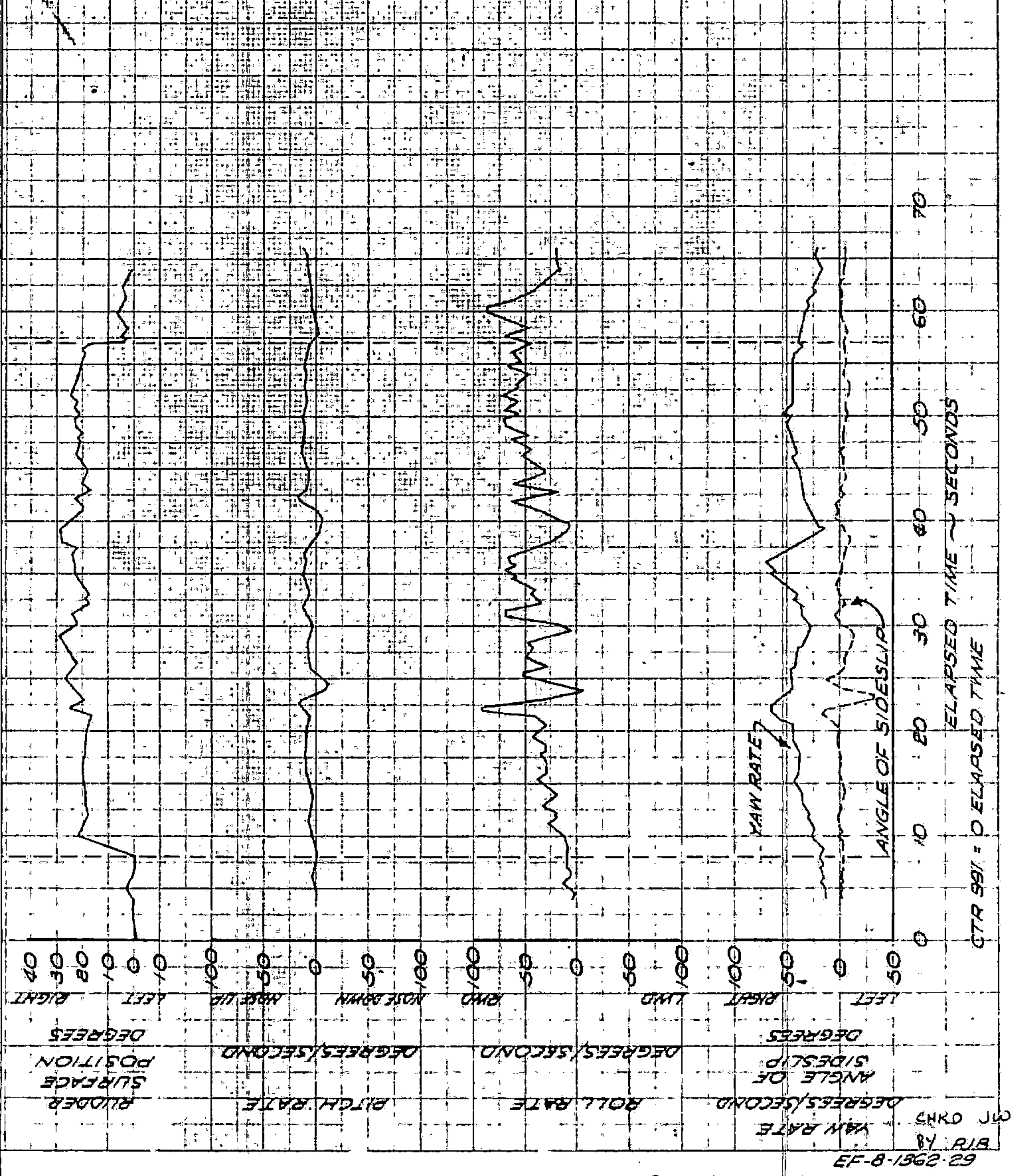
EFT-A-8-480

FIGURE 12 PAGE 28

SPIN TEST DATA

SIMULATED FULL ARMAMENT, FUEL LOW, AILERON NEUTRAL FOR ENTRY (RIGHT RUDDER)

TF-102A S/N 54-1362
FLT 8 10 JUNE 1957



CHKD JW
BY RJB
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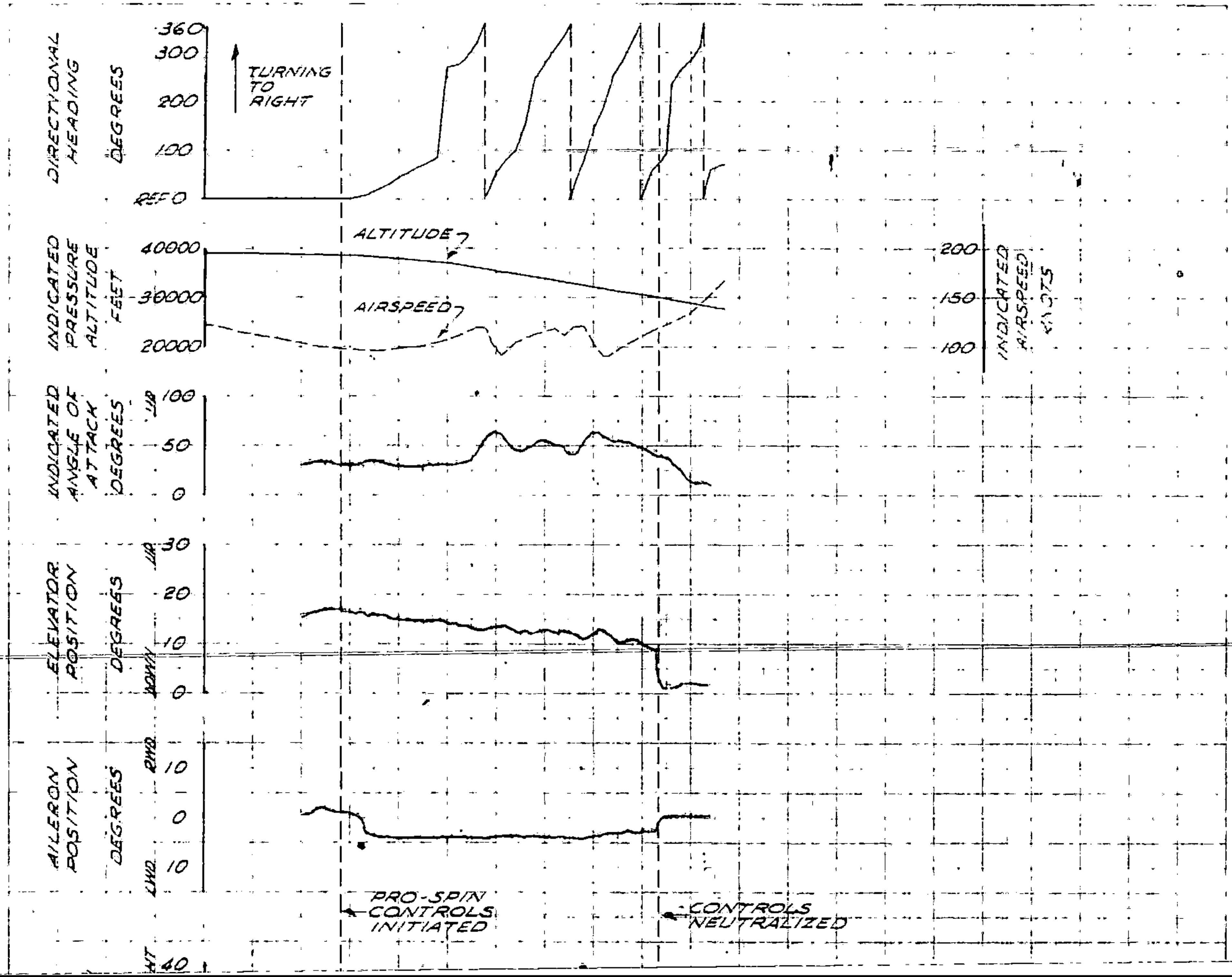


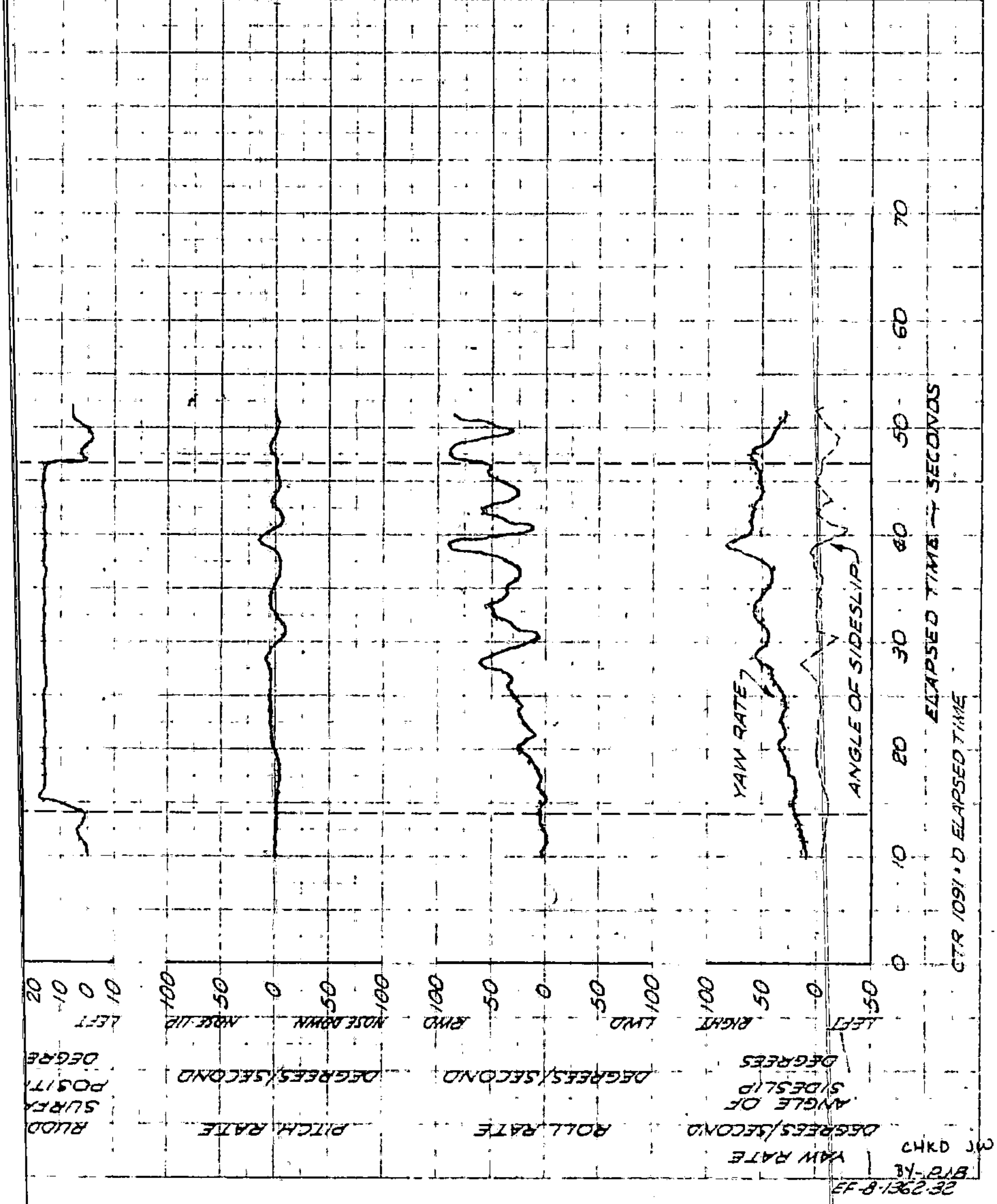
EFT-A-8-480

FIGURE 13 PAGE 29

SPIN TEST DATA

ARMAMENT EXPENDED, HEAVY FUEL LOAD, AILERON AGAINST FOR ENTRY (RIGHT RUDDER, LEFT AILERON)

TF-102A S/N 54-1362
FLT 10 12 JUNE 1957.



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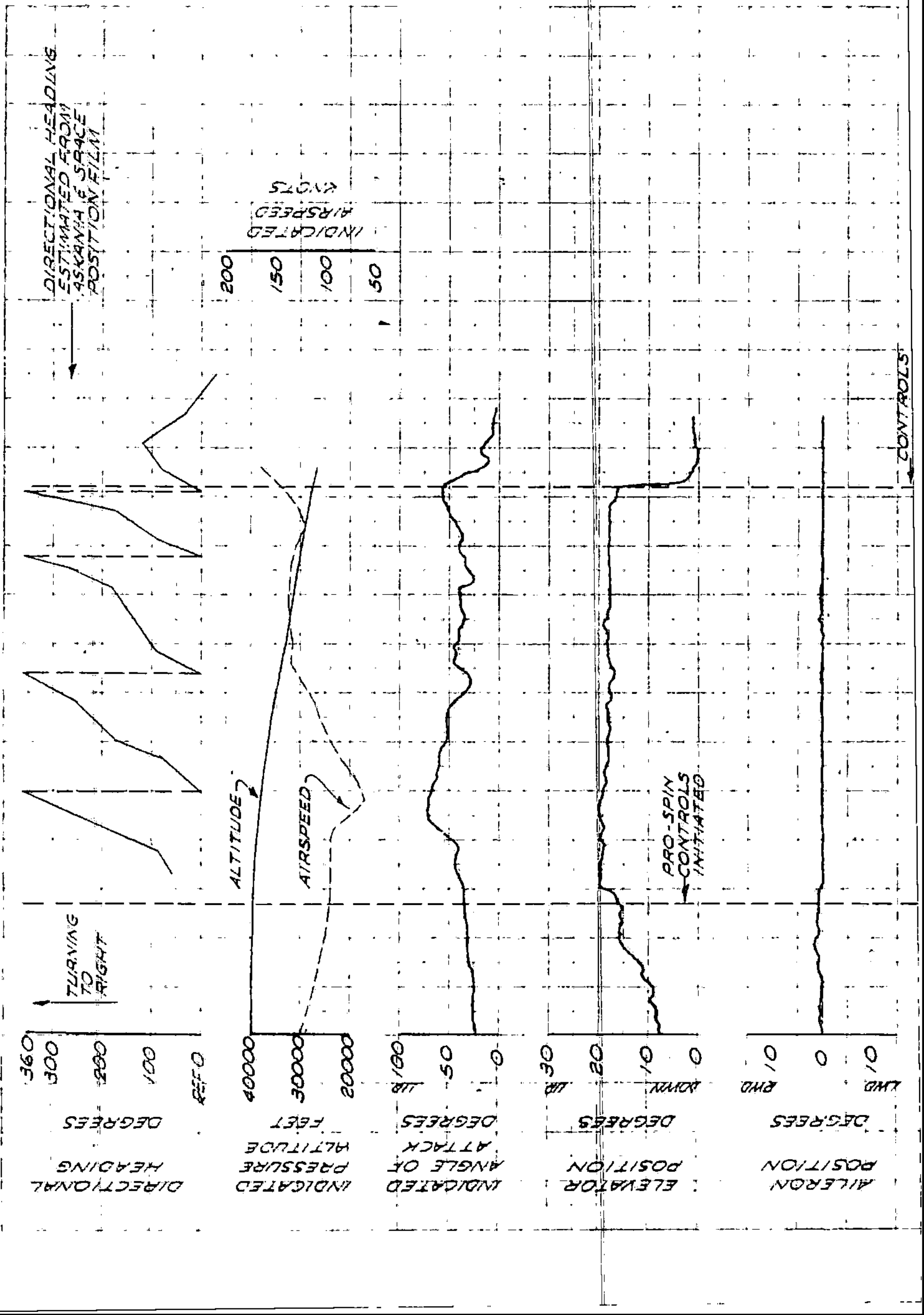
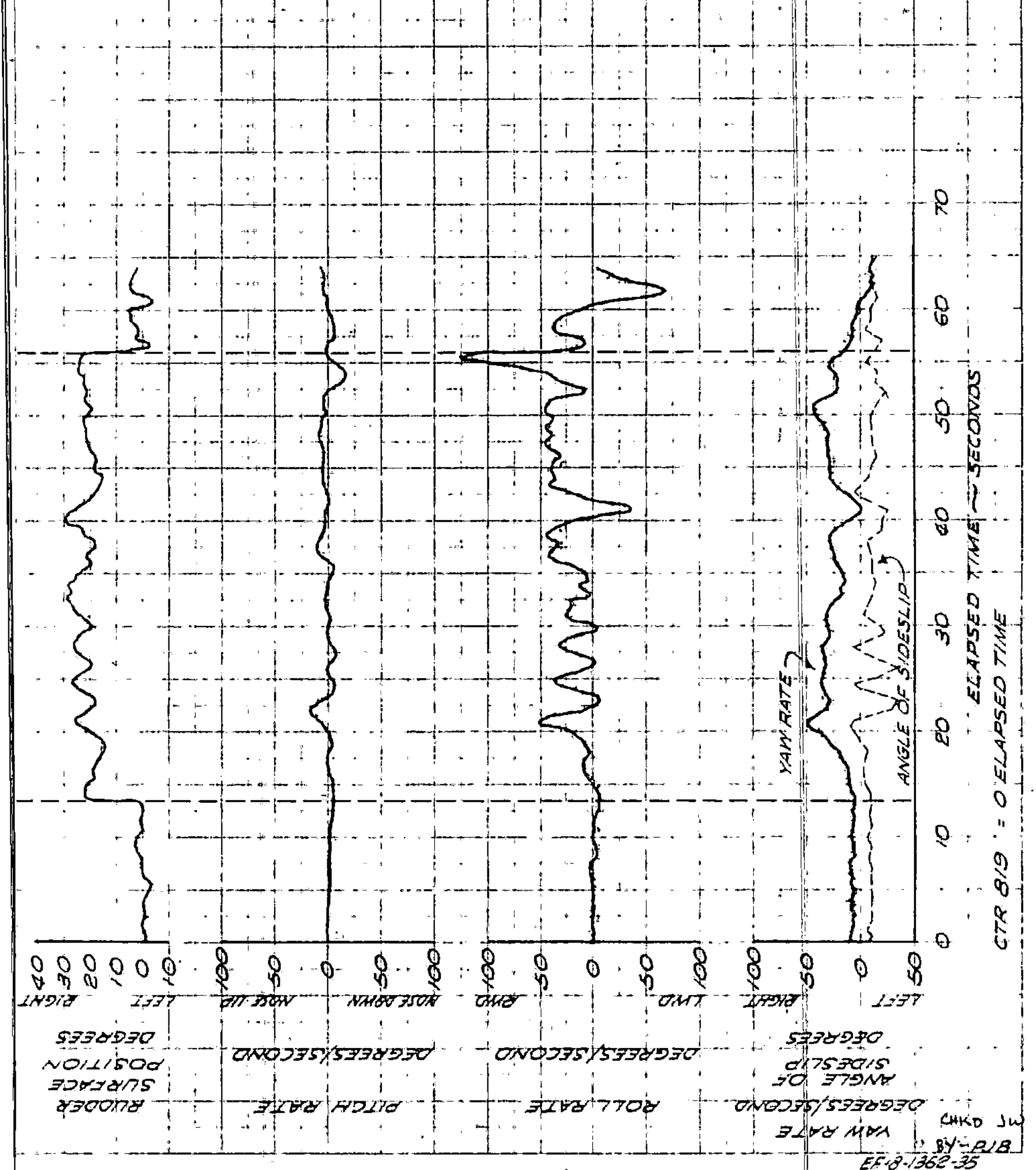


FIGURE 14 PAGE 30

SPIN TEST DATA

ARMAMENT EXPENDED, HEAVY FUEL LOAD, AILERON NEUTRAL (RIGHT RUDDER) FOR ENTRY

TF-102A S/N 54-1362
FLT 11 14 JUNE 1957



CHKD JW
BY-BJB
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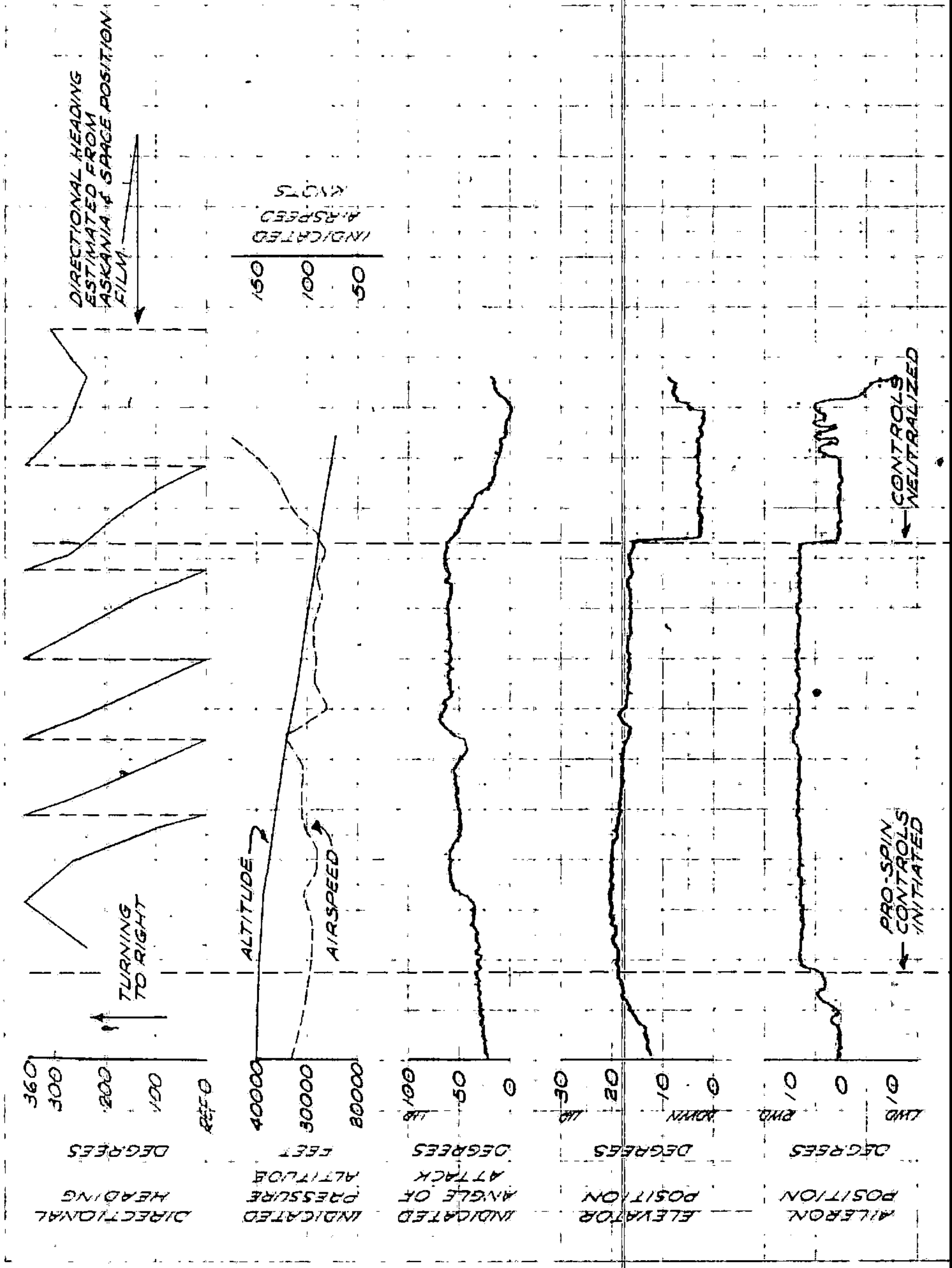
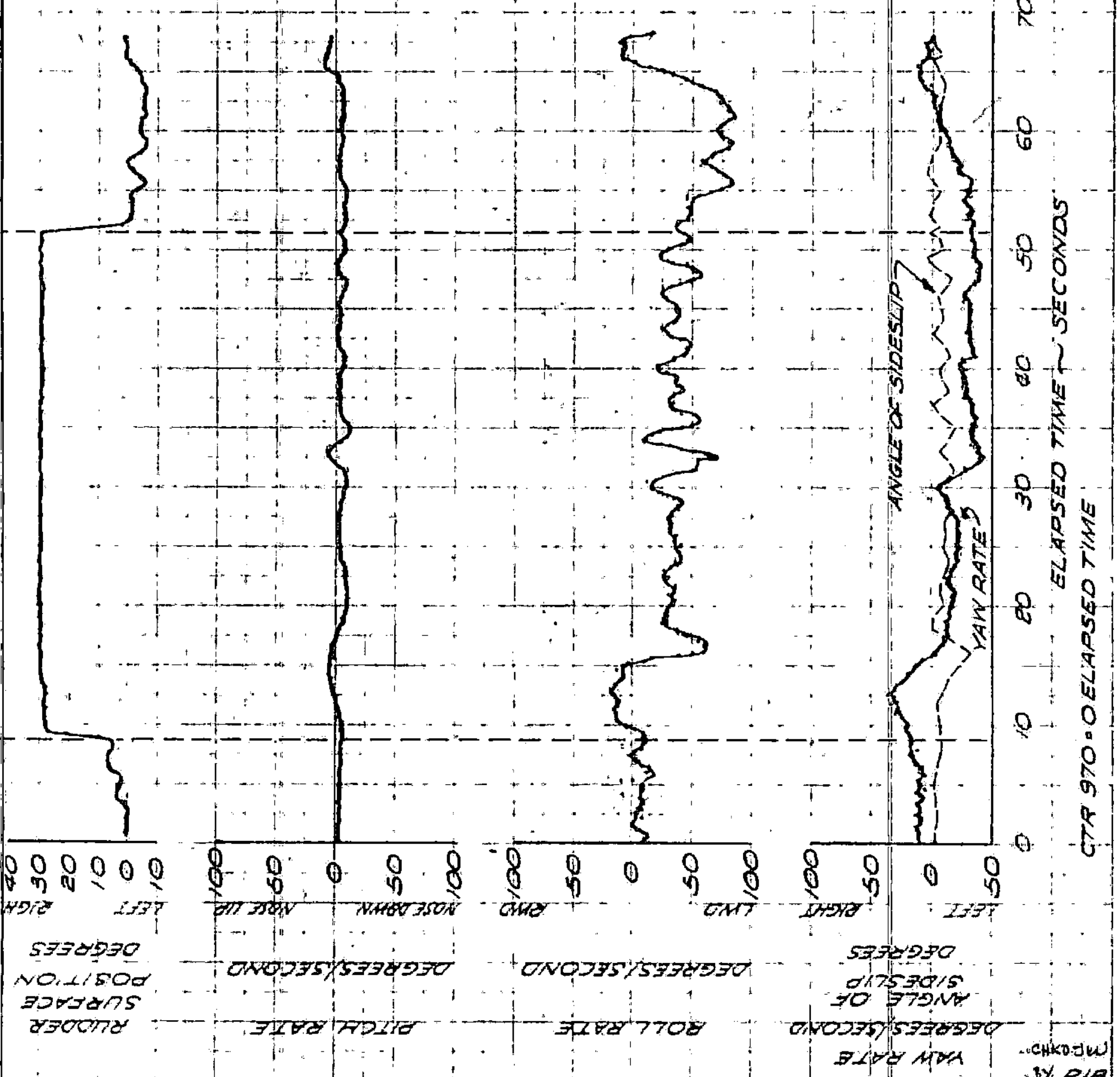


FIGURE 15 PAGE 31

SPIN TEST DATA

ARMAMENT EXPENDED, HEAVY FUEL LOAD, AILERON WITH FOR ENTRY
(RIGHT RUDDER, RIGHT AILERON)

TF-102A S/N 54-1362
FLT 12 14 JUNE 1957



CHKD JW
BY RJB
EF-8-1362-38

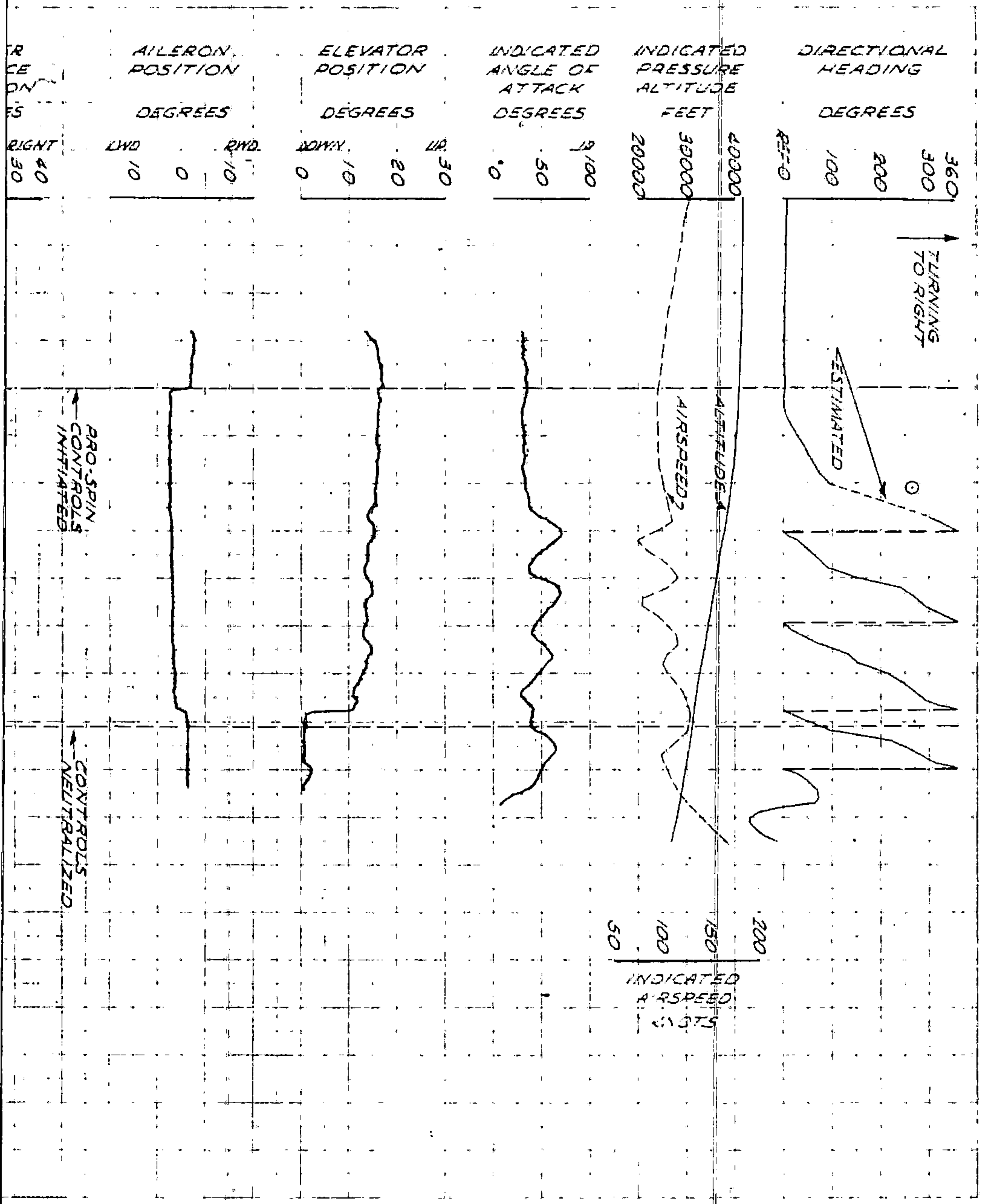


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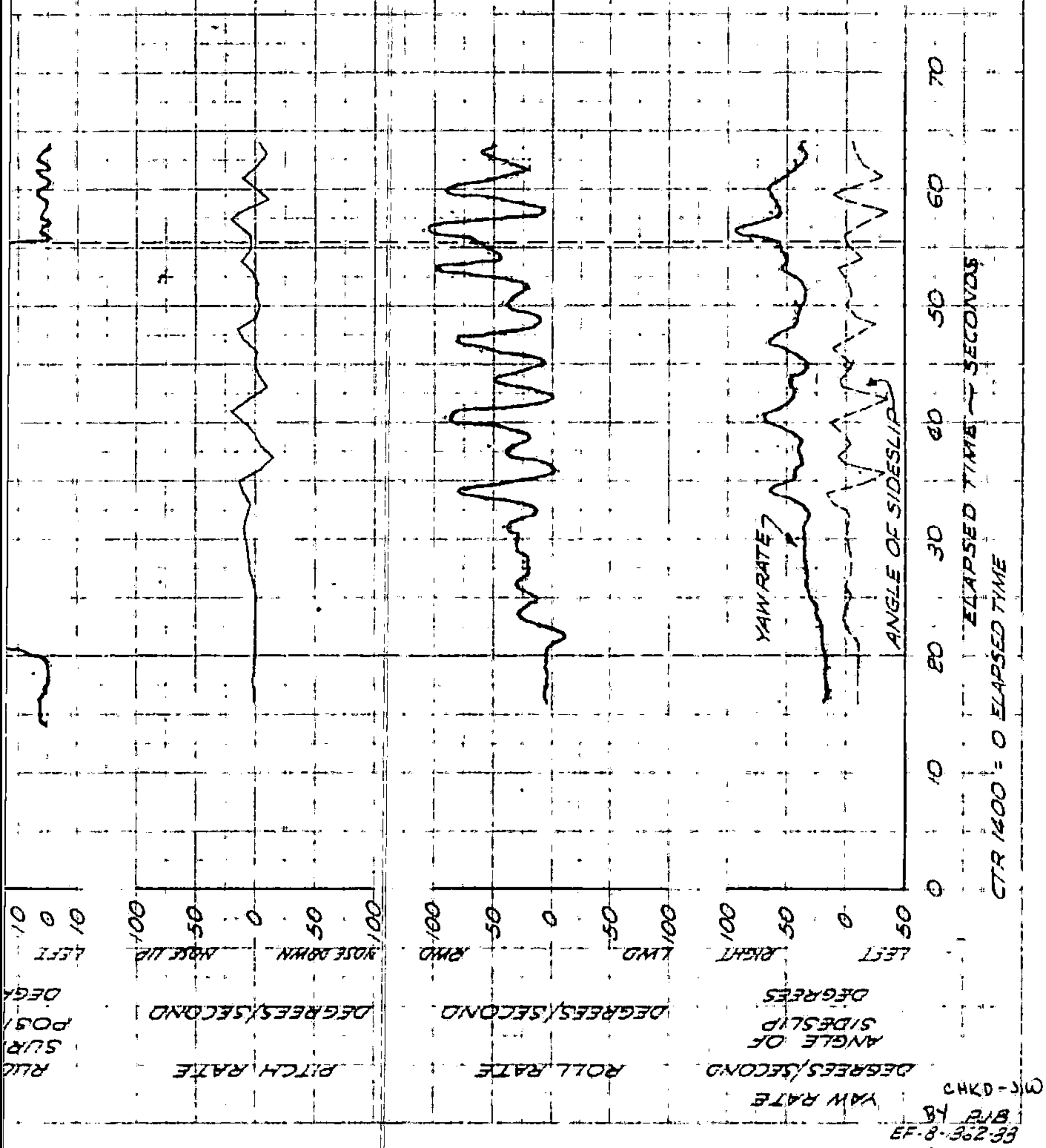
FIGURE 16 PAGE 32

SPIN TEST DATA

ARMAMENT EXPENDED, FUEL LOW, AILERON AGAINST FOR ENTRY (RIGHT RUDDER, LEFT AILERON)

TF-102A S/N 54-1362
FLT. 10 12 JUNE 1957



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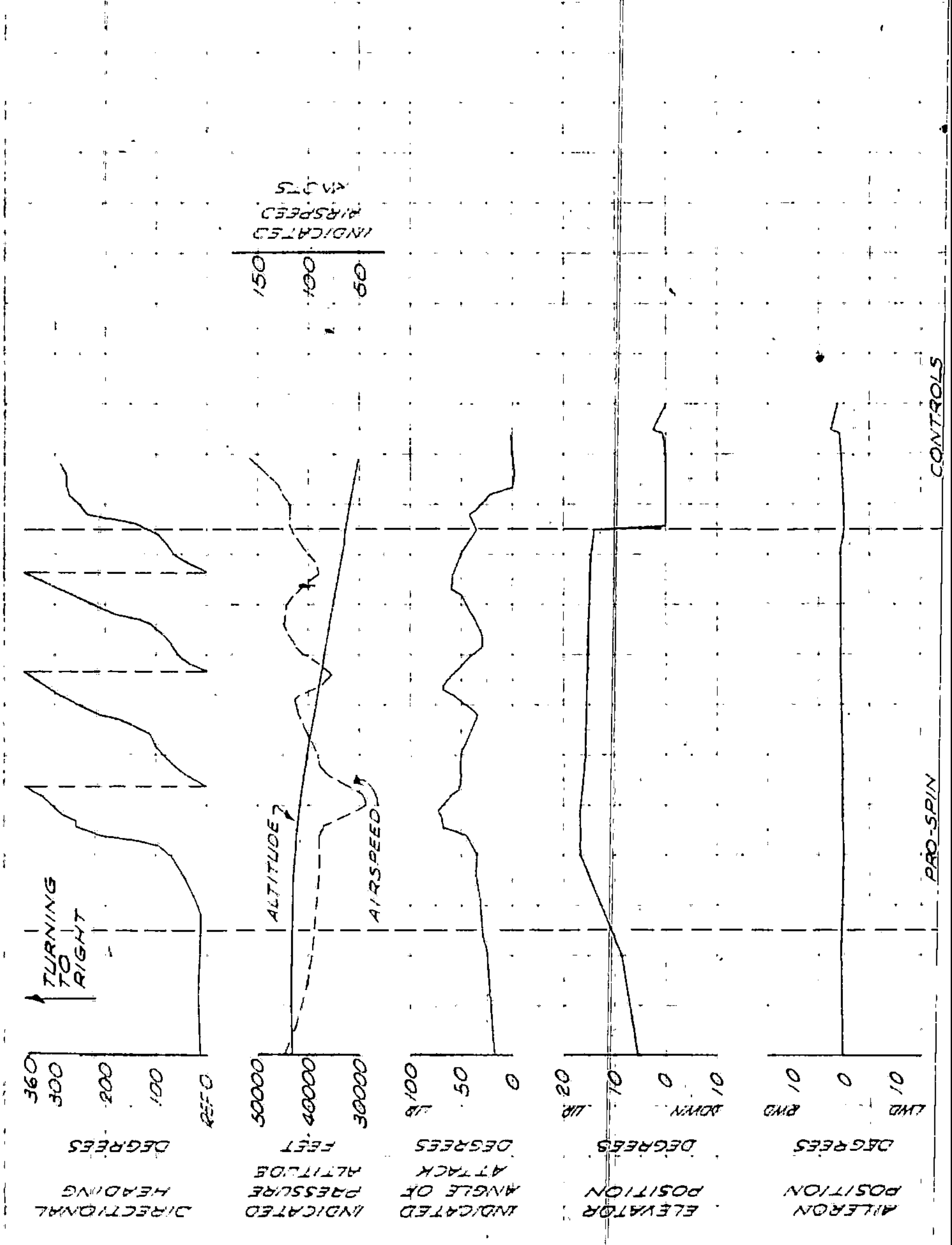
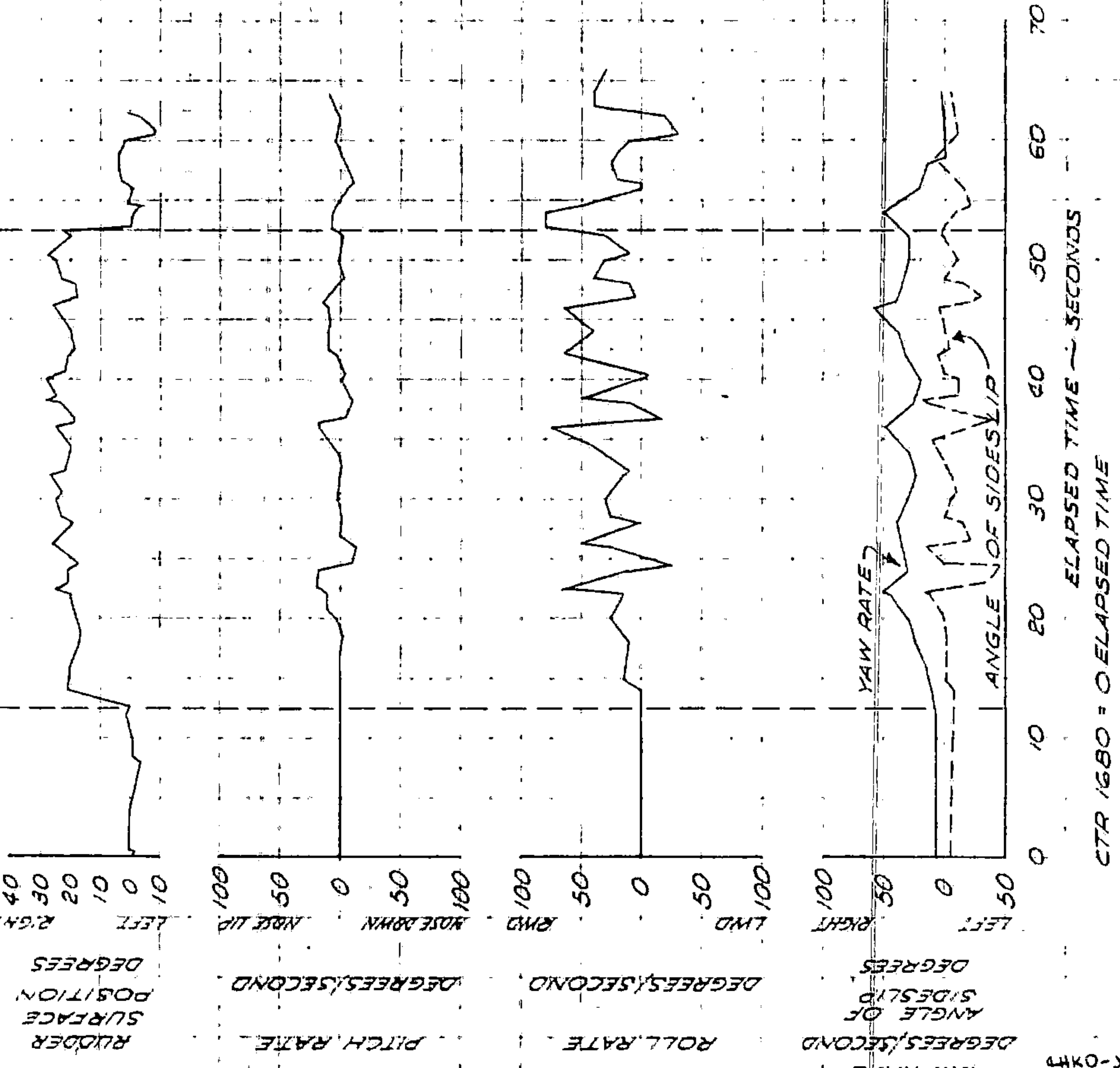


FIGURE 19 PAGE 33

SPIN TEST DATA

ARMAMENT EXPENDED, FUEL LOW,
AILERON NEUTRAL (RIGHT RUDDER
FOR ENTRY)

TF-102A S/N 54-1362
FLT 10 12 JUNE 1957



GHKO-310
BY-PJB
EF-8-1362-34

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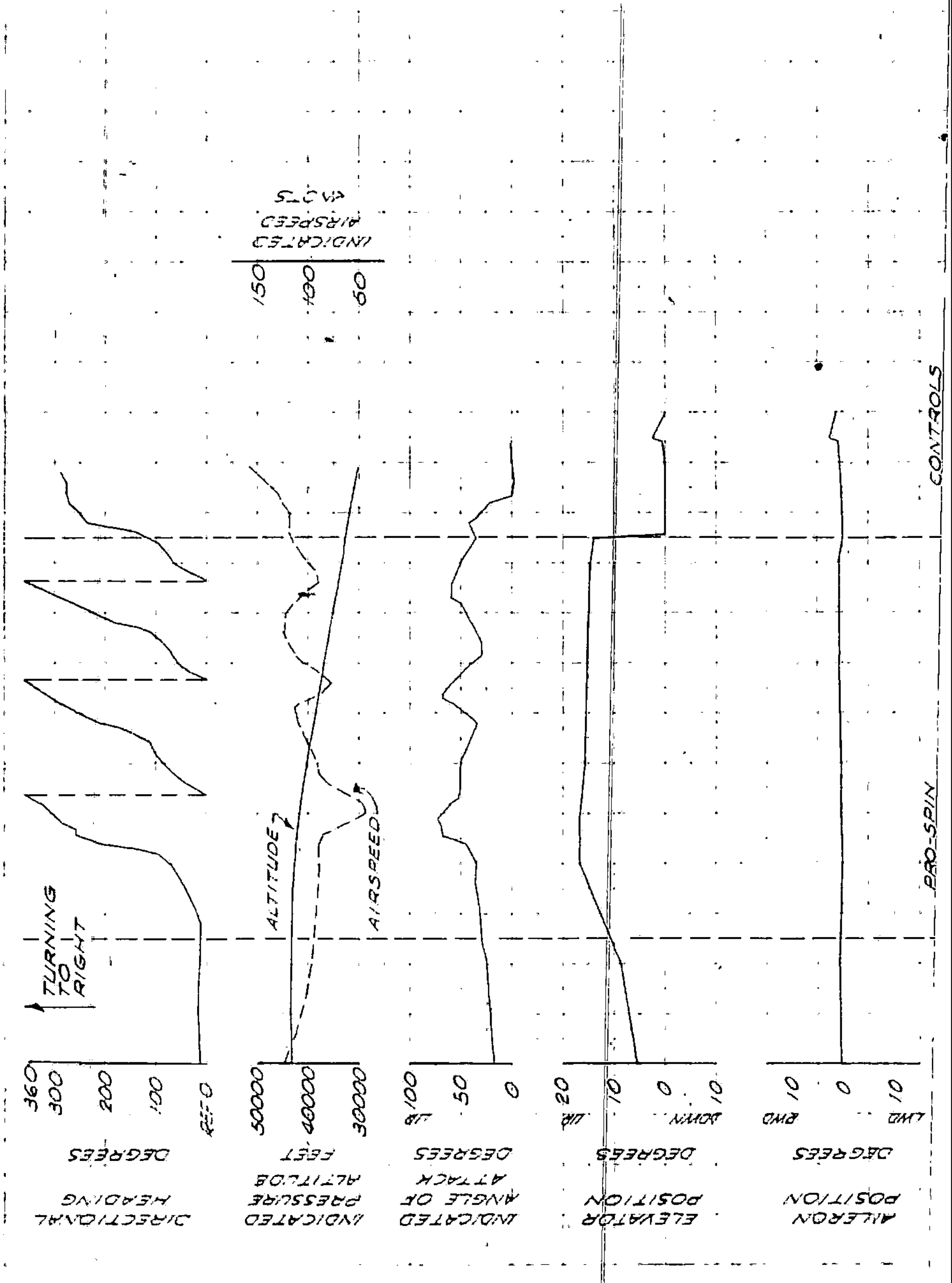

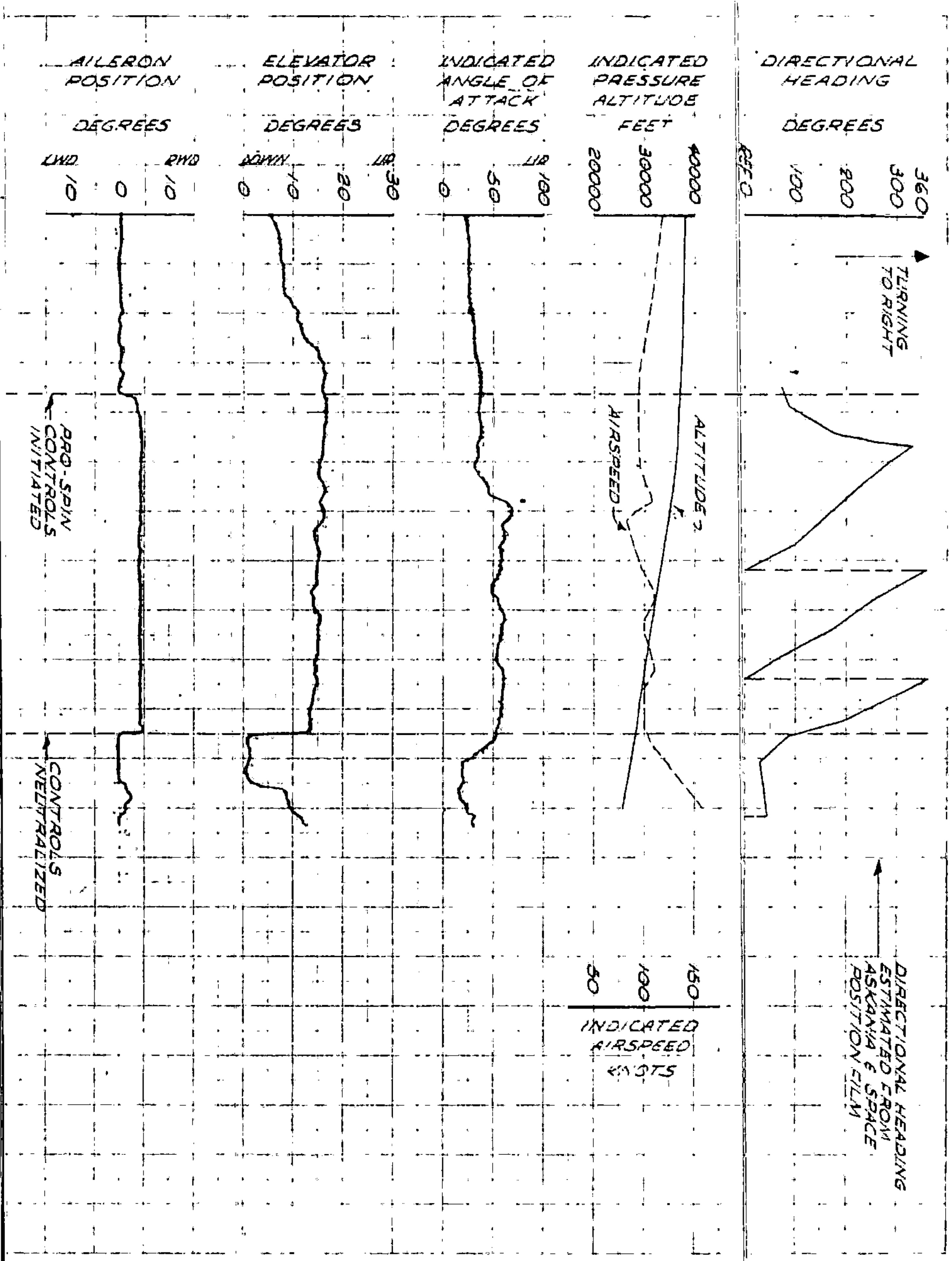

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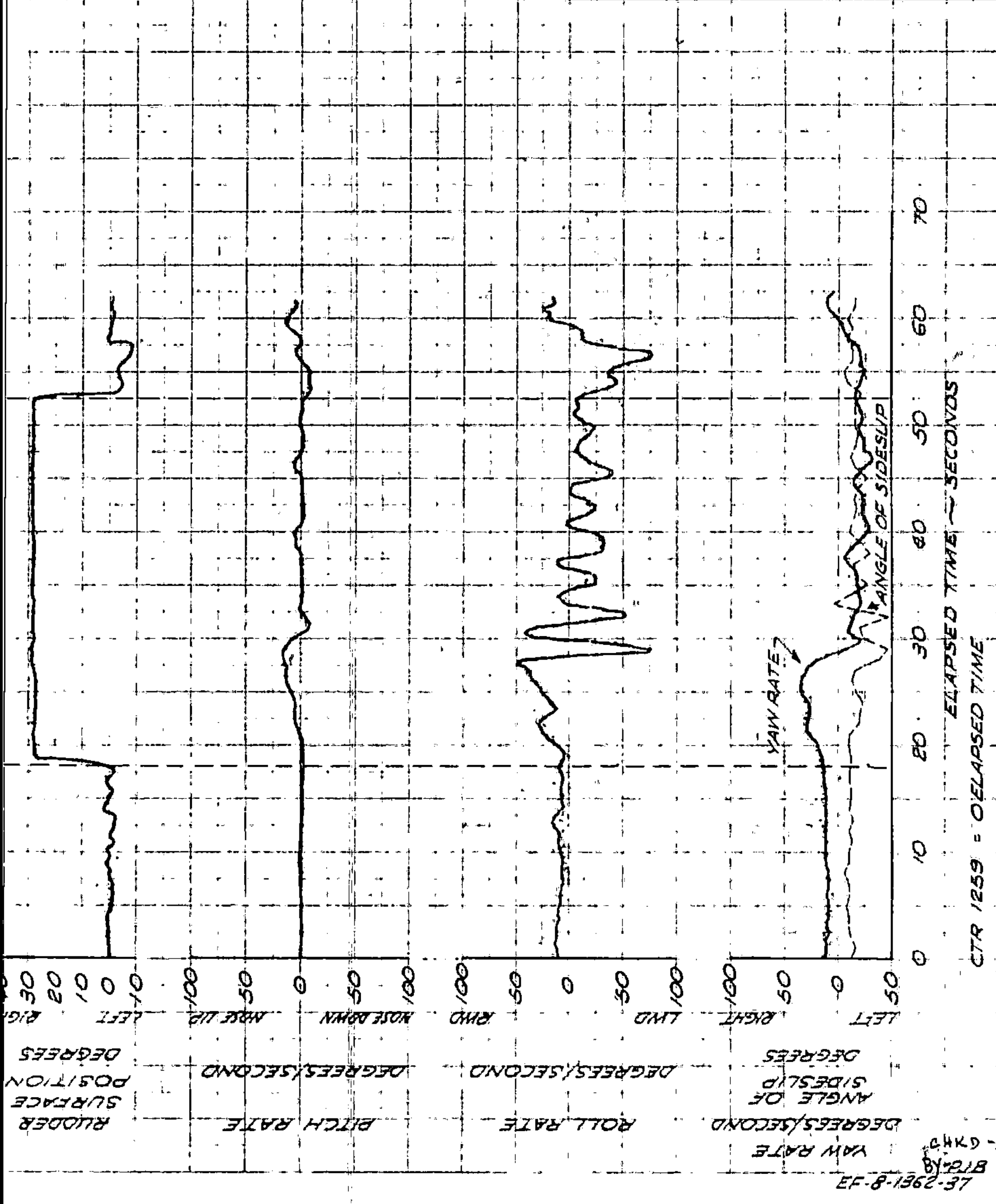
FIGURE 18 PAGE 34

SPIN TEST DATA

ARMAMENT EXPENDED, FUEL LOW;
AILERON WITH FOR ENTRY.
(RIGHT RUDDER, RIGHT AILERON)

TF-102A S/N 52-1362
FLT 11 14 JUNE 1957



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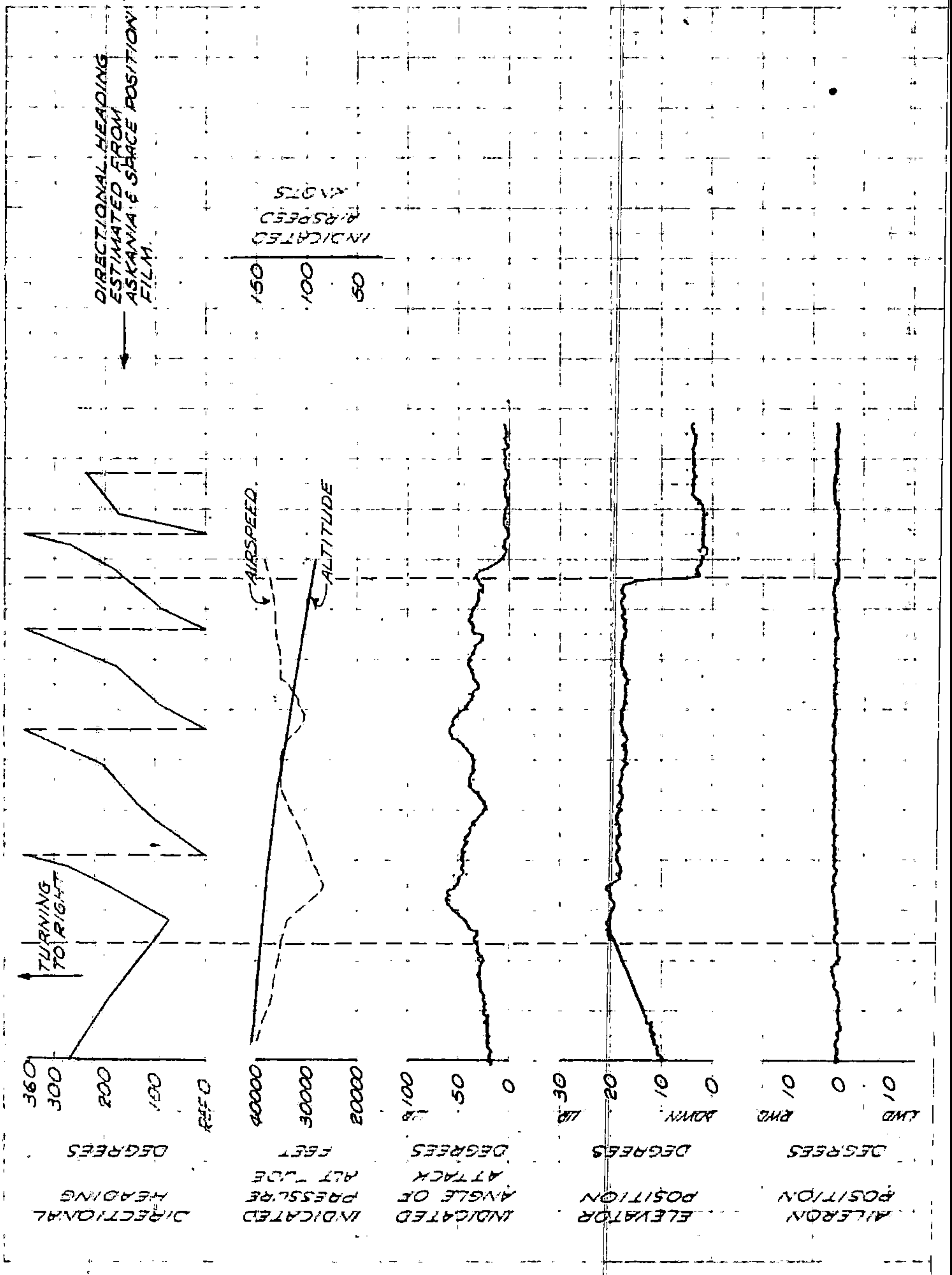
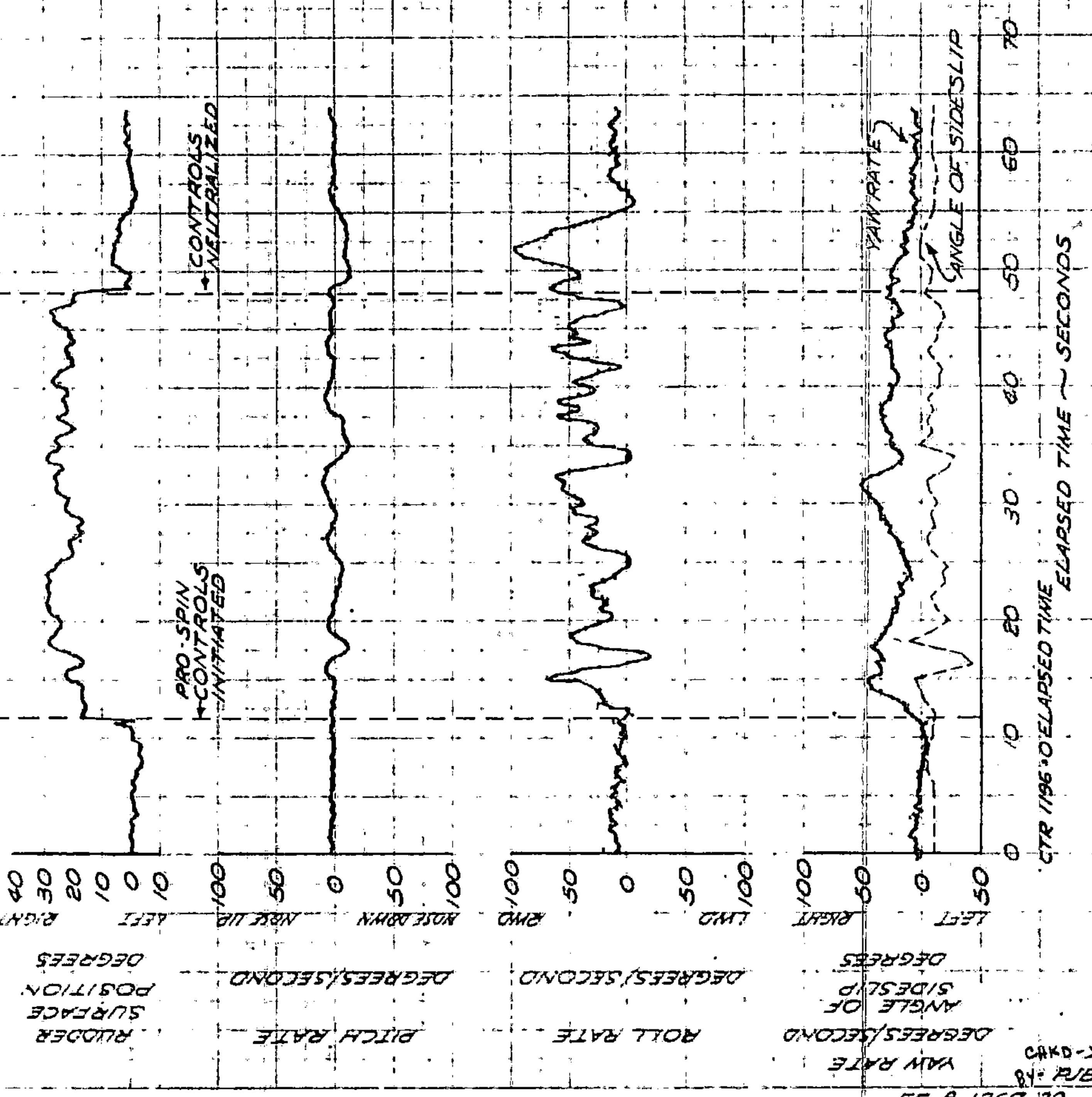


FIGURE 19 PAGE 35

SPIN TEST DATA

ACCELERATED ENTRY, ARMAMENT EXPENDED, LOW FUEL, AILERON NEUTRAL

TF-102A S/N 54-1362
FLT 18 14 JUNE 1957



CHKD-JW
BY: RJB

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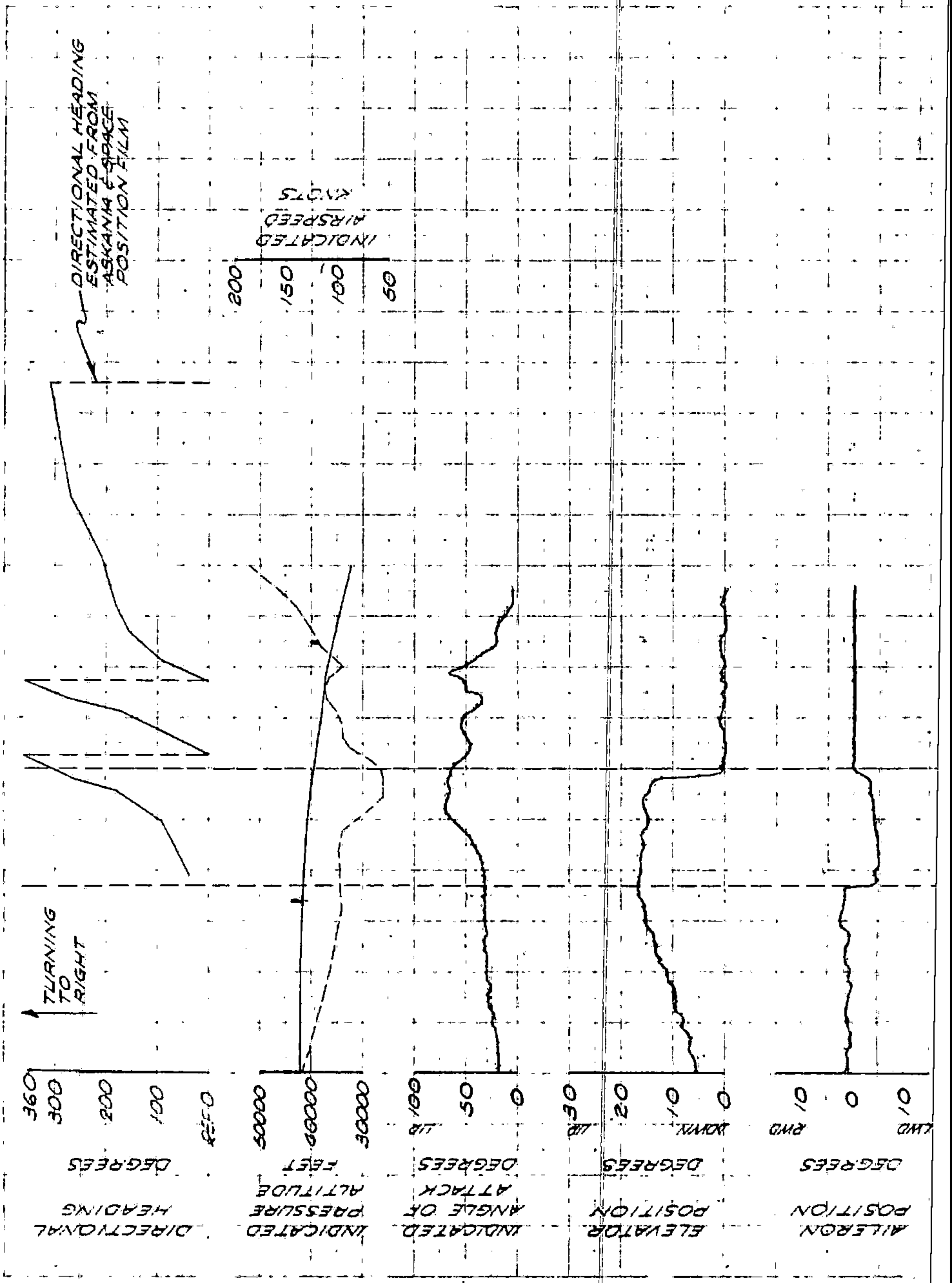
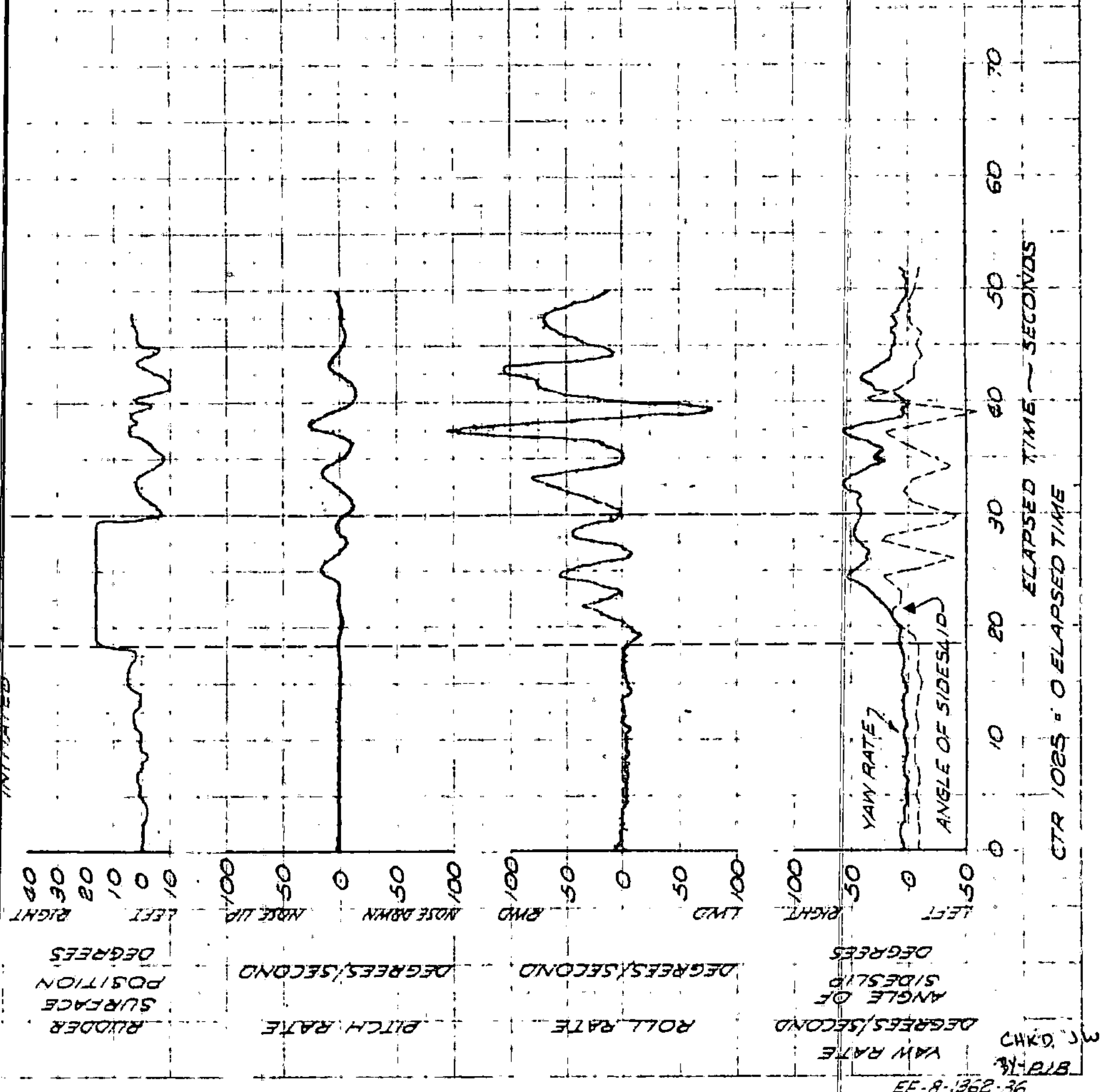


FIGURE 20 PAGE 36

SPIN TEST DATA

ARMAMENT EXPENDED, FUEL LOW, GEAR DOWN, FOR LANDING, AILERON AGAINST FOR ENTRY (RIGHT RUDDER, LEFT AILERON)

TF-102A S/N 54-1362
FLT 11 14 JUNE 1957



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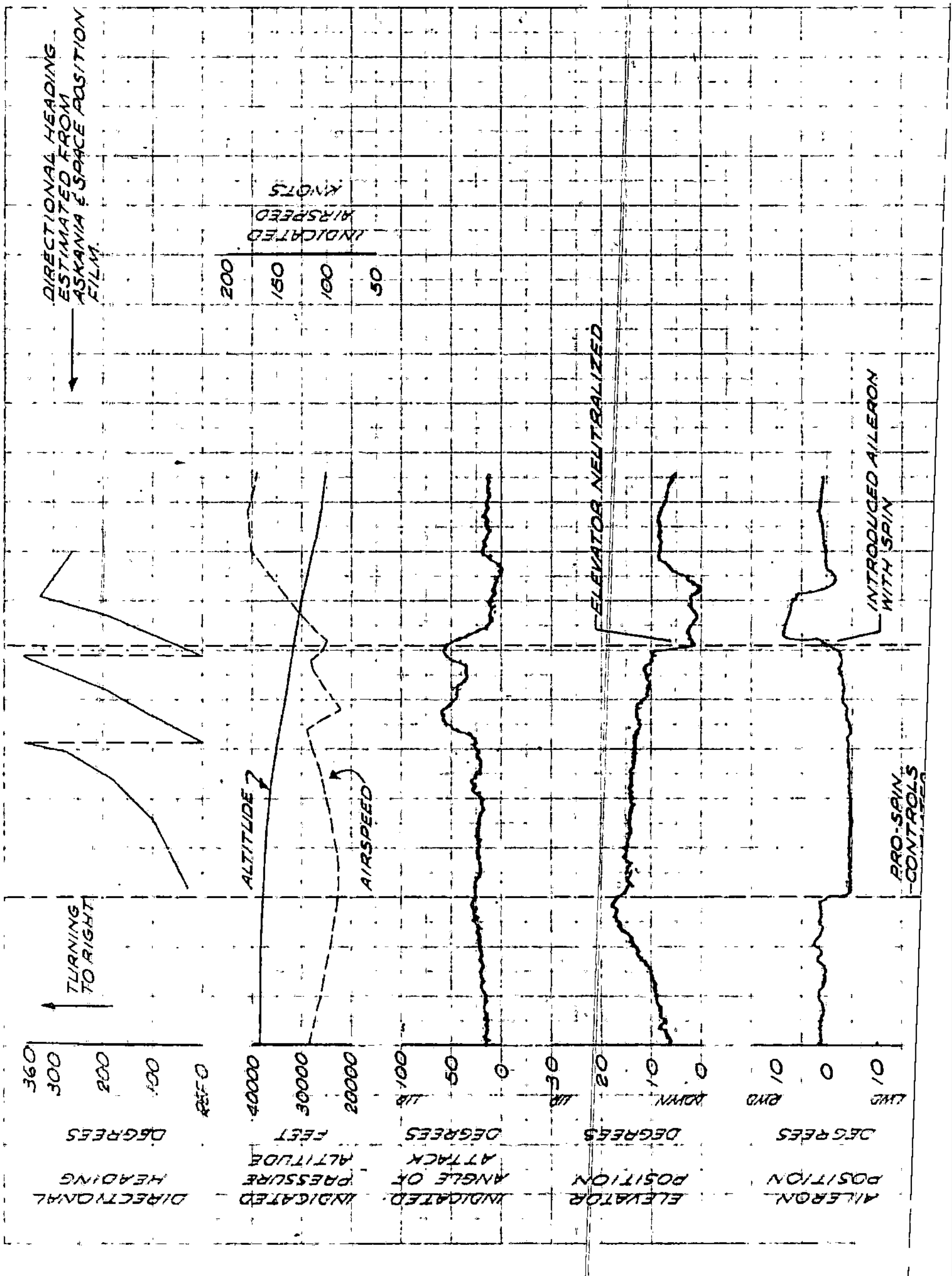
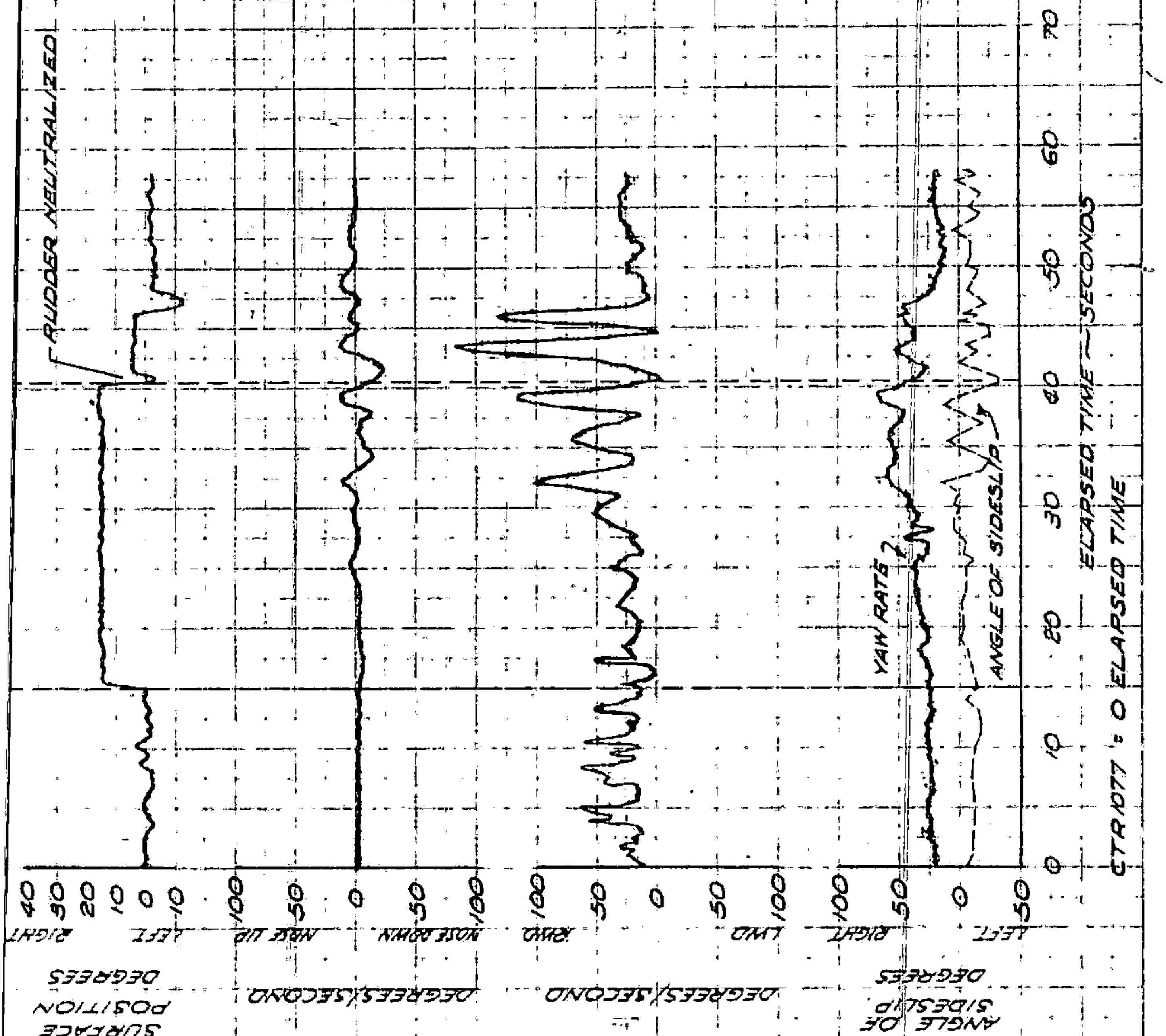


FIGURE 21

SPIN TEST DATA

ARMAMENT EXPENDED, FUEL LOW GEAR DOWN FOR LANDING, AILERON AGAINST FOR ENTRY (RIGHT RUDDER, LEFT AILERON)

TF-102A S/N 54-1362
FLT 18 19 JUNE 1957



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BY- RJB

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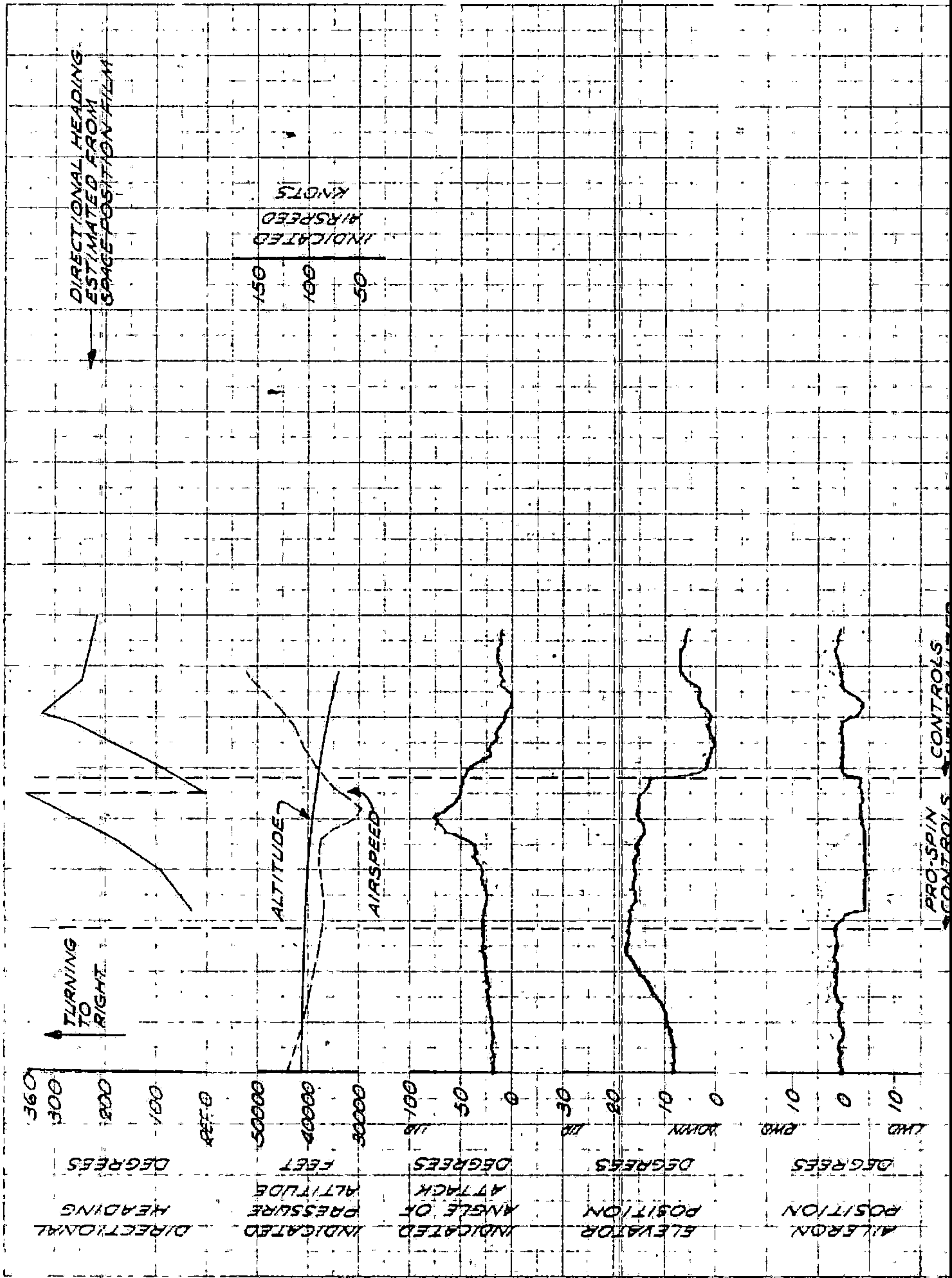
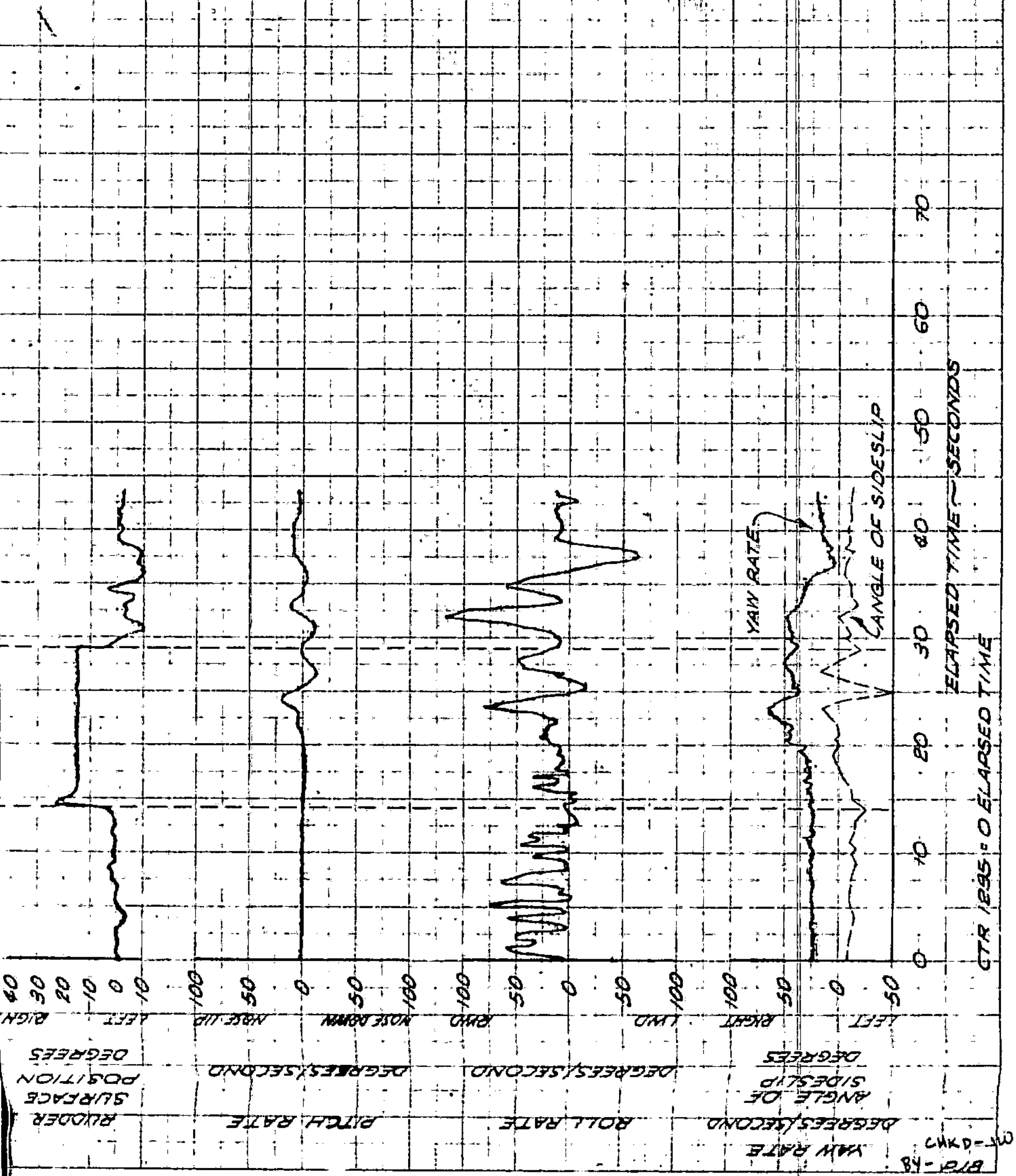


FIGURE 22 PAGE 38

SPIN TEST DATA

ARMAMENT EXPENDED, FUEL LOW, GEAR DOWN FOR LANDING, AILERON AGAINST FOR ENTRY (RIGHT RUDDER, LEFT AILERON)

TF-102A S/N 54-1362
FLT 13 19 JUNE 1957



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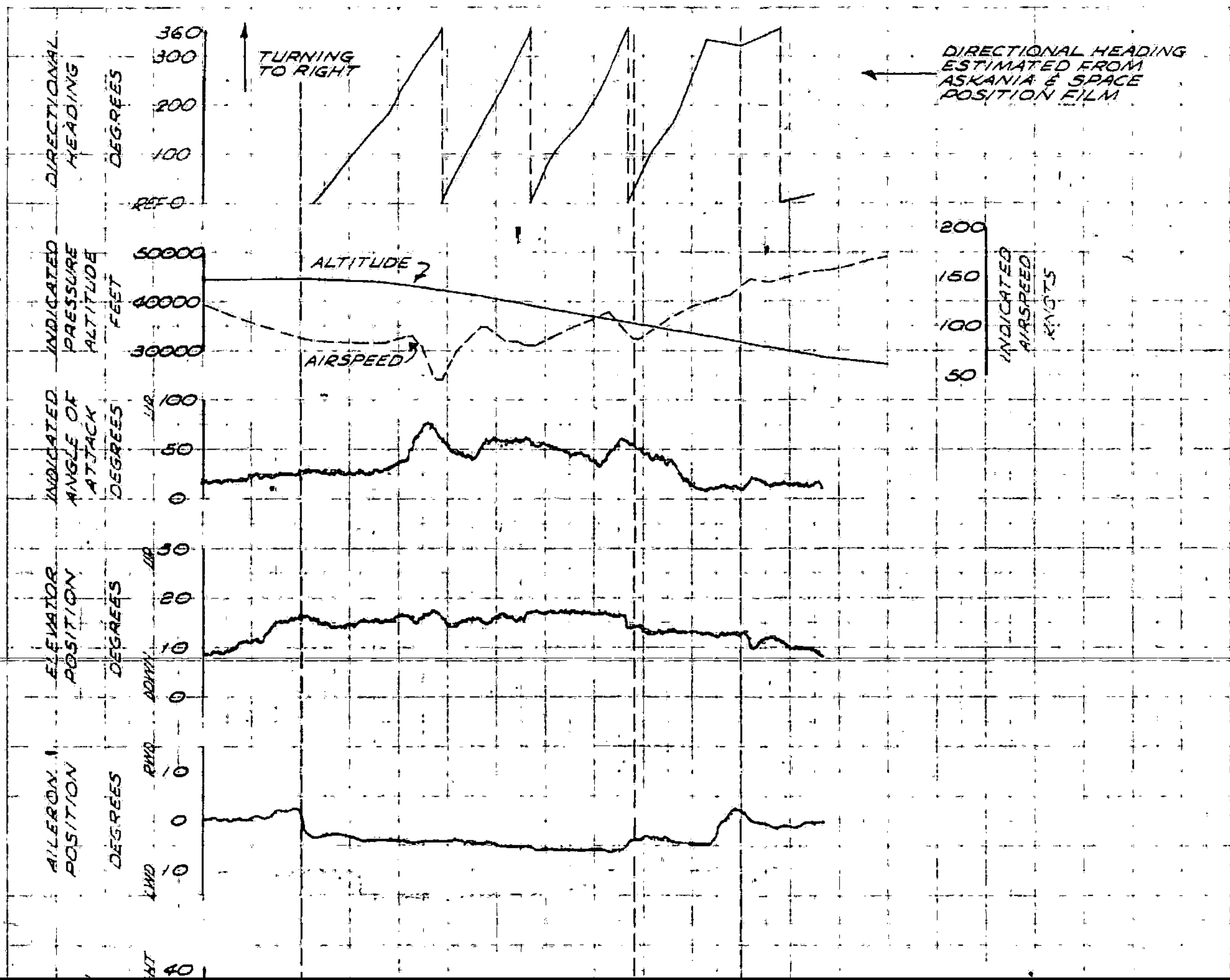


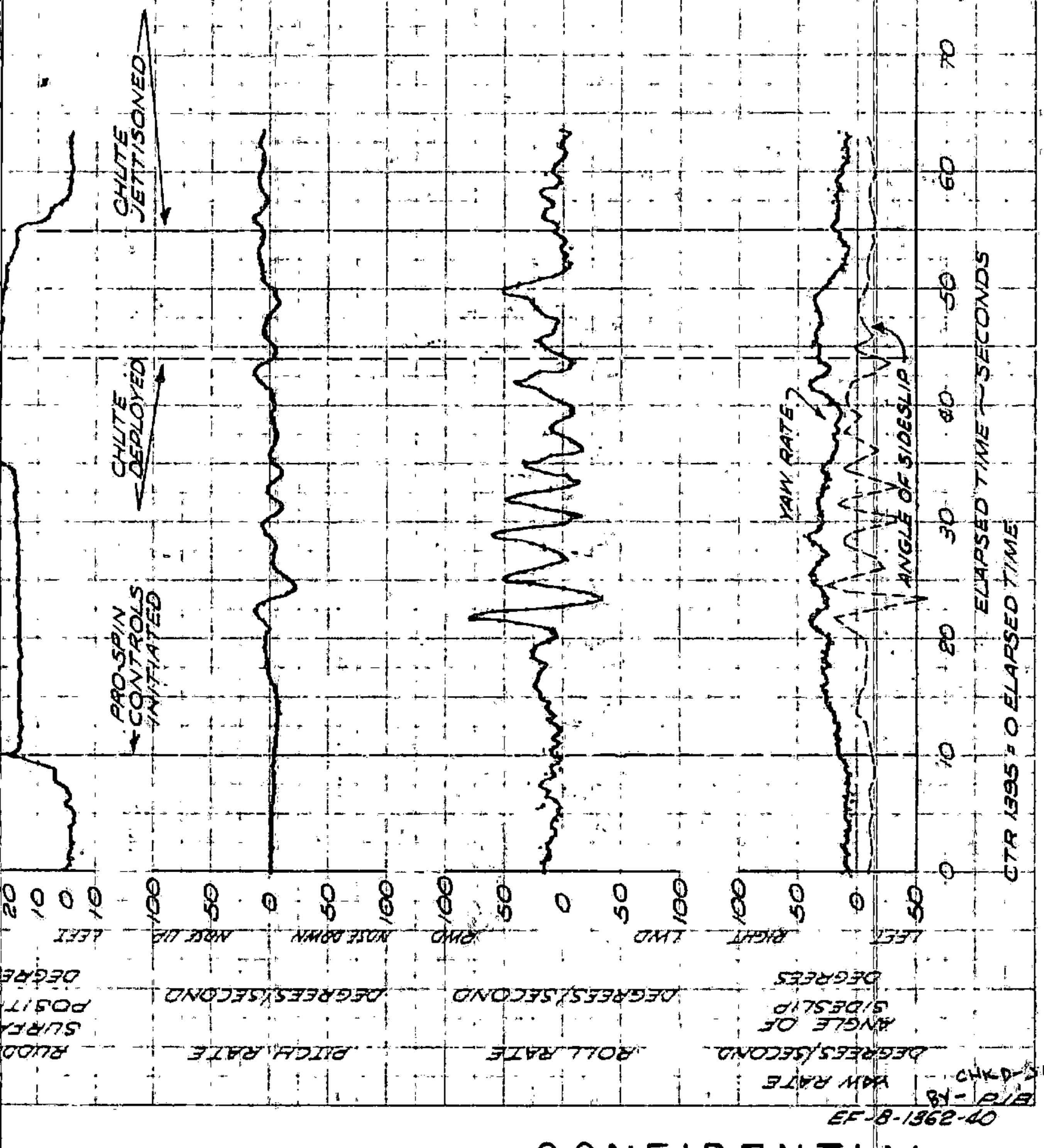
FIGURE 23

SPIN TEST DATA

ARMAMENT EXPENDED, FUEL LOW, AILERON AGAINST FOR ENTRY
(RIGHT RUDDER, LEFT AILERON)

TF-102A S/N 54-1362
FLT 12 14 JUNE 1957

NOTE: SPIN RECOVERY PARACHUTE DEPLOYED. (PRO-SPIN CONTROLS HELD.)



CHKD-SW
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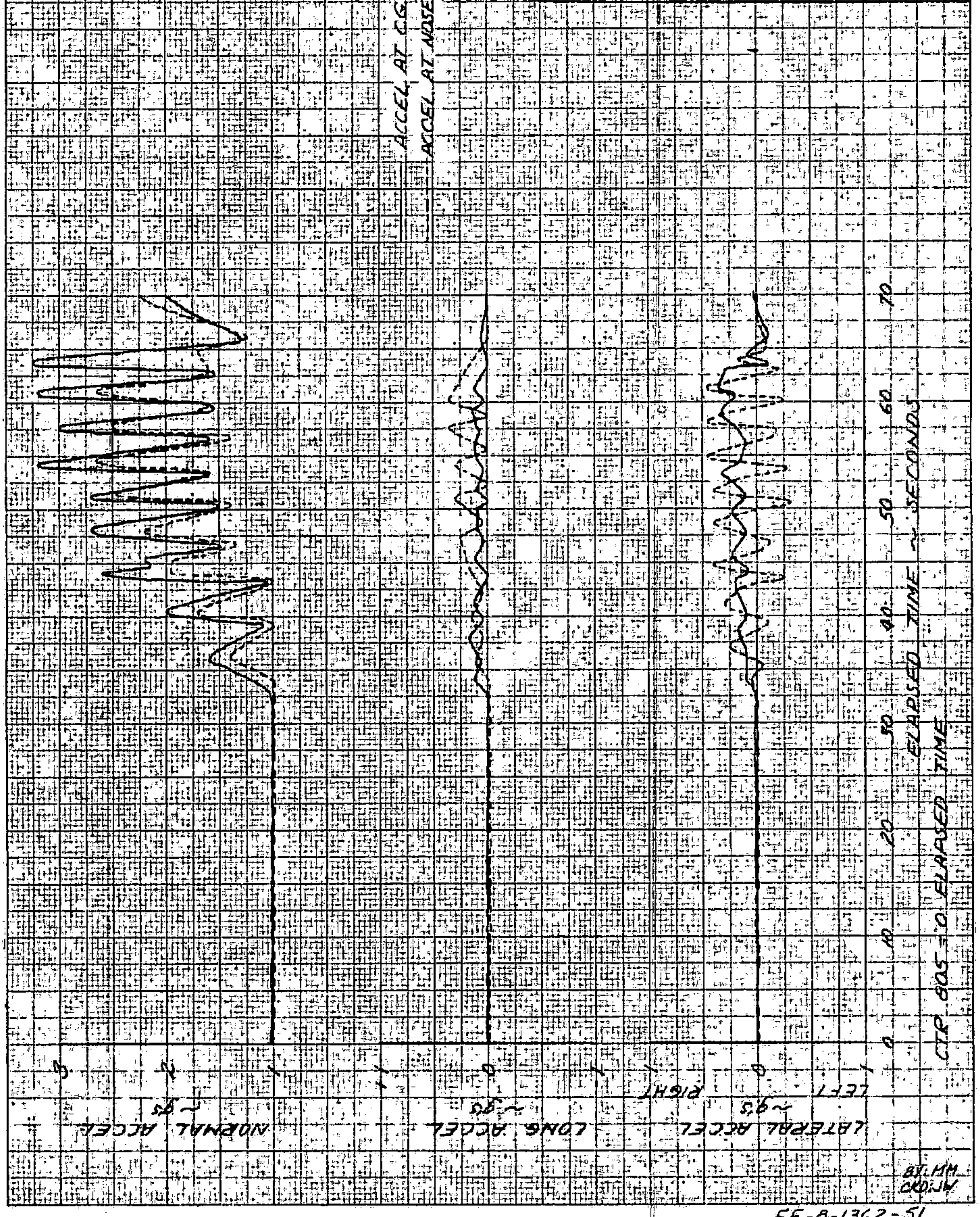
FIGURE 24 PAGE 40

ACCELERATIONS ENCOUNTERED DURING SPIN

TF-102A S/N 54-1362

FLT 8 10 JUNE 1957



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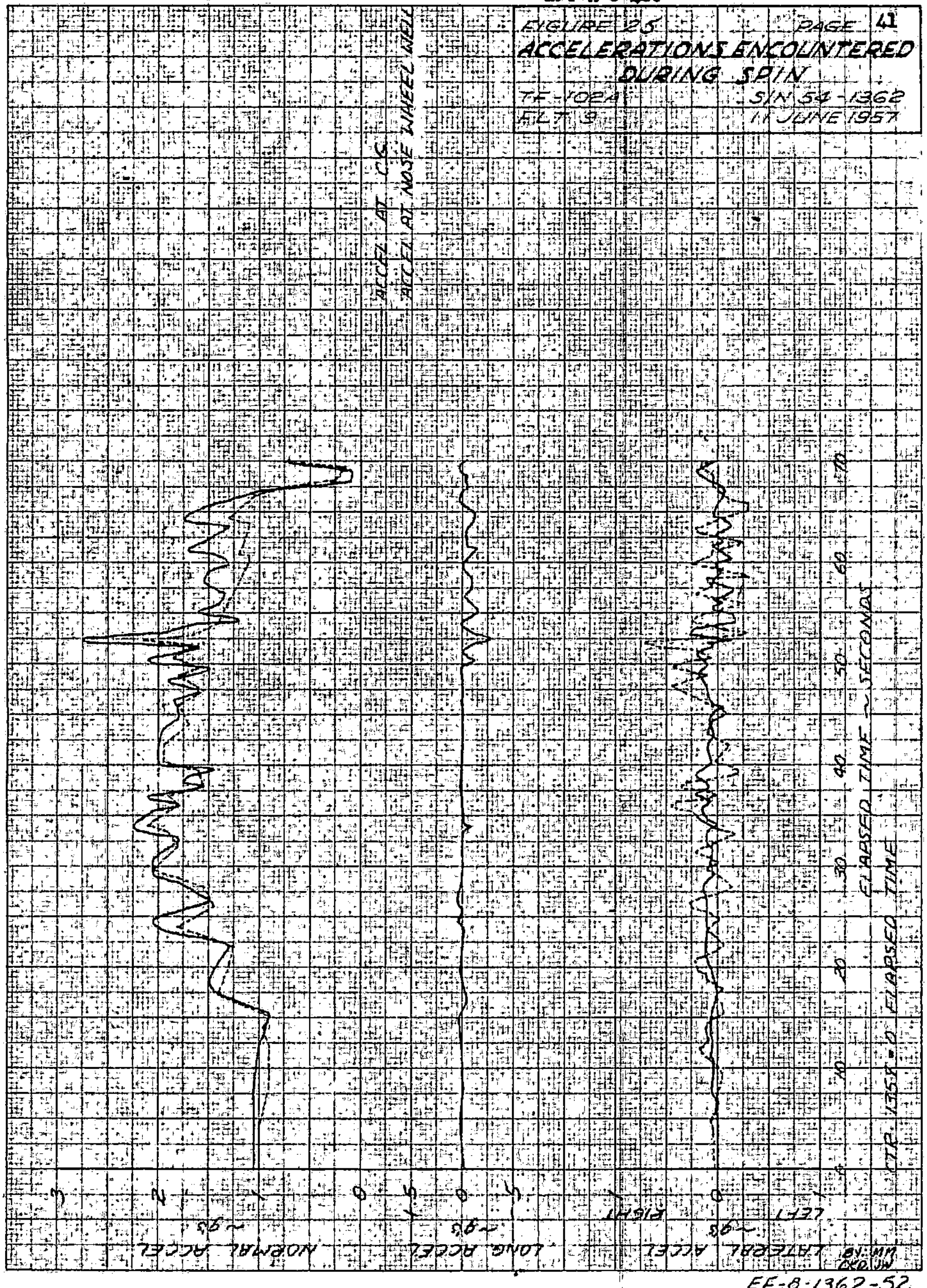
FIGURE 25 PAGE 41

ACCELERATIONS ENCOUNTERED DURING SPIN

TF-102A S/N 54-1362

FLT 9 11 JUNE 1957

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AFFTC-TR-56-23
December 1956
ASTIA DOCUMENT NO. 112426

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APPENDIX III

F-102A PHASE IV PERFORMANCE TEST

JESSE B. HOUSTON Jr.
1st Lt., USAF
Project Engineer

ROBERT D. HIPPERT
Captain, USAF
Project Pilot

CONVAIR - SECURITY LOG
DEPT 11 LOG NO. 1-6511 COPY 38

AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR RESEARCH AND DEVELOPMENT COMMAND
UNITED STATES AIR FORCE

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56-F-1916

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F-102A PHASE IV PERFORMANCE TESTS

APPENDIX III

AFFTC-TR-56-23
January 1957
ASTIA DOCUMENT NO. 112426

JESSE B. HOUSTON Jr.
1st Lt., USAF
Project Engineer

ROBERT D. HIPPERT
Captain, USAF
Project Pilot

NOTE: Due to the limited requirement for the information contained in this Appendix, it has been published in a separate volume. Copies of this Appendix can be obtained by writing to the Armed Services Technical Information Agency (see inside cover).

56-F-1916

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APPENDIX III

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Level Flight Accelerations 18

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Accelerated Stalls 46

Range Mission 52

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Thrust Run 83

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS
POWER MILITARY P_a = 27.690 IN. HG t_a = 9.0 °C
GROSS WEIGHT 27168 LBS. WIND CALM MPH AT — DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	-640	0	36.00	5330	65				
2.00	-610		37.00	5622	78				
4.00	-550		38.00	5922	90				
6.00	-440		39.00	6245	100				
8.00	-290		40.00	6570	112				
10.00	-110		41.00	6900	125				
12.00	+110		42.00	7250	138				
13.00	+240								
14.00	+360								
15.00	515								
16.00	675								
17.00	830								
18.00	1000								
19.00	1185								
20.00	1365								
21.00	1562								
22.00	1770								
23.00	1985								
24.00	2205								
25.00	2430								
25.50	2550								
26.00	2660	↓							
26.75	2860	T.O.							
27.00	2920	—							
28.00	3170	4							
29.00	3430	7							
30.00	3695	10							
31.00	3960	16							
32.00	4225	24							
33.00	4500	34							
34.00	4760	42							
34.75	4980	50							
35.00	5045	53							

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.755 IN. HG t_a = 15.8 °C
GROSS WEIGHT 27176 LBS. WIND 4.5 MPH AT 60 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-10046							
1	-10021							
2	-10004							
3	-9928							
4	-9853							
5	-9771							
6	-9663							
7	-9548							
8	-9415							
9	-9274							
10	-9120							
11	-8932							
12	-8745							
13	-8528							
14	-8316							
15	-8087							
16	-7842							
17	-7574							
18	-7303							
19	-7025							
20	-6728	10						
21	-6406	27						
22	-6095	50						
23	-5777	80						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.645 IN. HG t_a = 12.2 °C
GROSS WEIGHT 27217 LBS. WIND 9 MPH AT 75 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	-560	0	34.25	5220	50				
2.00	-500		35.00	5445	54				
4.00	-400		36.00	5760	62				
6.00	-260		37.00	6060	73				
8.00	-95		38.00	6400	83				
9.00	0		39.00	6740	92				
10.00	120		40.00	7105	100				
11.00	240								
12.00	365								
13.00	500								
14.00	645								
15.00	800								
16.00	960								
17.00	1120								
18.00	1300								
19.00	1500								
20.00	1690								
21.00	1900								
22.00	2100								
23.00	2318								
24.00	2540								
25.00	2785								
25.50	2900								
26.00	3025								
27.00	3265	↓							
27.25	3340	T.0							
28.00	3520	3							
29.00	3780	1.0							
30.00	4045	12							
31.00	4315	20							
32.00	4590	29							
33.00	4865	35							
34.00	5150	45							

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TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.640 IN. HG t_a = 15.5 °C
GROSS WEIGHT 27156 LBS. WIND 2.3 MPH AT 125 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	-10040	0	32.00	-4439	4				
1.00	-10002		33.00	-4139	14				
2.00	-9945		34.00	-3837	27				
3.00	-9886		35.00	-3533	38				
4.00	-9825		36.00	-3219	53				
5.00	-9743		37.00	-2925	66				
6.00	-9658		38.00	-2604	82				
7.00	-9565								
8.00	-9457								
9.00	-9343								
10.00	-9219								
11.00	-9095								
12.00	-8958								
13.00	-8800								
14.00	-8638								
15.00	-8475								
16.00	-8305								
17.00	-8123								
18.00	-7933								
19.00	-7731								
20.00	-7525								
21.00	-7292								
22.00	-7069								
23.00	-6840								
24.00	-6600								
25.00	-6358								
26.00	-6102								
27.00	-5848								
28.00	-5490								
29.00	-5307								
30.00	-5024								
31.00	-4734	↓							
31.50	-4600	T.O.							

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.695 IN. HG t_a = 24.3 °C
GROSS WEIGHT 27140 LBS. WIND 7.3 MPH AT 40 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-10133	0	32.00	-5337	1.6			
1.00	-10106		33.00	-5057	6.3			
2.00	-10084		33.50	-4930	8.8			
3.00	-10053		34.00	-4787	9.3			
4.00	-10018		34.50	-4662	9.7			
5.00	-9956		35.00	-4515	9.5			
6.00	-9892		35.50	-4386	8.6			
7.00	-9821		36.00	-4238	8.4			
8.00	-9745		36.50	-4089	9.6			
9.00	-9660		37.00	-3960	11.2			
10.00	-9561		37.50	-3808	14			
11.00	-9463		38.00	-3676	16			
12.00	-9336		38.50	-3525	19			
13.00	-9219		39.00	-3369	23			
14.00	-9093		39.50	-3232	26			
15.00	-8919		40.00	-3079	31			
16.00	-8770		41.00	-2804	39			
17.00	-8569		42.00	-2425	50			
18.00	-8410		43.00	-2118	59			
19.00	-8239		44.00	-1805	67			
20.00	-8048							
21.00	-7857							
22.00	-7666							
23.00	-7468							
24.00	-7254							
25.00	-7039							
26.00	-6819							
27.00	-6587							
28.00	-6337							
29.00	-6092							
30.00	-5846							
31.00	-5596							
31.81	-5372	T.O.						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY Pa = 27.450 IN. HG ta = 15.5 °C
GROSS WEIGHT 27168 LBS. WIND 20 MPH AT 15 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	4309	0	32.75	-354	T.O.			
1.00	4305		33.00	-404	1.5			
2.00	4293		34.00	-679	8			
3.00	4273		34.50	-819	12			
4.00	4243		35.00	-941	15			
5.00	4202		36.00	-1228	21			
6.00	4153		36.50	-1372	24			
7.00	4093		37.00	-1499	26			
8.00	4027		38.00	-1783	34			
9.00	3948		38.50	-1929	39			
10.00	3860		39.00	-2077	44			
11.00	3767		40.00	-2364	52			
12.00	3660		41.00	-2609	59			
13.00	3552		42.00	-2980	68			
14.00	3431		43.00	-3284	75			
15.00	3296		44.00	-3614	82			
16.00	3163		45.00	-3951	91			
17.00	3013							
18.00	2862							
19.00	2698							
20.00	2532							
21.00	2302							
22.00	2119							
23.00	1916							
24.00	1721							
25.00	1505							
26.00	1285							
27.00	1071							
28.00	833							
29.00	606							
30.00	354							
31.00	113							
32.00	-149							

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY Pa = 27.620 IN. HG ta = 13.9 °C
GROSS WEIGHT 27187 LBS. WIND CALM MPH AT ____ DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	4150	0	33	-556	2			
1	4149		34	-832	3			
2	4141		35	-1111	6			
3	4126		36	-1378	9			
4	4100		37	-1668	17			
5	4063		38	-1944	22			
6	4017		39	-2240	28			
7	3965		40	-2518	33			
8	3898		41	-2821	40			
9	3821		42	-3134	49			
10	3736		43	-3424	58			
11	3639		44	-3744	69			
12	3543		45	-4069	80			
13	3427							
14	3308							
15	3175							
16	3039							
17	2885							
18	2727							
19	2568							
20	2391							
21	2206							
22	2025							
23	1821							
24	1612							
25	1407							
26	1181							
27	960							
28	718							
29	471							
30	232							
31	-33	T.O						
32	-302	0.5						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.530 IN. HG t_a = 17.8 °C
GROSS WEIGHT 27147 LBS. WIND 11.5 MPH AT 7.5 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	-10141	0	34	-4676	7				
2	-10094		35	-4392	12				
4	-10010		36	-4086	17				
5	-9954		37	-3798	21				
6	-9883		38	-3503	28				
7	-9808		39	-3183	36				
8	-9732		40	-2881	46				
9	-9635		41	-2568	58				
10	-9532								
11	-9921								
12	-9304								
13	-9171								
14	-9042								
15	-8900								
16	-8747								
17	-8577								
18	-8412								
19	-8224								
20	-8039								
21	-7842								
22	-7649								
23	-7446								
24	-7211								
25	-6993								
26	-6761								
27	-6516								
28	-6278								
29	-6029								
30	-5755								
31	-5496								
32	-5228								
32.14	-5027	T.O.							
33.10	-4954	2							

CONFIDENTIAL

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MILITARY P_a = 27.665 IN. HG t_a = 18.3 °C
GROSS WEIGHT 27155 LBS. WIND CALM MPH AT — DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-10121	0	34	-4316	9			
2	-10058		35	-4015	15			
4	-9969		36	-3728	21			
5	-9909		37	-3422	29			
6	-9831		38	-3112	38			
7	-9750		39	-2795	48			
8	-9661		40	-2496	59			
9	-9561		41	-2170	71			
10	-9452							
11	-9324							
12	-9204							
13	-9071							
14	-8919							
15	-8764							
16	-8598							
17	-8426							
18	-8244							
19	-8062							
20	-7859							
21	-7653							
22	-7437							
23	-7208							
24	-6994							
25	-6752							
26	-6502							
27	-6249							
28	-6001							
29	-5728							
29.63	-5559	T.O.						
30	-5456	0.5						
31	-5171	3						
32	-4903	5						
33	-4611	7						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM P_a = 27.695 IN. HG t_a = 13.0 °C
GROSS WEIGHT 27167 LBS. WIND 11 MPH AT +75 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	6870	0						
4.00	6770							
7.00	6540							
8.31	6400							
9.87	6200							
11.00	6030							
12.00	5860							
13.00	5670							
14.00	5470							
15.00	5270							
16.50	4940							
17.50	4720							
17.87	4630	T.O.						
18.50	4450	5						
19.00	4340	8						
19.50	4200	13						
20.00	4060	19						
20.50	3920	26						
21.00	3790	33						
21.81	3560	50						
22.50	3470	70						
23.50	3080	100						
24.00	2930	120						
24.50	2770	140						
24.69	2700	150						
25.00	2630	158						
25.69	2430	187						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM P_a = 27.695 IN. HG t_a = 14.5 °C
GROSS WEIGHT 27169 LBS. WIND CALM MPH AT — DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-840	0						
4.00	-750							
6.12	-600							
7.25	-500							
8.50	-350							
9.50	-250							
10.50	-100							
11.50	+50							
12.50	+210							
13.50	400							
14.50	800							
16.00	900							
17.00	1140							
17.50	1260							
18.00	1370							
18.50	1490							
19.00	1620	T.O.						
19.50	1750	5						
20.00	1890	7						
20.50	2020	10						
21.00	2160	15						
21.50	2300	19						
22.00	2440	25						
22.50	2560	32						
23.00	2725	42						
23.56	2890	50						
24.00	3025	60						
25.00	3320	84						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM P_a = 27.700 IN. HG t_a = 8.0 °C
GROSS WEIGHT 27178 LBS. WIND 13 MPH AT + 82 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	6840	0						
4.00	6770							
6.50	6620							
8.50	6420							
9.50	6300							
10.50	6160							
11.50	6000							
12.50	5850							
13.50	5640							
14.50	5470							
15.50	5260							
16.50	5050							
17.50	4820							
17.93	4720	T.O.						
18.50	4580	4						
19.50	4330	8						
20.50	4060	17						
21.50	3800	28						
22.50	3510	45						
22.75	3440	50						
23.50	3230	67						
24.50	2930	94						
24.75	2860	100						
25.50	2630	123						
26.50	2320	152						
27.50	2010	185						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM P_a = 27.830 IN. HG t_a = 6.0 °C
GROSS WEIGHT 27166 LBS. WIND CALM MPH AT — DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-580	0						
4.00	-450							
6.50	-250							
9.00	+50							
10.00	200							
11.00	350							
12.00	540							
13.50	830							
15.00	1130							
16.00	1370							
17.00	1620							
17.19	1680	T.O.						
18.00	1880	8						
19.00	2140	17						
20.00	2420	27						
21.00	2710	40						
21.50	2850	50						
22.50	3160	73						
23.56	3520	100						
24.50	3830	128						
25.69	4050	150						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM P_a = 27.660 IN. HG t_a = 12.7 °C
GROSS WEIGHT 27,192 LBS. WIND 5.8 MPH AT 30 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-570	0						
2.00	-560	↓						
4.00	-505							
6.00	-400							
8.00	-220							
9.00	-120							
10.00	0							
11.00	+145							
12.00	332							
13.00	512							
14.00	700							
15.00	895							
16.00	1105							
17.00	1330							
18.00	1580							
19.00	1830							
19.25	1900	N.W.O.						
20.00	2100	T.O.						
21.00	2380	17						
22.00	2660	37						
22.50	2815	50						
23.00	2960	63						
24.00	3260	97						
25.00	3577	130						
25.50	3745	150						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM Pa = 27.66 0 IN. HG ta = 15 °C
GROSS WEIGHT 27164 LBS. WIND 15 MPH AT 15 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-600	0						
2.00	-550							
4.00	-450							
6.00	-280							
8.00	-60							
10.00	+220							
12.00	570							
13.00	745							
14.00	950							
15.00	1170							
16.00	1395							
16.75	1580							
17.00	1640							
17.50	1760	T.O.						
18.00	1900	2						
19.00	2150	13						
20.00	2425	25						
21.00	2720	37						
21.75	2940	50						
22.00	3025	53						
23.00	3325	70						
24.00	3655	93						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM Pa = 27.725 IN. HG ta = 13.0 °C
GROSS WEIGHT 27240 LBS. WIND CALM MPH AT — DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-580	0						
2.00	-550							
4.00	-460							
6.00	-315							
8.00	-110							
10.00	+150							
12.00	460							
14.00	840							
16.00	1260							
18.00	1750							
19.00	2022							
20.00	2295							
21.00	2580							
21.50	2738	T.O.						
22.00	2885	4						
23.00	3210	17						
24.00	3520	30						
25.00	3860	45						
25.25	3945	50						
26.00	4200	62						
27.00	4540	87						
28.00	4880	117						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TAKE-OFFS

POWER MAXIMUM P_a = 27.710 IN. HG t_a = 13.0 °C
GROSS WEIGHT 27226 LBS. WIND 5 MPH AT 37.5 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-600	0						
2.00	-560							
4.00	-480							
6.00	-325							
8.00	-120							
10.00	+135							
12.00	405							
14.00	780							
16.00	1200							
18.00	1680							
20.00	2205							
20.50	2360							
21.00	2475	T.O.						
22.00	2800	10						
23.00	3100	25						
24.00	3420	48						
24.12	3462	50						
25.00	3755	77						
26.00	4090	113						
27.00	4420	153						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (AFTERBURNING)
AVERAGE WEIGHT 25,800 LBS. t_a = −5° C

TIME SECONDS	IAS KNOTS	ALTITUDE FEET	N_1 RPM	N_2 RPM	Pt_7 IN. HG			
112.0	241.5	9910	5892	9351	56.29			
113.0	247.0	9890	5852	9351	56.53			
114.0	252.0	9890	5842	9360	56.82			
115.0	257.0	9870	5842	9360	57.16			
116.0	262.5	9870	5842	9370	57.45			
117.0	268.5	9870	5842	9379	57.74			
118.0	273.5	9870	5842	9379	58.09			
119.0	279.0	9870	5842	9379	58.19			
120.0	284.5	9870	5842	9389	58.57			
121.0	290.0	9870	5842	9389	58.81			
122.0	294.5	9870	5842	9389	59.10			
123.0	300.0	9880	5842	9398	59.25			
124.0	305.0	9890	5842	9398	59.54			
124.9	310.0	9890	5842	9398	59.68			
125.9	316.0	9900	5842	9398	60.08			
126.9	321.5	9910	5842	9398	60.28			
127.9	327.0	9920	5842	9398	60.64			
128.9	332.0	9930	5842	9398	60.99			
129.9	337.5	9930	5842	9398	61.15			
130.9	343.0	9930	5842	9398	61.50			
131.9	349.0	9930	5832	9398	61.86			
132.9	354.5	9930	5823	9398	62.22			
133.9	359.5	9930	5832	9408	62.42			
134.9	365.5	9920	5823	9408	62.88			
135.9	370.5	9920	5823	9408	63.14			
136.9	375.5	9920	5813	9408	63.34			
137.9	381.0	9920	5804	9408	63.74			
138.9	386.5	9930	5804	9408	63.95			
139.9	392.0	9930	5804	9408	64.43			
140.9	396.5	9930	5794	9413	64.62			
141.9	402.0	9950	5794	9418	64.86			
142.9	407.5	9970	5794	9418	65.10			
143.9	412.0	9990	5794	9418	65.58			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (AFTERBURNING)
AVERAGE WEIGHT 25,800 LBS. t_a = −5 °C

TIME SECONDS	IAS KNOTS	ALTITUDE FEET	N_1 RPM	N_2 RPM	P_{T7} IN. HG
144.9	418.0	10000	5794	9428	65.87
145.9	421.5	10030	5794	9428	66.25
146.9	427.5	10030	5793	9438	66.63
147.9	437.0	10030	5793	9447	66.82
149.9	442.0	10030	5793	9447	67.50
150.9	447.5	10030	5783	9447	67.78
151.9	453.0	10030	5783	9447	68.18
152.9	459.0	10030	5764	9447	68.58
153.9	464.5	10030	5754	9447	68.94
154.9	469.5	10030	5744	9447	69.29
155.9	475.0	10030	5744	9447	69.90
156.9	480.5	10030	5744	9447	70.20
157.9	485.0	10030	5744	9447	70.50
158.8	489.5	10030	5744	9452	70.91
159.8	494.5	10030	5744	9457	71.27
160.8	499.5	10030	5734	9457	71.62
161.8	503.5	10030	5734	9467	71.92
162.8	508.5	10030	5744	9467	72.71
163.8	511.5	10020	5744	9467	73.20
164.8	516.5	10010	5734	9467	73.49
165.8	520.5	9980	5725	9472	73.84
166.8	523.5	9940	5715	9477	74.17
168.8	528.0	9890	5705	9486	74.40
170.8	530.5	9870	5705	9495	74.99
176.8	534.5	9930	5695	9495	74.85

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (AFTERBURNING)
AVERAGE WEIGHT 24,300 LBS. t_a = -50°C.

TIME SECONDS	IAS KNOTS -	ALTITUDE FEET	N_1 RPM	N_2 RPM	P_{T7} IN. HG			
230.2	235.0	29970	6090	9304	24.56			
232.7	240.0	29950	5990	9314	31.60			
235.2	247.0	29890	5980	9314	32.10			
237.2	253.0	29840	5970	9323	32.60			
237.6	254.0	29840	5970	9323	32.60			
245.6	275.5	29740	5980	9332	34.10			
246.6	279.0	29720	5980	9332	34.30			
248.1	283.0	29720	5980	9332	34.50			
250.1	288.5	29690	5970	9342	34.84			
252.1	294.5	29680	5970	9342	35.30			
253.6	299.5	29660	5970	9342	35.55			
255.1	304.0	29650	5960	9350	35.80			
256.6	308.5	29640	5960	9350	36.10			
258.6	314.0	29630	5960	9350	36.44			
260.1	319.0	29630	5960	9350	36.73			
262.1	324.5	29620	5960	9350	37.06			
263.1	327.5	29620	5960	9350	37.25			
265.6	334.0	29620	5950	9360	37.74			
267.1	339.0	29620	5950	9360	38.02			
268.1	342.0	29620	5940	9370	38.22			
276.5	364.5	29510	5940	9370	39.88			
279.5	369.5	29390	5920	9379	40.58			
283.0	373.5	29230	5920	9379	40.89			
289.0	378.0	29050	5920	9389	41.30			
291.0	383.5	29930	5920	9389	41.30			
291.5	388.5	30030	5920	9389	41.35			
293.0	394.0	30380	5920	9389	41.55			
298.0	399.0	30480	5920	9389	41.95			
305.5	409.0	30340	5910	9389	42.66			
310.5	415.0	30200	5900	9410	43.32			
315.9	420.5	30110	5890	9410	43.82			
322.9	425.5	30060	5890	9410	44.22			
327.9	429.0	30020	5890	9410	44.52			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (AFTERBURNING)
AVERAGE WEIGHT 23,100 LBS. t_a = -72 °C

TIME SECONDS	IAS KNOTS	ALTITUDE FEET	N_1 RPM	N_2 RPM	P_{T7} IN. HG
291.0	214.5	39410	5990	9220	23.09
296.0	216.5	39430	6130	9220	20.66
301.5	226.0	39390	5970	9210	23.88
304.0	230.0	39370	5950	9210	24.16
306.4	235.0	39370	5940	9210	24.36
308.9	240.0	39350	5950	9220	24.60
311.9	245.0	39330	5940	9220	24.90
314.4	250.0	39330	5940	9220	25.14
316.9	255.0	39330	5940	9220	25.43
319.4	259.5	39330	5940	9210	27.67
324.0	267.5	39330	5940	9230	26.11
324.9	269.5	39330	5930	9230	26.59
332.9	285.0	39310	5930	9235	27.18
334.9	289.0	39270	5910	9235	27.46
337.3	294.0	39200	5910	9235	27.81
340.8	299.0	39010	5910	9245	28.30
344.8	304.0	38680	5910	9255	28.79
348.3	307.5	38420	5900	9255	29.19
348.8	313.5	39230	5900	9255	29.24
349.8	319.0	39520	5900	9255	29.43
352.8	325.0	39650	5900	9255	29.63
356.8	329.0	39650	5890	9255	30.12
363.8	335.5	39650	5890	9265	30.42
366.7	339.0	39650	5890	9275	30.71
371.7	344.0	39640	5890	9275	31.11
377.2	349.0	39590	5890	9275	31.50
383.2	354.0	39570	5890	9285	31.85
385.7	356.0	39570	5880	9285	31.90

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (AFTERBURNING)
AVERAGE WEIGHT 23,550 LBS. t_a = -62 °C

TIME SECONDS	IAS KNOTS	ALTITUDE FEET	N1 RPM	N2 RPM	PT7 IN. HG
0.0	239.5	44690	6150	9370	19.30
4.5	245.0	44645	6090	9360	19.93
9.6	247.5	44625	6090	9370	19.98
15.5	253.0	44615	6090	9370	20.16
22.4	259.5	44565	6090	9380	20.61
27.7	264.5	44375	6070	9385	20.99
31.9	269.5	44035	6070	9385	21.49
35.9	277.0	44165	6060	9395	21.82
38.3	284.0	44915	6050	9395	21.97
43.8	288.5	44975	6060	9395	22.32
52.3	297.0	44825	6050	9395	23.05
62.2	303.0	44525	6050	9395	23.60
67.2	305.0	44385	6040	9395	23.88

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (AFTERBURNING)
AVERAGE WEIGHT 23,000 LBS. t_a = −63 °C

TIME SECONDS	IAS KNOTS	ALTITUDE FEET	N_1 RPM	N_2 RPM	P_{T7} IN. HG
0.0	214.5	49990	6195	9350	15.30
8.5	219.5	50020	6275	9350	15.45
18.0	225.0	50050	6150	9350	15.70
28.1	232.0	49810	6140	9350	16.15
33.2	236.0	49430	6130	9350	16.48
43.3	240.0	49035	6110	9350	17.01
58.2	246.0	48260	6090	9350	17.54
61.2	252.0	48800	6090	9350	17.59
62.7	255.5	49105	6090	9360	17.69
66.1	259.5	49270	6090	9360	17.89
77.7	264.5	49190	6090	9370	18.22
88.1	273.0	49000	5815	9160	18.22

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (NON-AFTERBURNING)
AVERAGE WEIGHT 25,200 LBS. t_a = −3 °C

TIME SECONDS	IAS KNOTS	ALTITUDE FEET	N_1 RPM	N_2 RPM	P_{T7} IN. HG			
168.4	229.5	10090	5695	9020	49.00			
170.4	235.0	10050	5890	9285	53.00			
171.9	239.5	10030	5910	9320	53.55			
173.9	245.0	10010	5910	9350	54.20			
180.3	262.0	9970	5910	9390	55.60			
187.3	279.0	9980	5910	9400	56.63			
190.3	285.0	9990	5900	9400	57.06			
192.3	289.0	10000	5900	9400	57.11			
194.3	294.0	10010	5900	9400	57.45			
196.3	299.0	10030	5890	9400	57.65			
198.8	304.0	10040	5910	9420	57.79			
200.8	308.5	10070	5890	9410	58.19			
202.8	314.0	10050	5890	9420	58.44			
204.8	319.0	10040	5890	9420	58.86			
205.3	320.5	10040	5890	9420	58.81			
210.3	333.0	10010	5890	9420	59.30			
217.8	349.5	10000	5880	9430	60.18			
220.2	354.5	10010	5880	9450	60.38			
222.7	359.0	10010	5870	9435	60.70			
224.7	363.5	10010	5870	9435	61.00			
227.2	368.0	10010	5870	9435	61.15			
229.2	373.0	10010	5860	9435	61.45			
231.7	378.0	10010	5860	9445	61.70			
234.7	384.5	10010	5850	9445	62.02			
235.2	385.5	10010	5850	9445	62.02			
240.2	395.5	10010	5840	9445	62.63			
247.7	408.5	10010	5840	9445	63.29			
250.2	412.0	10010	5840	9445	63.55			
252.7	417.0	10000	5840	9445	63.85			
255.2	420.5	9990	5830	9455	64.18			
258.2	425.0	9970	5830	9455	64.42			
261.7	430.0	9960	5830	9455	64.75			
265.2	434.5	9960	5820	9465	65.05			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (NON-AFTERBURNING)
AVERAGE WEIGHT 25,200 LBS, t_a = −3 °C

TIME SECONDS	IAS KNOTS	ALTITUDE FEET	N_1 RPM	N_2 RPM	P_{T7} IN. HG			
270.2	441.0	9970	5820	9465	65.40			
277.6	449.0	9970	5810	9475	65.96			
281.6	455.5	9960	5810	9475	66.53			
286.6	461.5	9930	5800	9485	66.92			
289.6	462.5	9910	5675	9375	64.61			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (NON-AFTERBURNING)
AVERAGE WEIGHT 23,600 LBS. t_a = -48°C

TIME SECONDS	IAS KNOTS	ALTITUDE FEET	N_1 RPM	N_2 RPM	P_{T7} IN. HG
57.5	227.5	29950	5260	8455	24.36
65.5	235.5	29870	6020	9285	31.26
69.4	240.0	29830	6015	9305	31.85
73.5	245.0	29790	6015	9305	32.40
77.9	250.5	29750	6010	9315	32.70
81.4	254.0	29730	6010	9315	33.00
86.4	259.5	29710	6010	9320	33.30
94.4	267.0	29710	6000	9320	33.75
99.3	271.5	29730	5900	9330	34.00
104.3	276.0	29760	5900	9330	34.20
108.8	281.0	29780	5900	9330	34.40
114.3	286.5	29790	5980	9330	34.75
118.3	290.5	29800	5980	9330	35.05
125.3	297.5	29800	5970	9330	35.40
130.8	302.5	29790	5970	9340	35.65
137.3	308.0	29800	5960	9340	35.95
142.3	312.5	29820	5960	9340	36.20
147.2	318.0	29820	5960	9350	36.57
155.2	326.5	29780	5940	9350	37.05
159.2	331.0	29760	5940	9350	37.40
164.6	336.0	29690	5940	9350	37.74
167.6	339.0	29660	5940	9350	37.89
172.2	343.5	29610	5940	9350	38.32
175.7	347.0	29560	5940	9350	38.52
183.7	354.5	29420	5930	9360	39.10
189.1	359.0	29310	5920	9370	39.49
196.6	361.5	29140	5120	8615	30.27

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT ACCELERATIONS (NON-AFTERBURNING)
AVERAGE WEIGHT 22,650 LBS. t_a = — 70 ° C

TIME SECONDS	IAS - KNOTS	ALTITUDE FEET	N_1 RPM	N_2 RPM	P_{T7} IN. HG
32.9	236.0	39790	5982	9180	23.50
37.9	240.0	39740	5972	9190	23.85
45.9	245.0	39690	5982	9200	24.30
54.9	250.5	39650	5972	9200	24.55
64.3	254.5	39630	5962	9210	24.85
73.3	259.5	39630	5942	9210	25.20
83.3	263.5	39630	5942	9210	25.40
97.1	268.5	39700	5942	9210	25.70
107.1	273.5	39690	5932	9210	25.90
113.1	275.0	39670	5932	9210	26.05
123.1	279.0	39665	5803	9120	25.50

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: CHECK CLIMBS (MILITARY POWER)
W_{ES} 27,312 LBS.

TIME Min: Sec.	IAS Knots	ALTITUDE Feet	P_{t7} In. HG	N_1 RPM	N_2 RPM	F.A.T. DEG. C	T_{t7} DEG. C	F.U. Eng. LBS
0:55.5	448.0	3960	78.27	5950	9610	36	570	295
1:04.5	443.0	5120	76.17	5950	9620	35	565	320
1:09.5	417.5	6080	72.70	6000	9620	35	565	334
1:14.5	404.5	7000	70.20	6000	9620	32	575	347
1:20.5	385.5	7960	67.05	6000	9620	30	580	362
1:29.4	380.0	8940	64.90	6000	9620	29	580	385
1:41.4	370.0	9950	62.47	6000	9630	27	590	414
1:57.3	373.0	10970	60.90	6000	9630	26	590	450
2:09.2	369.0	12010	59.44	6000	9620	25	590	478
2:22.2	367.0	12980	58.17	6000	9620	22	590	507
2:35.1	366.0	13980	56.78	6000	9620	20	590	536
2:47.1	362.0	14960	55.10	6010	9620	19	590	562
3:00.0	359.5	15930	53.90	6010	9610	16	590	589
3:14.0	354.5	16950	52.50	6020	9610	14	600	618
3:25.9	350.5	17920	51.20	6030	9600	11	600	643
3:39.8	349.0	18970	50.00	6030	9600	9	600	670
3:53.7	345.0	19960	48.70	6050	9600	6	600	697
4:08.6	344.5	20920	47.50	6050	9600	4	600	726
4:22.6	340.0	21900	46.07	6050	9590	2	600	753
4:36.5	334.5	22880	44.72	6050	9590	0	600	777
4:48.4	329.0	23930	43.32	6050	9590	−2	600	798
5:00.3	325.0	24910	41.80	6050	9570	−5	600	819
5:13.2	317.5	25910	40.48	6050	9550	−7	600	841
5:28.2	312.5	26890	39.30	6050	9550	−10	600	866
5:45.1	309.0	27860	38.09	6050	9520	−12	600	892
6:02.0	305.0	28850	36.86	6050	9510	−16	600	919
6:18.9	298.0	29850	35.60	6050	9500	−17	600	944
6:41.8	292.5	30840	34.25	6050	9500	−21	600	978
7:08.6	294.5	31820	33.50	6050	9500	−22	600	1016
7:30.5	295.5	32800	32.70	6050	9490	−23	600	1046
7:48.4	288.0	33720	31.50	6050	9460	−26	605	1069
8:19.2	286.5	34760	30.52	6050	9450	−28	605	1111
8:49.1	280.0	35720	29.24	6050	9440	−29	605	1142

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: CHECK CLIMBS (MILITARY POWER)
W_{ES} 27,312 LBS.

TIME Min: Sec	IAS Knots	ALTITUDE Feet	P_{t7} In. HG	N_1 RPM	N_2 RPM	F.A.T. DEG. C	T_{t7} DEG. C	F.U. Eng. LBS.	
9:13.0	276.5	36720	28.20	6050	9420	-30	610	1178	
9:34.8	270.5	37740	27.08	6050	9410	-31	610	1203	
10:12.6	264.0	38670	25.72	6050	9400	-32	620	1246	
10:54.3	263.0	39680	24.94	6050	9400	-32	620	1292	
11:43.0	257.5	40700	23.77	6050	9390	-31	620	1344	
12:39.6	254.0	41680	22.42	6040	9390	-28	620	1401	
13:09.9	243.5	42720	21.14	6050	9390	-28	630	1430	
15:09.8	236.5	43720	20.18	6050	9390	-28	630	1538	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: CHECK CLIMBS (MILITARY POWER)
W_{ES} 27312 LBS.

TIME Min: Sec.	IAS Knots	ALTITUDE Feet	Pt7 In. HG	N1 RPM	N2 RPM	F.A.T. DEG. C	Tt7 DEG. C	F.U. Eng LBS.
2:08.5	394.0	5005	69.92	5990	9660	37	570	306
2:20.5	396.0	5995	69.60	5990	9660	36	570	339
2:31.4	386.0	6975	67.63	6030	9650	34	570	364
2:44.4	383.5	7975	66.11	6030	9660	31	580	397
2:56.4	387.5	8940	64.80	6010	9660	30	580	426
3:06.4	382.0	9940	63.02	6040	9660	29	580	450
3:16.4	375.0	10915	61.20	6040	9650	26	580	473
3:27.3	364.0	11900	59.10	6050	9650	24	580	499
3:42.3	362.0	12890	57.79	6050	9650	22	590	532
3:58.3	362.5	13890	56.40	6050	9650	20	590	566
4:15.3	365.5	14895	55.35	6050	9650	18	590	603
4:32.2	371.0	15880	54.27	6050	9650	17	590	638
4:45.2	366.0	16940	52.70	6060	9640	16	590	666
4:55.2	356.0	17890	50.95	6070	9640	14	590	686
5:06.1	346.0	18885	49.40	6070	9640	12	590	707
5:19.0	337.0	19910	47.67	6090	9620	9	600	732
5:35.0	334.0	20905	46.31	6090	9610	5	600	762
5:52.0	336.0	21860	45.51	6090	9600	3	600	793
6:06.9	325.0	22885	43.46	6090	9600	2	600	820
6:21.9	329.0	23870	42.50	6090	9600	0	600	845
6:36.8	321.0	24890	41.19	6090	9590	-4	600	871
6:52.8	312.5	25870	39.70	6090	9580	-6	600	897
7:11.7	311.5	26890	38.69	6090	9570	-9	610	927
7:28.6	309.5	27830	37.79	6090	9570	-10	610	954
7:45.6	305.0	28815	36.60	6090	9550	-12	610	980
8:04.5	298.0	29820	35.30	6100	9550	-16	610	1009
8:26.5	293.0	30825	34.05	6100	9530	-19	610	1040
8:49.4	292.0	31810	33.20	6090	9520	-21	610	1073
9:10.3	289.0	32780	32.25	6090	9500	-24	610	1102
9:29.2	282.0	33765	31.12	6090	9500	-27	610	1127
9:51.2	278.0	34745	29.87	6090	9470	-28	610	1155
10:19.1	271.0	35730	28.64	6090	9460	-30	610	1190
10:49.0	269.5	36700	27.65	6090	9450	-32	610	1227

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: CHECK CLIMBS (MILITARY POWER)
W_{ES} 27312 LBS.

TIME Min: Sec.	IAS Knots	ALTITUDE Feet	Pt7 In. HG	N1 RPM	N2 RPM	F.A.T. DEG. C	Tt7 DEG. C	F.U. Eng LBS.
11:18.9	264.5	37695	26.55	6090	9430	-32	610	1262
11.43.8	259.5	38680	25.42	6090	9420	-33	610	1290
12:12.7	248.0	39665	24.07	6090	9400	-34	620	1320
12:45.6	246.0	40675	23.00	6090	9400	-33	620	1354
13:14.5	233.0	41685	21.53	6080	9380	-33	630	1381
14:30.2	228.0	42700	20.45	6080	9380	-32	630	1447
15:33.0	220.0	43460	19.58	6080	9370	-32	630	1502

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: CHECK CLIMBS (MAXIMUM POWER)
W_{ES} 27,312 LBS.

TIME Min: Sec	IAS Knots	ALTITUDE Feet	P_{t7} In. HG	N_1 RPM	N_2 RPM	F.A.T. DEG. C	T_{t7} DEG. C	F.U. Eng LBS.	F.U. A/B LBS.
2:17.7	502.0	5915	79.57	5850	9620	45	560	300	635
2:47.7	515.0	6975	79.12	5840	9630	46	560	318	690
2:28.7	521.0	7825	78.42	5860	9630	47	560	329	720
2:32.7	507.5	8845	75.27	5880	9650	47	560	340	752
2:36.7	501.5	9930	73.05	5890	9650	46	570	349	780
2:40.7	488.5	11045	70.31	5900	9650	45	580	363	808
2:43.7	483.0	11900	68.58	5910	9640	43	580	366	831
2:50.7	464.5	13900	64.42	5940	9630	40	590	383	876
2:54.7	459.5	14985	62.64	5940	9630	37	590	392	900
2:58.6	451.5	16050	60.92	5960	9630	35	590	401	924
3:05.6	439.0	17980	57.74	5970	9620	30	600	416	964
3:12.5	425.0	19880	54.05	5990	9620	26	600	431	1003
3:16.5	416.5	20965	52.35	6000	9610	24	600	439	1024
3:19.5	408.0	21780	50.75	6010	9600	22	600	445	1039
3:23.5	400.0	22845	49.10	6040	9600	20	600	453	1058
3:27.5	392.0	23870	47.50	6040	9600	17	610	460	1077
3:31.5	384.5	24820	46.01	6040	9600	14	610	467	1095
3:35.4	378.0	25770	44.47	6040	9600	11	610	474	1112
3:40.4	371.0	26870	43.02	6050	9590	8	620	484	1133
3:44.4	362.5	27705	41.50	6060	9580	6	620	490	1150
3:49.4	351.0	28900	39.78	6090	9560	3	620	498	1171
3:53.4	345.0	29740	38.49	6090	9560	0	630	505	1186
3:59.4	340.0	30815	37.39	6090	9550	-5	630	514	1208
4:05.4	336.0	31810	36.25	6090	9550	-7	630	524	1231
4:11.4	332.0	32825	35.00	6090	9550	-10	630	532	1255
4:17.3	329.0	33800	33.70	6090	9550	-12	630	541	1280
4:23.3	317.0	34790	32.35	6090	9530	-14	630	549	1302
4:30.3	313.0	35665	31.40	6090	9520	-17	630	558	1327
4:38.3	305.5	36720	29.83	6110	9520	-20	630	569	1355
4:45.3	300.0	37655	28.79	6120	9500	-22	630	578	1377
4:53.3	295.0	38610	27.75	6090	9500	-23	640	588	1402
5:01.3	286.0	39735	26.32	6120	9500	-25	640	598	1428
5:09.2	281.0	40725	24.89	6090	9450	-26	640	606	1452

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: CHECK CLIMBS (MAXIMUM POWER)
W_{ES} 27312 LBS.

TIME Min: Sec	IAS Knots	ALTITUDE Feet	P_{t7} In. HG	N_1 RPM	N_2 RPM	F.A.T. DEG. C	T_{t7} DEG. C	F.U. Eng LBS.	F.U. A/B LBS.
5:18.2	275.0	41705	23.19	6040	9430	-25	640	615	1479
5:26.2	269.5	42660	22.12	6070	9430	-23	640	623	1501
5:35.2	259.5	43790	20.45	6090	9430	-21	640	632	1524
5:46.1	248.0	44730	19.65	6120	9440	-23	650	641	1552
6:03.0	251.0	45755	19.12	6080	9400	-23	640	655	1595
6:21.0	245.0	46720	18.22	6080	9400	-23	640	671	1638
6:35.0	236.0	47775	17.11	6090	9380	-24	650	681	1672
6:56.0	227.0	48790	16.10	6040	9320	-25	650	696	1721
7:25.9	229.5	49820	15.70	6040	9320	-25	640	717	1790
7:46.8	222.5	50810	14.91	6050	9310	-25	640	731	1838
8:07.7	211.0	51800	13.73	6070	9300	-25	650	743	1883
9:01.5	209.5	52835	13.13	6090	9300	-25	650	775	1992
10:08.2	202.5	53855	12.35	6090	9290	-26	650	812	2125

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: CHECK CLIMBS (MAXIMUM POWER)
W_{ES} 27312 LBS.

TIME Min: Sec	IAS Knots	ALTITUDE Feet	P_{t7} In. HG	N_1 RPM	N_2 RPM	F.A.T. DEG. C	T_{t7} DEG. C	F.U. Eng LBS.	F.U. A/B LBS.
0:24.8	465.5	3955	79.62	5910	9650	41.0	560	305	534
0:31.8	484.0	4940	79.67	5890	9650	45.0	560	323	588
0:36.8	492.0	5920	78.42	5900	9650	46.0	560	336	631
0:41.8	490.5	7065	76.52	5910	9650	46.0	560	348	667
0:45.8	487.5	7980	74.76	5940	9650	46.0	570	358	695
0:50.8	488.0	9120	73.15	5940	9650	44.0	575	373	731
0:54.8	488.0	9990	71.87	5950	9670	43.0	580	380	752
0:58.7	478.0	10945	69.50	5970	9660	42.0	580	389	789
1:03.7	476.0	12030	67.83	5970	9660	41.0	580	401	823
1:07.7	471.0	12900	66.11	5990	9660	39.0	585	410	851
1:13.7	466.5	13980	64.38	5990	9660	38.0	590	421	883
1:17.6	466.0	15105	63.08	5990	9660	36.0	595	432	913
1:21.6	460.0	16050	61.60	6000	9660	35.0	595	442	940
1:25.6	451.0	17050	59.68	6010	9650	32.0	600	450	963
1:29.6	442.0	18090	57.74	6030	9650	30.0	600	459	986
1:33.6	437.0	19105	56.30	6040	9650	27.0	600	467	1011
1:36.6	430.0	19940	54.80	6040	9650	25.0	600	473	1027
1:40.6	417.5	21170	52.25	6060	9650	24.0	600	481	1048
1:43.6	406.5	22060	50.10	6090	9630	21.0	600	487	1064
1:47.5	394.0	23235	48.20	6090	9630	19.0	600	495	1084
1:51.5	386.5	24340	46.45	6100	9620	15.0	600	501	1105
1:54.5	379.0	25135	45.12	6110	9620	14.0	605	506	1118
1:58.5	367.0	26240	43.17	6130	9600	10.0	610	513	1135
2:02.5	359.5	27195	41.80	6140	9600	7.0	610	520	1152
2:07.4	353.5	28220	40.59	6140	9600	4.0	615	529	1172
2:12.4	349.0	29190	39.40	6140	9600	1.0	620	536	1193
2:18.4	342.0	30300	37.84	6140	9590	-4.0	620	546	1218
2:24.3	339.0	31205	36.86	6140	9590	-6.0	625	555	1240
2:30.3	337.0	32105	36.08	6140	9580	-8.0	625	564	1265
2:37.3	331.0	33120	34.60	6160	9570	-11.0	625	574	1293
2:44.3	324.0	34335	33.35	6160	9570	-13.0	630	584	1319
2:50.2	318.0	35035	32.15	6180	9560	-15.0	630	592	1338
2:57.2	311.5	36015	30.97	6170	9550	-16.0	630	601	1360

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: CHECK CLIMBS (MAXIMUM POWER)
W_{ES} 27312 LBS.

TIME Min: Sec	IAS Knots	ALTITUDE Feet	P_{t7} In. HG	N_1 RPM	N_2 RPM	F.A.T. DEG. C	T_{t7} DEG. C	F.U. Eng LBS	F.U. A/B LBS.
3:04.2	305.0	37020	29.63	6170	9540	-19	635	610	1385
3:11.2	298.0	38000	28.44	6180	9520	-21	640	618	1407
3:19.1	291.5	39030	27.08	6180	9510	-23	645	627	1432
3:27.1	287.0	40020	25.95	6170	9500	-25	650	637	1456
3:37.1	282.5	41010	24.66	6170	9500	-25	650	647	1486
3:49.0	278.0	42000	23.39	6140	9500	-24	655	660	1522
3:59.0	267.0	43070	21.73	6150	9500	-21	650	669	1548
4:10.0	255.5	44075	20.36	6150	9460	-21	650	679	1576
4:25.0	258.0	44980	19.79	6120	9450	-20	645	692	1612
4:37.9	251.0	46000	18.75	6150	9450	-19	650	702	1646
4:49.9	238.5	47090	17.69	6170	9440	-21	645	712	1678
5:00.8	229.0	47965	16.77	6150	9400	-23	650	719	1702
5:17.7	227.0	48780	16.23	6150	9400	-26	650	731	1742
5:39.6	226.5	49810	15.60	6140	9380	-26	650	747	1793
5:57.4	221.0	50820	14.82	6140	9350	-25	650	758	1836
6:15.3	209.5	51800	13.58	6140	9320	-26	645	770	1873
7:18.0	208.5	52835	13.04	6160	9320	-26	650	804	2008
7:56.0	200.0	53745	12.25	6190	9300	-26	650	826	2089
8:20.8	194.0	54385	11.70	6190	9300	-27	645	839	2140

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (NON-AFTERBURNING)
WES 27286 LBS. W_S 24450 LBS. W_D 24000 LBS.

FLIGHT

IAS Knots	ALTITUDE Feet	F.A.T. °C	P_{t7} In. Hg	N_2 RPM	N_1 RPM	δ_E Avg. Deg.	W_f Engine Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
476.0	14790	27.0	63.26	9585	5940	-0.2	6300	CLOSED	24007
476.0	14725	27.0	63.54	9575	5940	-0.2	—		23950
462.5	14915	25.0	58.10	9350	5745	-0.2	—		23753
464.5	14960	25.0	58.14	9350	5745	-0.2	6320		23697
422.5	15715	18.0	47.80	9010	5420	-0.2	—		23433
426.0	15715	17.0	48.40	9010	5440	-0.2	4780		23365
378.0	15765	10.0	42.69	8770	5250	+0.2	—		23235
383.0	15695	10.0	42.89	8760	5240	+0.2	3960		23176
338.0	16235	6.0	35.70	8515	5000	+0.25	—		23075
336.0	16340	5.0	35.90	8515	5000	+0.25	3040		23026
305.5	16775	1.0	34.10	8465	4980	+0.43	—		22864
311.5	16660	1.0	34.20	8465	4980	+0.35	3020		22817
264.5	17115	-6.0	31.60	8355	4880	+0.80	—		22716
277.0	17235	-5.0	31.90	8365	4900	+0.85	2720		22656
225.5	17150	-8.0	28.05	8150	4695	+1.45	2275		22545
187.5	17275	-11.0	26.00	8015	4430	+2.65	1985		22353
173.0	17405	-13.0	26.00	8015	4555	+3.00	—		22179
4265	715	40	63.39	8980	5230	0	5550	CLOSED	25750
377.0	755	31	64.25	8675	4950	+0.1	4280		25500
324.0	625	26	5920	8465	4750	+0.45	3600		25300
227.5	930	22	40.07	8080	4110	+1.65	2438		24600

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (AFTERBURNING)
WES 27286 LBS. W_S 24450 LBS. W_D 24000 LBS.

FLIGHT

IAS Knots	ALTITUDE Feet	F_δA.T. C	Pt7 In. Hg	N_2 RPM	N_1 RPM	δ_E Avg. Deg.	W_f Total Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
561.5	12715	41.0	76.30	9595	5840	0	—	CLOSED	25262
551.0	12730	41.0	74.82	9595	5840	0	—	CLOSED	25039
532.5	10785	41.0	75.00	9595	5840	0	33750	CLOSED	24853
506.0	13320	35.0	66.44	9455	5695	0	30600	CLOSED	24326

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (NON-AFTERBURNING)
WES 27312 LBS. W_S 23940 LBS. W_D 240000

FLIGHT

IAS Knots	ALTITUDE Feet	F.A.T. °C	Pt7 In. Hg	N2 RPM.	N1 RPM	δ_E Avg. Deg.	Wf Engine Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
355.5	30815	-10.0	38.22	9540	6090	.48	5,820	CLOSED	21816
342.0	31490	-13.0	31.50	9000	5620	.25	4,095	↓	21660
329.0	27430	-10.0	27.94	8555	5100	.50	—	↓	25882
302.0	28635	-16.0	26.00	8415	5030	.90	—	↓	25755
294.0	27630	-17.0	26.20	8435	5060	.90	—	↓	25692
278.5	27630	-20.0	25.24	8350	5000	1.13	—	↓	25505
259.5	28025	-22.0	24.06	8300	4990	1.48	—	↓	25340
243.0	28240	-26.0	23.78	8300	4990	1.60	—	↓	24983
221.0	28485	-29.0	23.60	8260	4940	2.00	—	↓	24748
324	30420	-9.0	28.71	8790			2,925	CLOSED	26,200
312	30320	-10.0	27.66	8700			3,025	↓	25,600
289.5	29990	-14.5	26.00	8600			2,102	↓	25,250
270	29830	-17.0	23.88	8420			2,663	↓	24900
251	29940	-21.0	23.07	8390			2,190	↓	24800
232.5	30270	-25.0	22.97	8330			2276	↓	24470

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (AFTERBURNING)
WES 27312 LBS. W_S 23940 LBS. W_D 24000 LBS.

FLIGHT

IAS Knots	ALTITUDE Feet	F.A.T. °C	Pt7 In. Hg	N_2 RPM	N_1 RPM	δ_E Avg. Deg.	Wf Total Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
475.0	29895	19.0	51.15	9620	6040	1.05	25,180	CLOSED	24480
460.0	29300	14.0	47.60	9520	5970	1.53	—	CLOSED	23610
461.0	29250	14.0	47.80	9520	5960	1.53	—	CLOSED	23426
445.0	29860	10.0	44.52	9400	5840	1.53	21,920	CLOSED	22277

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (NON-AFTERBURNING)
WES 27286 LBS. W_S 23670 LBS. W_D 24000 LBS.

FLIGHT

IAS Knots	ALTITUDE Feet	F.A.T. -°C	P_{t7} In. Hg	N_2 RPM	N_1 RPM	δ_E Avg. Deg.	W_f Engine Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
289.0	39370	-12	25.19	9390	6010	+1.05	3125	CLOSED	23430
280.0	39310	-16	23.00	9100	5770	+0.75	2760		23250
258.5	39710	-19	19.20	8715	5430	—	2085		23075
241.0	40310	-23	18.12	8695	5400	—	1950		22960
202.5	40630	-31	17.54	8730	5510	—	2080		22700
181.5	40,715	-34	17.67	8880	5685	—	2295		22550

FLIGHT

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (AFTERBURNING)
WES 27286 LBS. W_S 23670 LBS. W_D 24000 LBS.

IAS Knots	ALTITUDE Feet	F.A.T. °C	P_{t7} In. Hg	N_2 RPM	N_1 RPM	δ_E Avg. Deg.	W_f Total Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
359.5	37650	+1.0	31.11	9445	5970	2.25	—	CLOSED	25161
368.0	37495	+4.0	31.90	9445	5960	2.25	—		25032
370.0	37585	+5.0	31.90	9455	5980	2.25	12,120		24939
348.0	38890	+0.0	28.44	9320	5890	2.25	—		24656
353.0	38750	+1.0	28.96	9320	5890	2.25	—		24570
355.5	38790	+1.0	29.14	9320	5890	2.25	—		24389
340.0	39310	−1.0	26.80	9180	5745	2.15	—		23700
342.5	39220	−1.0	27.08	9180	5745	2.15	12,840		23634

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (AFTERBURNING)
WES 27312 LBS. W_S 23400 LBS. W_D 24000 LBS.

FLIGHT

IAS Knots	ALTITUDE Feet	F.A.T. $^{\circ}$C	P_{t7} In. Hg	N_2 RPM	N_1 RPM	δ_E Avg. Deg.	W_f Total Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
242.0	48660	-18.0	16.78	9390	6090	3.65		CLOSED	23196
257.5	49620	-16.0	17.46	9420	6110	3.75		CLOSED	22892
261.0	49520	-16.0	17.40	9320	6020	3.25		CLOSED	22500
259.5	49450	-17.0	16.98	9220	5890	3.98		CLOSED	22204

TEST DATA CORRECTED FOR INSTRUMENT ERROR FLIGHT
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (~~AFTERBURNING~~) NON A/B
WES 27286 LBS. W_S 23550 LBS. W_D 24000 LBS.

IAS Knots	ALTITUDE Feet	F.A.T. °C	P_{t7} In. Hg	N_2 RPM	N_1 RPM	δ_E Avg. Deg.	Wf Total Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
253.5	44225	-20	19.87	9325	5990	+1.35	2625	CLOSED	23667
246.5	44590	-21	19.20	9255	5940	+1.13	2534	↓	23522
232.5	44730	-25	16.97	8945	5685	+1.40	2062	↓	23393
220.0	45060	-27	16.67	8970	5735	+1.65	2108	↓	23278
211.5	45260	-30	16.77	9070	5840	+1.98	2195	↓	23080
200.0	45385	-33	16.48	9060	5850	+2.18	2162	↓	22885

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LEVEL FLIGHT (AFTERBURNING)
WES 27286 LBS. W_S 23555 LBS. W_D 24000 LBS.

FLIGHT

IAS Knots	ALTITUDE Feet	F.A.T. °C	P_{t7} In. Hg	N_2 RPM	N_1 RPM	δ_E Avg. Deg.	W_f Total Lbs/Hr.	BLEED-VALVE	WEIGHT Lbs.
309.5	43880	-7	23.29	9400	6010	+3.00	12,530	CLOSED	24691
309.0	44135	-6	22.50	9305	5890	+3.15	—	CLOSED	24258
295.5	44650	-10	20.76	9130	5790	+3.00	10,930	CLOSED	23852

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: STRAIGHT AHEAD STALLS (1"G")

TIME SECONDS	IAS Knots	ALTITUDE Feet	N_Z "G"	WEIGHT Lbs			
0	144.0	24720					
10	143.0	24605	1.0	21680	BUFFET		
20	141.0	24485					
30	134.5	24240					
40	129.5	24090	0.95	21650	RECOVERY		
50	125.5	23910					
0	165.5	36835					
4	163.5	36825	1.0	23570	BUFFET		
8	162.0	36815					
114	125.5	33285					
116	123.5	33050	0.90	23510	HEAVY BUFFET-RECOVERY		
118	127.5	32750					
0	164.0	37115					
4	162.5	37100	1.0	23260	BUFFET		
8	161.0	37080					
32	148.5	36650					
36	146.5	36570	1.0	23240	HEAVY BUFFET		
40	145.0	36485					
120	125.5	32910					
122	125.5	32780	0.98	23220	RECOVERY		
124	126.5	32640					

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: (ACCELERATED STALLS)

TIME Seconds	IAS Knots	ALTITUDE Feet	N_Z "G"	WEIGHT Lbs				
0	201.0	21525						
2	201.0	21390	1.75	25170	BUFFET			
4	200.5	21270						
16	192.5	19940						
18	191.0	19580	2.30	25160	RECOVERY			
20	191.0	19220						
0	261.5	21170						
1	261.5	21035	2.40	25530	BUFFET			
2	261.5	20895						
3	262.0	20755						
11	261.5	19240						
13	261.5	18690	3.32	25520	RECOVERY			
15	257.5	18200						
0	329.0	21370						
1	331.0	21260						
2	333.0	21110	2.50	25750	BUFFET			
3	335.0	20915						
15	348.5	18010						
17	345.0	17450	4.30	25740	RECOVERY			
19	344.0	16810						
0	404.5	20450						
1	406.5	20170	4.0	25940	BUFFET			
2	409.5	19820						
3	411.0	19500						
5	415.5	18760	4.30	25930	RECOVERY			
7	422.0	18010						

NOTE: The accelerated stalls at 20,000 feet were flown by a pilot other than the phase pilot; hence, the buffet call-outs are subject to a different criteria.

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: (ACCELERATED STALLS)

TIME Seconds	IAS Knots	ALTITUDE Feet	N_Z "G"	WEIGHT Lbs			
0	184.5	37360					
2	183.5	37175	2.10	23580	RECOVERY		
4	179.5	37010					
0	227.0	37995					
2	223.0	37785	2.30	23640	RECOVERY		
4	219.5	37570					
0	308.5	34590					
10	314.0	33950	2.45	22470	BUFFET		
20	317.0	32875					
28	322.5	31580					
30	324.0	31270	3.15	22450	RECOVERY		
32	326.5	30915					
0	399.0	30425					
2	400.0	30220	3.13	23730	RECOVERY		
4	400.5	29970					
0	392.5	32025					
5	406.0	30685	2.78	22590	RECOVERY		
6	409.0	30395					
0	419.5	31930					
6	426.5	30130	3.85	22040	RECOVERY		
7	426.5	29770					
0	423.0	31680					
10	442.5	29555	3.25	22830	RECOVERY		
16	452.5	28220					

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: (ACCELERATED STALLS)

TIME Seconds	IAS Knots	ALTITUDE Feet	N_Z "G"	WEIGHT Lbs				
0	457.5	29750						
4	468.0	28790	2.90	21750		RECOVERY		
6	472.0	28340						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: (ACCELERATED STALLS)

TIME Seconds	IAS Knots	ALTITUDE Feet	N_Z "G"	WEIGHT Lbs	
0	172.5	43555			
2	172.0	43535	1.10	24150	BUFFET
4	171.5	43415			
34	169.5	42255			
36	169.5	42110	1.35	24120	RECOVERY
38	171.5	41970			
0	193.0	44295			
2	192.0	44185	1.30	24430	BUFFET
4	192.0	44065			
28	204.0	41070			
30	203.0	40790	1.80	24410	RECOVERY
32	202.0	40455			
0	236.5	43980			
2	238.5	43660	1.65	22790	BUFFET
4	239.0	43580			
12	241.0	43100			
14	242.0	42930	1.90	22780	RECOVERY
16	243.0	42620			
0	256.0	45300			
2	255.5	45190	1.70	22420	BUFFET
4	255.5	45130			
22	260.5	43080			
24	262.5	42630	2.60	22400	RECOVERY
28	265.5	41810			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: (ACCELERATED STALLS)

TIME Seconds	IAS Knots	ALTITUDE Feet	N_Z "G"	WEIGHT Lbs.				
0	258.0	45090						
6	257.5	44840	1.60	22900	BUFFET			
10	255.5	44700						
28	249.5	43980						
30	249.0	43720	1.99	22870	RECOVERY			
32	250.0	43445						
0	321.5	41250						
2	334.0	40315	2.65	23660	BUFFET			
4	338.0	39475						
18	355.0	37270						
20	358.0	36785	3.14	23590	RECOVERY			
22	361.0	36250						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: (ACCELERATED STALLS)

TIME Seconds	IAS Knots	ALTITUDE Feet	N_Z "G"	WEIGHT Pounds				
0	240	48210						
2	240	48160	1.70	24470	RECOVERY			
4	238	48130						
0	216.5	48150						
2	214.5	48080	2.20	24450	RECOVERY			
4	211.5	47845						
0	195.0	46910						
2	192.5	46700	1.85	24090	RECOVERY			
4	190.0	46425						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: RANGE MISSION - POINT INTERCEPT

TIME	IAS Knots	ALTITUDE Feet	F.A.T. °C	F.U. Lbs.	T_{t7} °C	A/B	P_{t7}/P_{t0}	
9:16:27	147.0	2230	+20	466	590	ON	2.29	
9:17:45	443.0	10090	38	1186	590		2.15	
9:18:34	426.5	20040	30	1564	600		2.24	
9:18:45	399.0	23245	24	1640	600		2.32	
9:19:14	347.0	30330	4		620		2.52	
9:20:05	286.0	39275	-23		650		2.73	
9:22:54	226.5	49800	-26		650		2.70	
9:23:35	214.0	51300	-26		660		2.70	
9:23:49	206.5	51780	-26		660		2.70	
9:24:08	202.0	52315	-28		660		2.71	
9:24:38	205.5	52485	-28		660		2.70	
9:27:40	241.5	48780	-21		650		2.66	
9:28:40	231.0	49840	-23		660		2.69	
9:29:30	221.0	50920	-26		660		2.70	
9:29:36	221.5	51000	-26		660	OFF	2.58	
9:30:00	232.0	49800	-25		580		2.35	
9:30:20	229.0	48860	-26	3480	460		1.61	
9:31:31	240.0	44750	-30		270		1.27	
9:32:55	259.5	39665	-34		190		1.20	
9:34:44	260.5	35740	-30		150		1.20	
9:36:00					420		2.15	
9:40:00					400		2.02	
9:45:00					410		2.10	
9:49:00					400		2.08	
9:53:00					400		2.08	
9:57:00					400		2.07	
10:01:00					400		2.04	
10:05:00					400		2.06	
10:10:00					400		2.04	
10:20:00					400		2.05	
10:25:32	247.0	34840	-31	5510	390		2.04	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: RANGE MISSION - AREA INTERCEPT

TIME	IAS Knots	ALTITUDE Feet	F.A.T. °C	F.U. Lbs.	T_{t7} °C	A/B	P_{t7}/P_{t0}
10:34:17	154.0	2070	+25	377	590	OFF	2.21
10:35:53	329.5	4850	+30	576	580	↓	2.20
10:37:03	358.5	9710	+24	734	590	↓	2.25
10:38:13	352.0	14865	+16	860	600	↓	2.32
10:39:23	325.0	20230	+3	984	600	↓	2.45
10:42:10	293.0	29820	-14	1225	610	↓	2.60
10:47:07	256.5	39685	-28	1540	630	↓	2.62
10:48:19	232.5	41645	-28	1597	635	↓	2.76
10:51:51	252.0	42030	-23	1770	550	↓	2.44
10:55:02	250.0	41240	-25	1905	550	↓	2.49
10:57:00	252.0	41310	-26	2000	560	↓	2.52
11:00:40	254.5	41790	-25	2155	560	↓	2.50
11:02:20	260.5	41585	-23	2225	550	↓	2.42
11:04:00	254.5	41720	-26	2295	540	↓	2.41
11:07:00	253.5	41565	-26	2415	530	↓	2.44
11:16:15	253.5	41735	-28	2785	540	↓	2.44
11:17:07	253.0	41605	-28	2825	540	↓	2.46
11:22:19	256.5	41495	-26	2955	590	ON	2.46
11:22:58	269.5	42905	-22	3184	620	↓	2.69
11:23:40	252.5	45940	-21	3321	640	↓	2.70
11:24:20	229.5	48660	-23	3439	650	↓	2.70
11:25:00	228.5	50280	-23	3548	650	↓	2.70
11:26:38	194.0	54425	-30	3783	650	↓	2.37
11:27:00	212.5	53600	-28	3794	645	OFF	2.76
11:27:39	231.5	49150	-22	3814	645	↓	2.76
11:28:00	229.5	48660	-23	3837	640	↓	2.57
11:28:33	240.0	47540	-23	3891	650	ON	2.30
11:30:10	215.5	47985	-29	4157	640	ON	2.20
11:31:40	269.5	43345	-21	4308	450	OFF	1.89
11:34:48	252.5	41485	-25	4418	540	↓	2.66
11:43:30	254.5	41685	-26	4759	520	↓	2.40
11:51:00	259.5	41365	-27	5060	510	↓	2.40
11:59:40	266.0	40855	-26	5403	500	↓	2.34
12:04:20	258.0	41200	-27	5579	370	↓	1.31

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: TURN CAPABILITIES
(DIRECTIONAL HEADING INSTRUMENT INOPERATIVE)

IAS Knots	ALTITUDE Feet	N_Z "G"	WEIGHT Lbs.	P_{T7} In. HG	F.A.T. °C	FLIGHT NO.
389.0	34685	1.55	24216	37.31	-9	14
362.0	35065	1.65	23053	34.73	-12	14
329.5	34970	2.17	22129	32.10	-19	14
327.5	34430	2.40	24515	32.53	-19	15
396.0	35740	1.60	23566	37.20	-8	15
365.0	34060	2.05	24669	36.77	-14	23
299.5	44165	1.14	25012	22.20	-11	13
265.0	44125	1.51	24207	20.40	-15	13
244.5	44415	1.67	23766	19.15	-20	13
225.5	44555	1.60	23368	18.19	-25	13
298.5	44425	1.21	24782	22.48	-16	16
272.5	43210	1.55	24281	21.57	-20	16
306.5	45040	1.20	22757	22.70	-11	16
284.5	45155	1.17	24509	20.20	-17	18
263.0	44630	1.35	23758	20.11	-20	18
235.5	49400	1.05	24663	15.96	-21	17
226.0	50215	1.19	24248	16.00	-23	17
246.0	48310	1.08	22808	17.34	-21	17

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: DESCENT (SPEED BRAKES OPEN)

IAS Knots	ALTITUDE Feet	F.A.T. Deg. C	P_{t7} In. HG	TIME Minutes	WEIGHT Lbs.	N_2 RPM	
230.5	46640	-27	9.70	0	22566	7900	
237.0	45530	-27	9.15	.167	22564	7720	
242.5	44540	-27	9.15	.298	22562	7660	
249.5	43310	-26	9.30	.465	22560	7610	
251.0	42210	-26	9.40	.630	22558	7560	
256.5	41000	-26	9.70	.797	22556	7510	
265.0	39790	-26	10.00	.930	22554	7450	
270.5	38600	-28	10.35	1.062	22553	7400	
277.0	37620	-29	10.65	1.162	22551	7360	
282.0	36750	-30	11.00	1.262	22550	7330	
289.0	35630	-29	11.35	1.360	22549	7340	
292.5	34550	-28	11.65	1.460	22548	7310	
300.5	33740	-27	11.90	1.527	22547	7310	
307.0	32380	-25	12.55	1.627	22546	7280	
314.0	31350	-23	12.95	1.693	22545	7270	
321.5	30260	-21	13.35	1.760	22544	7260	
328.0	29110	-19	13.75	1.825	22543	7260	
336.5	27930	-16	14.20	1.892	22542	7260	
348.0	26660	-13	14.60	1.958	22542	7260	
359.0	25390	-11	15.20	2.025	22541	7240	
366.5	24100	-7	15.80	2.092	22540	7260	
375.5	22760	-2	16.40	2.158	22539	7260	
382.5	21420	0	17.05	2.225	22538	7260	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: DESCENT (SPEED BRAKES OPEN)

IAS Knots	ALTITUDE Feet	F.A.T. Deg. C	P_{t7} In. HG	TIME Minutes	WEIGHT Lbs.	N2 RPM	
197.5	39490	-37	9.35	0	22100	7460	
198.5	38660	-40	9.50	.296	22096	7400	
198.0	37680	-43	9.70	.728	22091	7320	
191.0	36640	-45	9.90	1.093	22086	7280	
191.5	35680	-44	10.10	1.360	22083	7260	
191.0	34700	-43	10.35	1.625	22080	7250	
193.5	33690	-42	10.70	1.857	22077	7230	
194.5	32790	-41	10.95	2.057	22074	7210	
200.0	31660	-38	11.30	2.288	22071	7210	
203.0	30810	-37	11.60	2.455	22069	7190	
210.5	29710	-34	12.00	2.688	22066	7190	
211.5	28810	-32	12.35	2.920	22063	7160	
215.0	27750	-29	12.75	3.153	22060	7160	
218.5	26740	-27	13.15	3.352	22058	7150	
224.0	25830	-25	13.55	3.517	22056	7140	
228.0	24770	-22	14.05	3.717	22054	7130	
234.0	23730	-19	14.50	3.917	22052	7120	
241.0	22780	-16	15.00	4.083	22050	7120	
244.0	21840	-13	15.40	4.248	22048	7110	
247.5	20890	-10	15.75	4.415	22046	7110	
253.0	19790	-8	16.45	4.582	22044	7100	
259.5	18850	-6	16.95	4.715	22042	7090	
261.5	17880	-5	17.50	4.847	22040	7070	
268.5	16830	-2	18.20	4.980	22038	7060	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: DESCENT (SPEED BRAKES OPEN)

IAS Knots	ALTITUDE Feet	F.A.T. Deg. C	P_{t7} In. HG	TIME Minutes	WEIGHT Lbs.	N2 RPM	
229.0	40635	-23	9.00	.930	23504	7540	
236.0	39615	-23	9.30	1.097	23502	7520	
242.0	38570	-23	9.70	1.262	23500	7520	
245.5	37600	-23	9.90	1.395	23499	7450	
250.0	36620	-23	10.20	1.528	23497	7410	
259.5	35620	-21	10.10	1.662	23495	7410	
261.5	34585	-20	10.80	1.812	23493	7400	
266.0	33660	-18	11.15	1.945	23492	7380	
274.0	32750	-17	11.50	2.077	23490	7360	
274.0	31710	-16	11.75	2.243	23488	7360	
279.0	30775	-15	12.25	2.358	23487	7360	
287.0	29670	-14	12.75	2.492	23485	7320	
291.0	28675	-13	13.15	2.608	23483	7310	
305.0	27670	-12	13.65	2.707	23482	7290	
311.0	26800	-12	14.10	2.807	23481	7260	
321.5	25730	-11	14.55	2.923	23479	7230	
323.5	24810	-11	14.95	3.040	23478	7210	
326.5	23830	-11	15.30	3.157	23477	7190	
330.0	22835	-10	15.75	3.255	23475	7170	
330.0	21765	-8	16.20	3.355	23474	7160	
330.0	20725	-7	16.75	3.438	23473	7160	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: DESCENT (SPEED BRAKES CLOSED)

IAS Knots	ALTITUDE Feet	F.A.T. Deg. C	P_{t7} In. HG	TIME Minutes	WEIGHT Lbs.	N_2 RPM	
326.0	32005	-21	14.45	0	22411	7410	
336.0	31015	-19	14.45	.117	22409	7350	
339.5	30040	-15	14.50	.284	22407	7270	
342.0	28960	-12	14.65	.467	22405	7260	
349.5	27930	-10	15.05	.617	22403	7240	
357.5	26910	-7	15.50	.767	22401	7350	
363.5	25870	-6	15.80	.900	22399	7210	
371.5	24830	-4	16.45	1.017	22398	7210	
380.0	23830	0	16.85	1.117	22397	7200	
393.0	22815	+2	17.55	1.217	22395	7200	
398.0	21880	+4	18.15	1.315	22394	7200	
401.5	20855	+5	18.60	1.432	22393	7190	
408.0	19930	+7	19.20	1.532	22391	7170	
413.0	18830	+10	19.80	1.649	22390	7160	
418.5	17795	+11	20.45	1.749	22389	7160	
427.0	16750	+15	21.15	1.832	22388	7160	
438.0	15910	+15	21.70	1.899	22387	7150	
438.0	15360	+18	22.05	1.965	22386	7150	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: DESCENT (SPEED BRAKES CLOSED)

IAS Knots	ALTITUDE Feet	F.A.T. Deg. C	P_{t7} In. HG	TIME Minutes	WEIGHT Lbs.	N_2 RPM	
197.0	36225	-32	10.00	0	22101	7320	
195.0	35255	-33	10.27	.515	22095	7270	
200.5	34270	-33	10.64	.947	22089	7240	
199.5	33290	-38	10.96	1.643	22080	7140	
202.0	32285	-38	11.26	1.994	22076	7130	
212.0	31285	-37	11.66	2.243	22073	7110	
217.5	30320	-34	12.06	2.542	22069	7080	
218.0	29330	-32	12.30	2.942	22064	7080	
220.0	28330	-30	12.64	3.307	22059	7060	
227.0	27360	-28	13.04	3.590	22056	7040	
231.5	26365	-25	13.48	3.905	22052	7020	
233.5	25365	-22	13.92	4.222	22048	7010	
241.0	24350	-21	14.42	4.472	22045	7010	
249.0	23390	-17	15.01	4.705	22042	6990	
257.5	22355	-15	15.55	4.972	22039	6980	
259.0	21360	-13	16.05	5.237	22035	6960	
260.5	20415	-11	16.53	5.487	22032	6940	
267.5	19375	-9	17.16	5.737	22029	6920	
271.5	18410	-7	17.74	5.937	22026	6910	
276.0	17455	-6	18.32	6.120	22024	6890	
287.0	16430	-4	19.00	6.285	22022	6870	
293.5	15445	-1	19.68	6.452	22020	6860	
300.5	14450	+1	20.36	6.618	22018	6850	
305.5	13470	+4	21.00	6.784	22016	6820	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: DESCENT (SPEED BRAKES CLOSED)

IAS Knots	ALTITUDE Feet	F.A.T. Deg. C	Pt7 In. HG	TIME Minutes	WEIGHT Lbs.	N2 RPM
260.0	40910	-21	10.55	0	22236	7660
259.5	39670	-21	10.20	.432	22231	7520
261.0	38630	-21	10.30	.795	22226	7480
257.5	37650	-23	10.45	1.328	22219	7410
261.5	36740	-26	10.75	1.627	22216	7370
262.5	35715	-28	11.05	2.158	22209	7310
258.5	34730	-28	11.30	2.657	22203	7270
259.5	33730	-27	11.50	2.990	22199	7260
256.5	32705	-27	11.85	3.288	22195	7210
260.5	31795	-27	12.25	3.555	22192	7210
264.5	30790	-27	12.65	3.855	22188	7160
264.5	29785	-27	12.95	4.252	22183	7130
262.5	28835	-26	13.25	4.618	22178	7110
260.5	27840	-26	13.55	5.085	22173	7090
260.5	26825	-23	13.75	5.517	22168	7060
258.5	25985	-21	14.10	5.883	22163	7040
255.5	24850	-20	14.50	6.282	22158	7020
256.0	23850	-18	14.95	6.613	22154	7010
255.5	22755	-17	15.40	6.913	22150	6990
257.5	21890	-15	15.85	7.213	22147	6980
257.5	20870	-12	16.35	7.577	22142	6960
256.5	19895	-11	16.80	7.877	22138	6920
258.0	18835	-10	17.45	8.210	22134	6900
259.5	17740	-7	18.05	8.592	22130	6870
258.5	16715	-5	18.45	8.907	22125	6830
256.5	15735	-5	19.20	9.240	22121	6800
256.5	14725	-1	19.80	9.572	22117	6760

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: DESCENT (SPEED BRAKES CLOSED)

IAS Knots	ALTITUDE Feet	F.A.T. Deg. C	P_{t7} In. HG	TIME Minutes	WEIGHT Lbs.	N_2 RPM	
294.0	30870	-27	13.08	1.046	23081	7210	
300.0	29870	-25	13.48	1.263	23078	721.0	
306.5	28835	-22	13.97	1.461	23076	7210	
309.0	27810	-19	14.37	1.628	23074	7200	
318.0	26790	-17	14.81	1.795	23072	7180	
325.5	25790	-15	15.31	1.976	23069	7170	
330.5	24830	-11	15.60	2.110	23068	7170	
336.0	23810	-9	16.10	2.245	23066	7160	
344.0	22835	-6	16.63	2.358	23065	7160	
352.5	21860	-4	17.16	2.475	23063	7160	
358.5	20930	0	17.74	2.591	23062	7160	
365.5	19910	+2	18.32	2.708	23060	7160	
375.0	18895	+5	18.96	2.825	23059	7160	
380.5	17820	+9	19.59	2.958	23057	7160	
385.0	16850	+11	20.18	3.075	23056	7160	
391.0	15940	+14	20.81	3.173	23054	7150	
398.0	14945	+15	21.44	3.273	23053	7130	
403.5	13920	+18	22.17	3.373	23052	7120	
410.5	12860	+20	22.95	3.473	23051	7120	

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITHOUT DRAG CHUTE
P_a = 27.660 t_a = 14°C GROSS WEIGHT 21920
WIND 19 MPH AT 10 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	7450	20							
1.0	7220	10							
2.0	7010	6							
2.87	6850	T. D.							
4.0	6640								
6.0	6260								
8.0	5920								
10.0	5580								
12.0	5250								
14.0	4940								
16.0	4630								
18.0	4330								
20.0	4060								
22.0	3800								
24.0	3570								
26.0	3360								
28.0	3180								
30.0	3040								
33.0	2860								
36.0	2740								
39.0	2690								
40.0	2690								

NOTE: No data is available from 50 foot obstacle to 20 foot height on this landing.

CONFIDENTIAL

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITHOUT DRAG CHUTE
P_a = 27.650 t_a = 15°C GROSS WEIGHT 21840
WIND 20.7 MPH AT 15 DEGREES FROM NOSE

TIME Seconds	DISTANCE - Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	
0	-925	17	0	-1000	30	32	4385		
.39	-1035	20	0.5	-900	25	34	4583		
1.03	-1210	25	1.0	-790	20	36	4760		
1.57	-1410	31	1.5	-670	12	38	4920		
2.35	-1575	35	2.0	-560	8	40	5060		
3.00	-1760	40	2.5	-450	5	42	5185		
3.60	-1935	45	3.0	-330	5	44	5290		
4.11	-2080	50	3.5	-230	4	46	5380		
5.15	-2390	60	4.0	-120	3	48	5450		
5.76	-2700	71	4.5	0	2	50	5510		
			5.5	+197	1	52	5560		
			6.0	300	T.D.	53	5580		
			7.0	500		54	5590		
			8.0	695		55	5600	(STOP)	
			9.0	880					
Above is Fairchild data.			10.0	1065	Above is Photo-grid data.				
			12.0	1430					
			14.0	1780					
			16.0	2140					
			20.0	2805					
			24.0	3408					
			28.0	3930					
			30.0	4170					

This landing data was obtained from two (2) supplementary camera systems (Photo-grid and Fairchild cameras). Time is measured backwards to 50 foot height on the Fairchild system and forward to full stop on the Photo-grid. Overlap point is approx. zero time. (Photo-grid time 0.5 sec. = Fairchild time 0 sec.)

CONFIDENTIAL

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITHOUT DRAG CHUTE
P_a = 27.695 t_a = 14.5°C GROSS WEIGHT 21950
WIND CALM MPH AT — DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-1020	13						
0.5	-900	12						
1.0	-780	7						
2.0	-510	4						
2.5	-390	3						
3.5	-140	T.D.						
4.5	+100							
6.0	440							
8.0	900							
10.0	1340							
12.0	1760							
14.0	2180							
16.0	2560							
18.0	2970							
20.0	3350							
22.0	3700							
24.0	4040							
26.0	4360							
28.0	4650							
30.0	4910							
32.0	5130							
34.0	5330							
36.0	5520							
38.0	5660							
40.0	5790							
42.0	5890							
44.0	5960							

NOTE: No data is available from 50 foot obstacle to 13 foot height on this landing.

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITH DRAG CHUTE
P_a = 27.850 t_a = 11°C GROSS WEIGHT 21718
WIND CALM MPH AT —— DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-1100	40
1.0	-880	33
2.0	-630	23
3.0	-390	13
4.0	-160	6
5.0	+70	1
5.31	150	T.D.
6.5	400	
8.0	750	
9.5	1080	
12.0	1570	
14.5	2030	
16.5	2370	
19.0	2730	
22.0	3090	
25.0	3370	
27.5	3540	
30.0	3650	
32.5	3700	
33.5	3710	

NOTE: No data is available prior to 40 foot height on this landing.

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITH DRAG CHUTE
Pa 27.660 ta 12°C GROSS WEIGHT 21800
WIND 5 MPH AT 30 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-885	19	0	-700	25
.50	-1080	24	0.5	-570	20
1.19	-1235	28	1.0	-450	18
1.64	-1380	32	1.5	-330	15
2.07	-1520	37	2.0	-200	8
2.55	-1660	43	2.5	-80	5
3.12	-1840	50	3.0	+20	3
3.58	-1985	56	3.5	160	2
3.97	-2120	62	4.0	290	2
4.61	-2325	73	4.5	400	1
			5.0	515	T.D.
			6.0	745	
			8.0	1190	
Above is Fairchild data.			10.0	1600	Above is Photo-grid data.
			12	2000	
			14	2363	
			16	2700	
			18	3032	
			20	3320	
			22	3578	
			24	3832	
			26	4010	
			28	4180	
			30	4315	
			32	4415	
			34	4493	
			36	4538	
			37	4550	

This landing data was obtained from two (2) supplementary camera systems (Photo-grid and Fairchild cameras). Time is measured backwards to 50 foot height on the Fairchild system and forward to full stop on the Photo-grid. Overlap point is approx. zero time. (Photo-grid time 0.8 sec. = Fairchild time 0 sec.)

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITH DRAG CHUTE
P_a = 27.710 t_a = 21 °C GROSS WEIGHT 22353
WIND CALM MPH AT — DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-900	16	0	-1000	25	26	3725	
0.84	-1195	20	0.5	-880	18	28	3904	
1.81	-1545	32	1.0	-750	10	30	4050	
2.34	-1740	40	1.5	-615	8	32	4180	
2.97	-1975	50	2.0	-500	6	34	4276	
3.52	-2190	60	2.5	-375	5	36	4350	
			3.0	-255	4	38	4405	
			3.5	-140	3	40	4435	
			4.0	-20	2	41	4437	
			4.5	+100	1			
			5.0	210	T.D			
			7.0	660				
			7.75	820				
Above is Fairchild data.			8.0	880	Above is Photo-grid data.			
			8.5	980				
			9.0	1090				
			10	1297				
			12	1690				
			14	2055				
			16	2400				
			18	2720				
			20	3022				
			22	3297				
			24	3525				

This landing data was obtained from two (2) supplementary camera systems (Photo-grid and Fairchild cameras). Time is measured backwards to 50 foot height on the Fairchild system and forward to full stop on the Photo-grid. Overlap point is approx. zero time. (Photo-grid time 0.4 sec ≅ Fairchild time 0 sec.)

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITH DRAG CHUTE
P_a = 27.780 t_a = 19°C GROSS WEIGHT 22,374
WIND CALM MPH AT —— DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-915	7	0	-990	12
.36	-1020	8	1.0	-750	6
1.19	-1265	13	1.5	-630	4
1.79	-1425	17	2.0	-510	4
2.38	-1615	21	3.25	-220	T.D.
3.10	-1830	25	4.5	+20	
3.45	-1935	28	6.5	460	
3.82	-2045	32	8.5	880	
4.21	-2165	36	10.5	1270	
4.53	-2265	40	13.0	1740	
4.88	-2370	45	15.5	2200	
5.22	-2475	50	18.0	2610	
5.59	-2590	54	20.5	2980	
5.91	-2695	60	23.0	3300	
			25.5	3560	
			28.0	3750	
Above is Fairchild data.			30.5	3900	Above is Photo-grid data.
			33.0	3980	
			35.0	4020	
			36.0	4030	

This landing data was obtained from two (2) supplementary camera systems (Photo-grid and Fairchild cameras). Time is measured backwards to 50 foot height on the Fairchild system and forward to full stop on the Photo-grid. Overlap point is approx. zero time. (Photo-grid time 0.3 sec. = Fairchild time 0 sec.)

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITH DRAG CHUTE
P_a = 27.690 t_a = 24 °C GROSS WEIGHT 21842
WIND 9.2 MPH AT 105 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-900	8	0	-1000	12	26	3280	
1.11	-1250	11	0.5	-880	10	28	3425	
2.07	-1565	14	1.0	-760	8	30	3553	
2.90	-1850	16	1.5	-650	5	32	3645	
3.75	-2130	18	2.0	-530	3	34	3706	
4.58	-2415	22	2.5	-420	1	36	3738	
5.47	-2720	27	3.0	-310	T.D.	36.5	3740	
5.96	-2890	33	3.5	-210				
6.65	-3130	42	4.0	-100				
			5	+110				
			6	315				
Above is	Fairchild	data.	7	520	Above	is Photo-grid data.		
			9	910				
			11	1280				
			13	1615				
			15	1940				
			17	2243				
			20	2650				
			22	2885				
			24	3095				

This landing data was obtained from two (2) supplementary camera systems (Photo-grid and Fairchild cameras). Time is measured backwards to 50 foot height on the Fairchild system and forward to full stop on the Photo-grid. Overlap point is approx. zero time. (Photo-grid time 0.4 sec. = Fairchild time 0 sec.)

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MAXIMUM BRAKING WITHOUT DRAG CHUTE
P_a = 27.495 t_a = 20.4°C GROSS WEIGHT 22708
WIND 9.3 MPH AT 30 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-11920	66
1	-11626	51
2	-11316	38
4	-10781	25
6	-10232	11
8	-9700	2
9	-9452	-2
9.06	-9439	T.D.
10	-9210	
11	-8976	
12	-8735	
14	-8259	
16	-7820	
18	-7427	
20	-7058	
22	-6717	
24	-6411	
26	-6143	
28	-5905	
30	-5703	
32	-5533	
34	-5403	
36	-5296	
37	-5254	
38	-5217	
39	-5195	
40	-5180	

Akeley camera was used to obtain this data.

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MILITARY BRAKING WITH DRAG CHUTE
P_a = 27.736 t_a = 1.0°C GROSS WEIGHT 21672
WIND ________ MPH AT 165 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-910	15	0	-990	25	32	4072	
.36	-1010	17	0.5	-850	15	34	4170	
.98	-1190	21	1.0	-730	10	36	4240	
1.83	-1410	24	1.5	-600	8	38	4290	
2.60	-1640	29	2.0	-480	7	39	4305	
3.34	-1860	38	2.5	-360	6	40	4315	
3.68	-1965	44	3.0	-250	5	41	4319	
4.02	-2070	50	3.5	-130	2			
4.33	-2170	56	4.0	0	1			
4.80	-2320	65	4.5	+110	T.D			
5.23	-2455	75	5.0	220				
			5.5	330				
			6.0	440				
			7.0	665				
			8.0	870				
Above is Fairchild data.			9.0	1090	Above is Photo-grid data.			
			10	1285				
			12	1685				
			14	2042				
			16	2378				
			18	2677				
			20	2970				
			22	3218				
			24	3440				
			26	3640				
			28	3805				
			30	3953				

This landing data was obtained from two (2) supplementary camera systems (Photo-grid and Fairchild cameras). Time is measured backwards to 50 foot height on the Fairchild system and forward to full stop on the Photo-grid. Overlap point is approx. zero time. (Photo-grid time 0.3 sec. Fairchild time 0 sec.)

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MAXIMUM BRAKING WITHOUT DRAG CHUTE
P_a = 27.600 t_a = 22°C GROSS WEIGHT 21443
WIND 6 MPH AT 20 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet	TIME Seconds	DISTANCE Feet	HEIGHT Feet			
0	6312	79	32	972				
1	6043	59	33	941				
2	5782	41	34	922				
3	5518	25	35	912				
4	5263	15						
5	5019	6.3						
6	4791	2.2						
6.5	4661	T.D.						
7	4560							
8	4334							
9	4059							
10	3840							
11	3638							
12	3439							
13	3234							
14	3035							
15	2843							
16	2663	Akeley camera was used to obtain this data.						
17	2501							
18	2336							
19	2175							
20	2028							
21	1888							
22	1768							
23	1648							
24	1541							
25	1434							
26	1346							
27	1259							
28	1182							
29	1116							
30	1062							
31	1011							

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MAXIMUM BRAKING WITH DRAG CHUTE
P_a = 27.640 t_a = 15.88 GROSS WEIGHT 21752
WIND 2.5 MPH AT 30 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet
0	-9679	89.8
1	-9389	61.9
2	-9106	40.5
3	-8809	23.0
4	-8535	16.1
5	-8267	9.6
6	-8000	8
7	-7742	4.7
8	-7487	1.5
8.5	-7364	T.D.
9	-7237	
10	-7001	
11	-6754	
12	-6523	
13	-6299	
14	-6083	
16	-5679	
18	-5323	
20	-5007	
22	-4738	
24	-4515	
25	-4416	
26	-4367	

Akeley camera was used to obtain this data.

CONFIDENTIAL

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: LANDINGS - MAXIMUM BRAKING WITH DRAG CHUTE
P_a = 27.640 t_a = 17.3°C GROSS WEIGHT 21480
WIND 9.5 MPH AT 20 DEGREES FROM NOSE

TIME Seconds	DISTANCE Feet	HEIGHT Feet						
0	-11149	118						
1	-10868	87						
2	-10604	63						
3	-10343	40.5						
4	-10101	24.6						
5	-9860	14.7						
6	-9613	8.3						
7	-9383	5.4						
8	-9162	1.6						
8.31	-9095	T.D.						
9	-8948							
10	-8731							
11	-8512							
13	-8222							
15	-7853							
17	-7553							
19	-7294							
21	-7088							
23	-6931							
25	-6821							
27	-6760							
29	-6746							

Akeley camera was used to obtain this data.

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: AIRSPEED CALIBRATION - TOWER FLY-BY
$t_a = +1.5^{\circ}C$ $P_a = 27.775$ In. HG

IAS Knots	ALTITUDE Feet	AIRCRAFT Ht. Ft.	ALT. GND. BLOCK
534.5	1675	264	2055
506.0	1645	167	2045
456.5	1740	150	2045
416.0	1780	125	2045
355.0	1875	117	2045
306.0	1920	111	2045
251.0	2000	128	2045
547.0	1525	158	2045
499.5	1645	113	2040
443.0	1730	120	2040

CONFIDENTIAL

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: AIRSPEED CALIBRATIONS - AIRCRAFT FLY-BY
(PACER AIRCRAFT F-100A, USAF NO. 53-1663)

PACER V_c	PACER H_c	IAS Knots	ALTITUDE Feet
293.0	35944	305.0	35240
		313.5	35145
		324.5	34995
		329.0	34860
		330.5	34790
		342.0	34645
		347.0	35340
		350.0	35505
		365.0	35650
		352.5	35700
		345.5	34940
		341.0	34760
		338.5	34570
		335.0	34575
		334.0	34535
		332.0	34545
		325.5	34945
		324.5	34995
		324.0	35000
		322.0	34980
		320.0	34990
		313.5	35040
		279.0	35465
		289.0	35430
		299.5	35350
		309.0	35230
		319.0	35075
		329.0	34860
		339.0	35370
		349.0	35380
		351.5	35605
		353.5	35650

CONFIDENTIAL

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: AIRSPEED CALIBRATION - PACER
(PACER AIRCRAFT F-100A, USAF NO. 53-1663)

PACER V_c	PACER H_c	PACER M	IAS Knots	ALTITUDE Feet			
244.0	20250	0.535	238.0	19990			
289.0	20110	0.630	282.0	19765			
325.0	20400	0.710	318.5	19915			
365.5	20700	0.785	357.0	20165			
409.0	21020	0.885	401.5	20420			
444.0	20730	0.950	433.5	20015			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: AIRSPEED CALIBRATION - PACER
(PACER AIRCRAFT JF-86E, USAF NO. 50-582A)

PACER V_c	PACER H_c	PACER	IAS Knots	ALTITUDE Feet		t_{ic} Deg. C	t_a Deg. C
203.0	10972	0.374	199.0	10725		+4.0	-2
233.3	10721	0.4275	228.5	10505		+6.0	-3
265.3	10930	0.489	259.5	10735		+9.0	-4
290.1	10847	0.532	283.5	10525		+11.0	-3
319.5	11172	0.588	312.5	10855		+14.0	-3
355.4	10838	0.649	348.0	10540		+18.0	-3
384.5	10804	0.700	376.5	10290		+21.0	-3
414.0	10844	0.752	407.0	10260		+25.0	-3
447.5	10576	0.806	437.0	9950		+29.0	-2
471.8	10744	0.850	464.0	10055		+32.5	-3.5
488.7	11126	0.885	479.0	10360		+36.0	-3.5

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: AIRSPEED CALIBRATIONS - PACER
(PACER AIRCRAFT F-100A, USAF NO. 53-1663)

PACER V_c	PACER H_c	IAS Knots	ALTITUDE Feet				
278.8	35341	272.5	34690				
318.5	36020	309.5	35180				
331.0	35599	323.0	34475				
373.0	37250	373.5	37185				
396.0	36505	397.0	36255				

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TEST
TEST: AIRSPEED CALIBRATION - LANDING CONFIGURATION - PACER
(PACER AIRCRAFT T-28, USAF NO. 49-1494)

PACER V_c	PACER H_c	F.A.T. Deg. C	IAS Knots	ALTITUDE Feet			
198.5	6230	14	192.5	6135			
207.5	5750	15	203.0	5615			
217.0	5300	18	211.0	5175			

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: AIRSPEED CALIBRATION - LANDING CONFIGURATION TOWER FLY-BY
t_a = +16°C P_a = 27.53° V_w = 12 KTS. S.W.

IAS Knots	ALTITUDE Feet		AIRCRAFT Ht.- Ft.	ALT. GND. BLOCK			
216.5	2255		90	2286			
228.0	2265		105				
237.5	2220		67.5				

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: THRUST RUN
DATE 13 FEBRUARY 1956 P_a 27.540 t_a 17.0

F_g Lbs.	P_{t7} In. Hg	N_2 RPM	W_f Eng. Lbs/Hr.	W_f A/B Lbs/Hr.	T_{t7} Degrees C		
13345	61.23	9515			620		
8545	60.18	9535			603		
11950	58.03	9325			560		
7820	57.31	9380			560		
7105	54.60	9230			525		
5750	49.14	8920			455		
4305	43.88	8570			360		
2835	37.94	8160			279		
1530	33.00	7510			224		
395	28.94	5445			240		
2285	35.58	7920			260		
2805	37.33	8130			275		
4960	46.78	8705			370		
6395	52.54	9020			460		
7210	54.78	9260			540		
7950	57.70	9425			580		
8345	59.15	9550			605		
13025	60.82	9550			620		

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: THRUST RUN
DATE 20 FEBRUARY 56 P_a 27.765 t_a 13

F_g Lbs.	P_{t7} In. Hg	N_2 RPM	W_f Eng. Lbs/Hr.	W_f A/B Lbs/Hr.	T_{t7} Degrees C		
8875	61.55	9520			610		
13680	62.42	9535			620		
440	29.29	5505			230		
7820	57.40	9280			540		
7310	54.90	9165			510		
5900	49.75	8850			445		
570	29.67	5950			230		
755	30.45	6490			220		
1105	31.65	6980			220		
1570	33.30	7450			220		
2745	37.28	8040			260		
4480	45.02	8515			320		
390	29.11	5425			220		
8470	59.93	9475			595		
395	29.14	5465			220		
12255	59.53	9320			565		
7570	56.45	9265			540		
2285	35.62	7900			255		
12615	59.98	9380			580		

TEST DATA CORRECTED FOR INSTRUMENT ERROR
F-102A, USAF NO. 53-1791
PHASE IV PERFORMANCE FLIGHT TESTS
TEST: THRUST RUN
DATE 6 APRIL 1956 P_a 27.610 t_a 26.0

F_g Lbs.	P_{t7} In. Hg	N_2 RPM	Wf Eng. Lbs/Hr.	Wf A/B Lbs/Hr.	T_{t7} Degrees C		
12815	60.14	9675					
11670	57.31	9465	7340	18950			
5995	49.75	9130	5100				
1565	32.75	7610	1550				
760	30.17	6610	1120				
8190	58.44	9650	7500				
1960	34.30	7910	1920				
5560	49.03	9000	5000				
955	30.90	6920	1180				
11985	58.19	9545	7050	19610			
3045	38.32	8365	2870				
6965	53.78	9330	6200				
5190	46.85	8935	4600				
1465	32.50	7550	1500				
680	29.93	6370	1000				
6425	52.40	9210	5800				
12500	59.57	9650	6910	19600			
3700	41.09	8570	3200				
440	29.04	5660	880				

APPENDIX III

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